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14. ABSTRACT: Flight competition report regarding the Knightsat, Nanosat-4 competition for the Air Force Research Laboratory. Knightsat, being a student ran organization, was split into three sub-systems, Attitude Determining Control Systems, Structures and Payload, and Power and Communications. Knightsat's primary goal is to obtain a stereo image of one point on the earth's surface. Knightsat's program manager and primary mentor was Dr Roger Johnson, who also coordinated assistance for the project with NASA's prototype shop and the Florida Space Institute. Knightsat's design has been approved and the integration of all of the sub-systems is under way.

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Knightsat

Flight Design Review
August 3, 2007

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1.0 Introduction

This report contains vital information regarding the involvement of the University of Central Florida in the University Nanosatellite-4 competition. The University Nanosatellite-4 (Nanosat) competition is a joint program between the Air Force Office of Scientific Research (AFOSR), the Air Force Research Laboratory's Space Vehicles Directorate (AFRL/VS), and the American Institute of Aeronautics and Astronautics. The purpose of this report is to give a complete and accurate analysis of the current University of Central Florida's Nanosat-4 status. It is also the purpose of this report to state the current mission, the mission objectives, updated projection of goals, the success criteria, the requirements, critical and conceptual designs and to present a reasonable timeline for the completion of the satellite.

1.1 History

The University of Central Florida first joined the Nanosat program in 2005. The original design for the Nanosat called for two satellites using laser-based on-orbit navigation and an autonomous, intelligent vehicle health management system. However, based on recent feedback from the Air Force in April 2006, the mission and objectives have been redesigned along with the satellite. The current satellite calls for an optical (Earth observing) payload which is capable of providing a resolution of at least one-hundred meters per pixel and a gravity gradient boom capable of stabilizing the satellite. The current bus is an isocagon, a polygon with twenty sides, and will contain a payload (a modified Cassegrain Telescope with CCD camera), power system, communications, Pulse Plasma Thrusters (PPT's), and a gravity gradient boom. An external view of the satellite is shown below in figure 1.

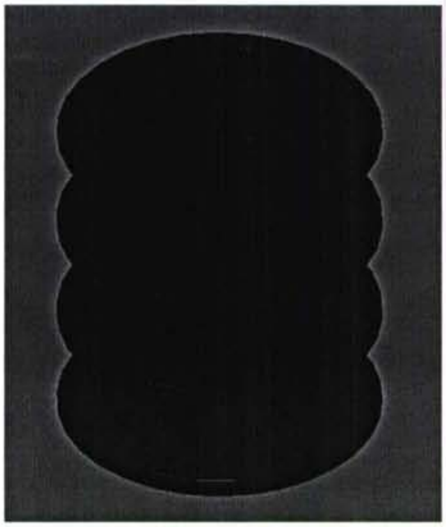


Figure 1: Peripheral Design of Satellite

2.0 Mission Objectives / Goals

The KnightSat mission is to develop a cost efficient satellite that utilizes a gravity gradient boom for stabilization and accuracy of the ground imaging system, which would be used for urban planning, disaster areas, or related missions. The primary and secondary objectives are listed as follows:

Primary Objective

- Provide at least two images of a point on the earth at better than 100 meters per pixel of an identifiable area over land and provide this image to a designated ground station.
- Requesting Sun-sync orbit at 300 to 700km for the imaging mission.

Secondary Objective:

- Demonstrate the University of Central Florida's Gravity Gradient Boom.
- Utilize Pulse Plasma Thrusters to De-Orbit the space-craft

2.1 Success Criteria

The mission of the Nanosat will be considered a success if all pertinent program deadlines are met;

- All requirements of the launch provider are met
- An acceptable evaluation score is achieved

- The primary mission of an imaging system using a modified Cassegrain telescope is a success
- The use of the UCF Gravity Gradient Boom is a success
- All relevant subsystems and components work successfully; and the overall health of the satellite is maintained.

3.0 Requirements

The requirements for the Nanosat as stated by the Air Force and the supervising director are to first meet all program deadlines and for each student to participate in all aspects of the design and implementation process. Furthermore, the design team is encouraged to meet the requirements of the launch provider, which include the system meeting the strength and material requirements; the system utilizing the materials and fasteners recommended; the system being built considering assembly and testing requirements; and the avoidance of all Phillips or flat-head fasteners in the system. The system must not contain sealed or pressured containers and must be launched in a "dead" (no battery charge) state. The system is also required to use NiCad (Nickel Cadmium) batteries and should make use of machined metal primary structures.

3.1 Mission Requirements

The mission of the Nanosat is to be designed to withstand launch and for the launch vehicle to survive in orbit without failure to structure, power, attitude control systems, and communication systems. This also includes the leaking of hazardous fluids and the releasing of anything from the Nanosat that could damage either the Launch Vehicle or cause injury to the ground handling crew. Nanosat is to be designed to fulfill NASA's requirements of structural integrity including, NSTS 1700.7B, *Safety Policy and Requirements for Payloads using the Space Transportation System*, and NASA-STD-5003, *Fracture Control Requirements for Payloads using the Space Shuttle.* The fundamental frequency for the Nanosat will be above 100 Hz given a fixed-base condition at the SIP. The structure must be designed to withstand an atmospheric temperature of 900K. The Nanosat structure design limit load factors are as follows:

$N_x = \pm 20.0$ $N_y = \pm 20.0$ $N_z = \pm 20.0$. These are generalized design limit load factors. These loads take into account the worst-case launch load environment. These loads also take into consideration steady state, low frequency, transient loads and high frequency vibration loads. The load factors are in g's and are applied on each spacecraft axis independently. Accelerations are to be applied through the center of mass of the analyzed component using the NS-4 coordinate system. Thermally induced loading, including on-orbit thermal loading, shall be combined with the above loads.

It is necessary that the craft be deployed in the orbit previously determined in order for all of the ADCS, Attitude Dynamics and Control Systems, components to work properly and without any error. All tasks must be performed in order to prevent any attitude control mistakes. The requirement of providing efficient, reliable power better than what exists while also testing and demonstrating solar cells in space is considered necessary. This mission also includes providing essential power to subsystems of the satellite with redundancy while communicating via radio waves establishing a complete link from the satellite to the ground station.

3.2 Structural Requirements

The initial design for the payload called for a primary, secondary, and third mirrors to still be constructed from nickel plated 6061-T6 aluminum. Where the primary, parabolic mirror's mass is approximately 0.2268 kg, the second, hyperbolic mirror's mass is 0.6355kg, and the third, flat mirror's mass is roughly 0.0489 kg. Further dimensions can be seen below. The imaging system was to be assembled in a Nasmyth Telescope approach, which can be seen below in Figure 2 or in Appendix A in Figure 30, instead of the previous Newtonian design. However due to budget limitations, the Nasmyth, or modified Cassigrain, was replaced with a Maksutov-Cassegrain purchased from Celestron. The new design will allow the primary mirror to focus the light from Earth to a spherical, secondary, mirror positioned at the earth facing end of the telescope. Then the spherical mirror will then re-focus the light to a 45 degree angled flat mirror, which will be positioned in the back of the optical tube, behind the primary, and will project the image to the CCD camera positioned on the back side of the optical tube. By implementing this design, the effective focal length of the imaging system will be tripled, to 52.7 inches, enabling a 10-meter-per-pixel image resolution of the ground, possibly even enhanced. Figure 2, below shows the side view of the old Nasmyth telescope design. While Figure 4 shows the new mirrors that were removed from the Celestron telescope.

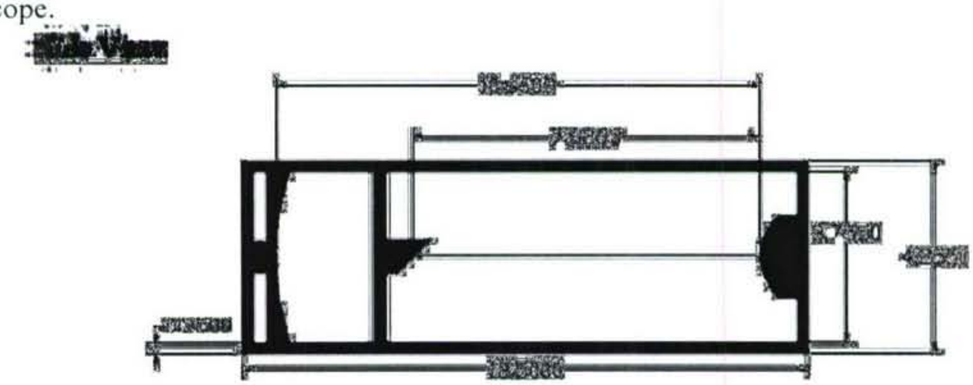


Figure 2- Side View of Nasmyth Design

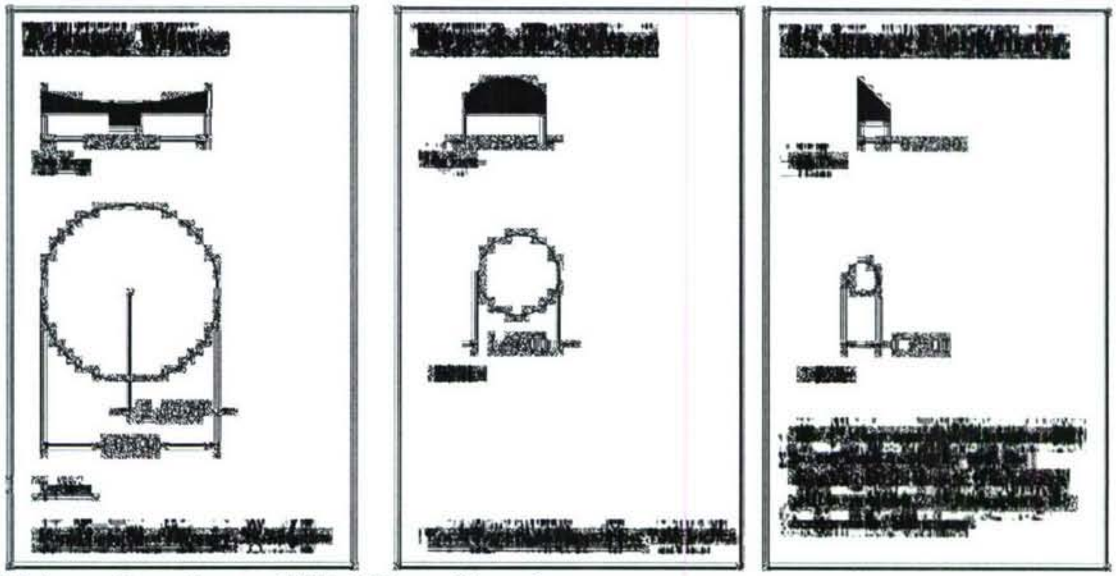


Figure 3-Primary, Secondary, and Third Mirror Dimensions

In order to do the proper testing, the computer, shown below, was removed. A diagram of how the new Celestron Cassegrain system is shown below in Figure 5. Some of the basic specifications are as follows:

- ☐ New Primary Focal Length: 1325 mm (52.17inches)
- ☐ Optical Tube: Aluminum
- ☐ Secondary Mirror Obstruction: 1.38 inches (approximately 11% of incoming photons)
- ☐ Current Telescope weight: 4.99 kg (will be reduced)

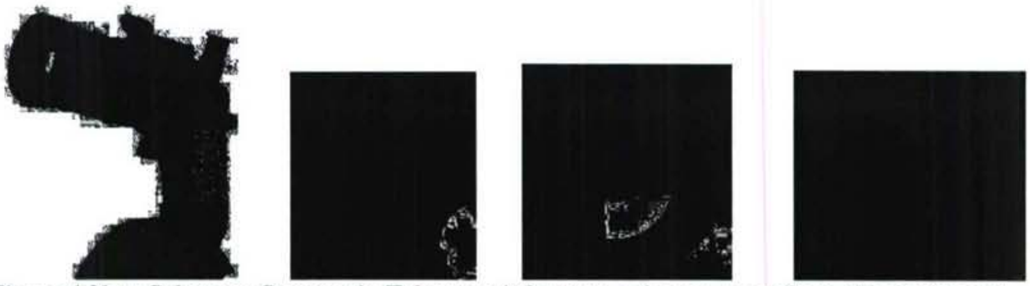


Figure 4-New Celestron Cassegrain Telescope (telescope, primary, secondary, third mirrors)

Figure 5: Maksutov Cassegrain Telescope

A three mega pixel, CCD camera with a resolution of 2048 X 1536 pixels manufactured by XL Imaging has been selected for use in the imaging system. It has dimensions of 45 X 83 X 17 mm with a weight of 56 grams. The camera is based on a plug-n-play USB 2.0 interface, and will communicate to the on-board computer through a Linux driver, obtained through XL Imaging. An issue has been brought to the team's attention concerning the USB connection system in reference to vibration reliability. The possibility of hardwiring the connections and doing away with the USB connectors will be pursued as stated above in previous sections.

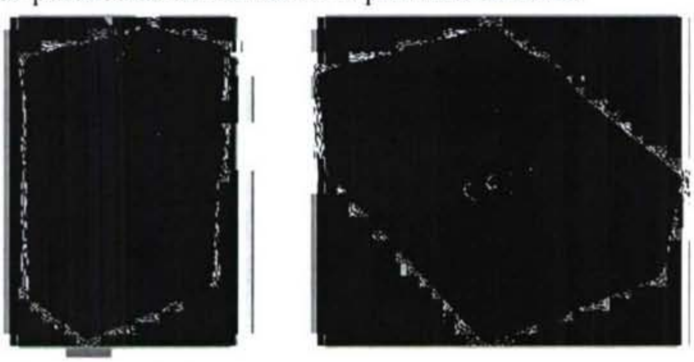


Figure 6: CCD Camera

The interfacing requirements for the payload have an opening for the reflected photons in enter, a connection mechanism, and sufficient space for the CCD camera and casing. An opening was created at the bottom of the satellite (the portion that will be facing the ground) for the ground image to enter the optical tube. The payload will be fastened to the intermediate iso-grid using stainless steel screws in conjunction with helicoils, which will be fastened to the main structure of the bus. The bus was designed with the amount of room needed for the payload in mind.

The Payload is required to interface with the onboard computer by means of a USB port that is currently within the design of the imaging CCD Camera. It will be necessary to connect the USB, with a substantial amount of security (i.e. bolting or perhaps solder), to the computer. This will provide the confidence that is needed for this system to withstand the required forces and not become unfastened from the computing system. This connection will be essential to relay the images to a designated ground station. The optical/imaging system will require 0.056kg for the CCD camera. It is known that the CCD will call for approximately 0.75 watts of power. In addition, the camera will require much less than a 1GHz processor. Previously, this was an issue since the Gumstix processor selected for Knightsat is only capable of 400 mHz, however, after further research, the Gumstix will be able to provide adequate processing. This is due to Linux processing speed requiring much less processing power.

Communication between ADCS and payload is necessary to assure proper nadir pointing toward earth giving an accurate and precise image. The payload will require that the attitude of the spacecraft be within 1 degree of accuracy of the desired area. The payload, including the optical tube and the CCD Camera, will consume a space within the bus of approximately 5.5 inches in diameter and 12.50 inches long. Essentially it will be a cylindrical shape. The entire mass budget of the payload is estimated to be approximately 5.0 kg. This is significant in order to interface with the structure of the bus.

The purpose of the shutter system is to be able to block debris during launch and ADCS maneuvering protecting the telescope lens located at the top of the bus. As indicated in figure 7, the shutter system consists of a 4.25 inches circular cover divided in two equal sections with a ¼ inch wide extension to attach the supporting wire. Each half is supported by its individual spring loaded hinge and is hold in the closed position by an aluminum wire. The shutter must be able to withstand a minimum of 20 G's that could be experienced during launch along with accompanying vibrations. The shutter is to be constructed with aluminum 7075 alloy of 1/16 inches in thickness. The total mass of each shutter is estimated to be 40.68 grams with the center of gravity located 2.29 cm from its straight edge. The load on each hinge at 20 G's acting parallel to the open shutter would be 7.98 N. When each shutter half is closed, the load applied on the hinge at 20 G's would be 3.386 N and the load on the wire, 4.595 N. If only one wire is used, the load on the wire due to two shutters would add up to 9.19 N. In order to maintain the shutter closed the wire must also counteract the force of the spring loaded hinges, which will require a force of approximately 2.5 N. Therefore, the total load to be supported by the wire would be 11.69 N.

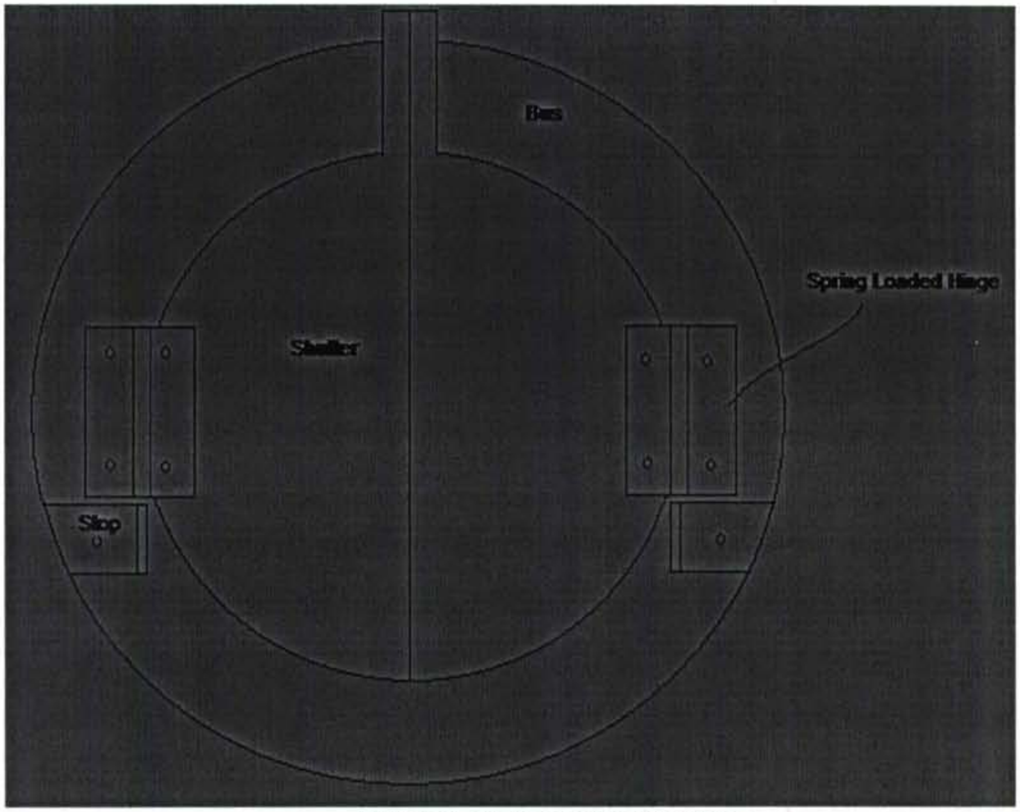


Figure 7: Outline of Shutter Design.

In order to allow the shutter to open, a current is passed thru the aluminum fuse type wire burning the wire and releasing the shutter system to be opened by the spring loaded hinges. The aluminum wire could be obtained from California Fine Wire Company. Due to the fact that data is not provided regarding the electrical power needed to burn the aluminum wire, this data would have to be obtained throughout experimental testing of different gauge wires to select the proper gauge to be used. The bus of the satellite is used as the electrical ground of the satellite and is also to be used as the electrical ground of the shutter deployment system. The positive charge applied to the aluminum wire is to be passed throughout the pin that mechanically holds the shutter closed. Refer to figure 8 for details. This pin should be mounted to the bus utilizing an electrical insulating material such as plastic or Micarta. When sufficient voltage is supplied to the pin, the current would flow thru the aluminum wire elevating its temperature and producing a rupture in the wire due to melting of the aluminum, releasing the shutter mechanism.

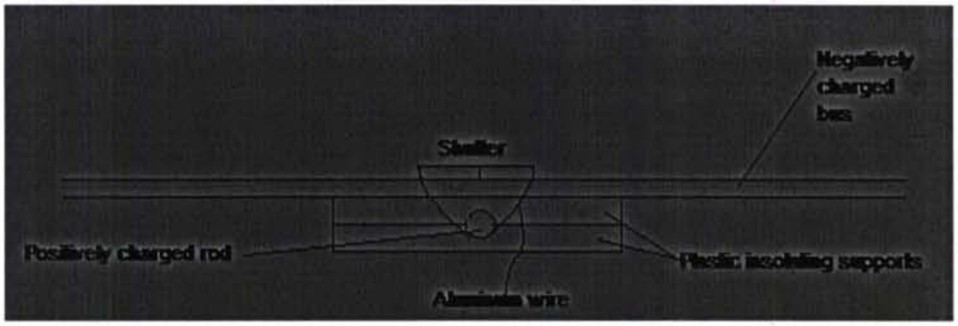


Figure 8: Locking Mechanism of Shutter

A prototype was built to ensure the design was successful. The prototype consisted of 1:1 scale shutters constructed with 1/16 inch thick 2024 aluminum alloy, material similar to the alloy to be used in actual construction. The stainless steel spring loaded hinges were obtained from McMaster under part number 15205A42. The fasteners used to attach each shutter to its corresponding hinge were two countersunk rivets 3/32 inch diameter commonly used in airplane construction under part number AN426-3-1. The hinges attach to a 0.050 inch thick 2024 aluminum alloy plate that simulates the bus at which the shutter is to be installed. The fasteners used to attach each hinge to this plate were two AN470-3-3 rivets. In order to provide some protection from solar UV radiation that could affect the telescope when the shutters are opened, the shutters are kept perpendicular to the bus by means of two mechanical stops. These stops were constructed using 0.050 inch thick 2024 aluminum alloy. The stops consist of a 90 degree aluminum angle attached to the bus by one AN470-3-3 rivet.

The structure must be able to with stand the worst case of a launch environment. The possible altitude of operation is between the ranges of 350km and 700km. The maximum temperature of these conditions is 900K. However, this temperature is due to radiation and not convection or convention. The structure will also need to have a FOS of 2.0 for yielding strength when 20g's is applied to each axis independently. During analysis the acceleration force is to be applied in conjunction with the temperature load that it will experience while in orbit. The maximum temperature encountered by the structures due to electronics and operations on board have not yet been determined. However, the highest temperatures that the structure can with stand during launch, and maintain a FOS above 2.0 is 325K, which resulted in a FOS of 2.2. From the analysis, the operating temperature was concluded to be below 325K. The altitude that corresponds to this temperature has not been determined. The reason for the inclusion of operating altitude is due to the ultraviolet radiation that causes these high temperatures. The temperature at 350km is 900K, however, because it is caused by radiation there is very little heat. This means that even though the temperature is high it is not hot. This situation makes it difficult to determine what the non radiation temperature would be. This temperature is necessary in order to conduct an accurate thermal analysis on the bus and other subsystems. It is favorable that the components composed of stainless steel and aluminum 6061-T6 is able to withstand the conditions between the altitude of 350km and 700km, however there are tests that need to be performed to prove this.

Since free convection is unavailable in space due to the lack of gravity, passive control elements must be used in order to thermally protect the satellite bus. Passive thermal control elements include the use of coatings, insulation blankets, sun shields, radiating fins, and heat pipes. Currently with the information that we do have, the satellite will be operating at an altitude between 350km and 700km. This puts the structure in the range of the thermosphere/exosphere.

In order to withstand the harsh environment of space, the structure and all of its aluminum components will be anodized. Anodizing aluminum converts the aluminum surface into an extremely hard, corrosion resistant, durable, and long-lasting aluminum oxide finish. Anodizing also responds favorably to current government regulations because it is an environmentally friendly process. Therefore, the anodized satellite bus will be chemically stable; will not decompose and will be heat resistant to the melting point of aluminum (1,221 degrees F).

The bus structure will need to interface with the following components: the light band, PPT's, Solar Cells, gravity gradient boom (GGB), and Payload. The light band interfaces best with a cylindrical surface; this is the primary reason for the cylindrical interface plane located on the bottom of the bus (SIP). The light band will be a Planetary Systems Corporation (PSC) motorized system, which will need to be assembled to the bottom of the bus. In order to assemble the light band to the bottom of the bus, there are 24 drilled holes where the fasteners will be located. The holes will be of diameter 0.281inches and the diameter of the circle that the holes will be drilled on is 15 inches. The fastener holes will need to have backout prevention such as locking helicoils.

The solar cells were a dominate factor in the determination of the geometry of the bus. The solar cells that were chosen were not flexible solar cells, thereby needing a flat surface to be mounted on. To compensate for this feature the bus was redesigned to have a icosagon exterior in place of the original cylindrical exterior. The design change resulted in a total of 20 flat surfaces, by which can hold one 12 volt cell each. This created room for a total of 54 solar cells.

The function of this design is to house and protect all components of the satellite. The conceptual design consists of an isocagon shaped structure, manufactured out of Aluminum 6061-T6. The bus is be machined as three separate components and then mounted together using stainless steel fasteners. The minimum wall thickness of the bus is 1/16 of an inch and is designed with stand 20g's on all three axis independently. The structural integrity of the bus must survive an acceleration of 20g's as well as a vibration acoustic test. The bus will also be required to have a natural frequency of 100Hz. The weight and volume of this design is 41.77 lbs and 428.26 cubic inches, respectively. It is important that the center of gravity (CG) is located in the center in order to minimize stress and tumbling of the satellite when launched.

To reinforce the CG location the payload is located in the center of the structure. On each end of the structure, as well as in between each component, there are isogrids that are used for mounting purposes as well as stability reinforcement. The isogrids were designed to be lightweight, mounting surfaces for the multi-components located within the bus. . There will be battery cases, an IMU, a computer, torque rods, and a gravity gradient boom attached to the structure, which will be discussed in detail with in the

ADCS portion of the report. Figure 9, below, shows the interior and integration of parts within the bus.

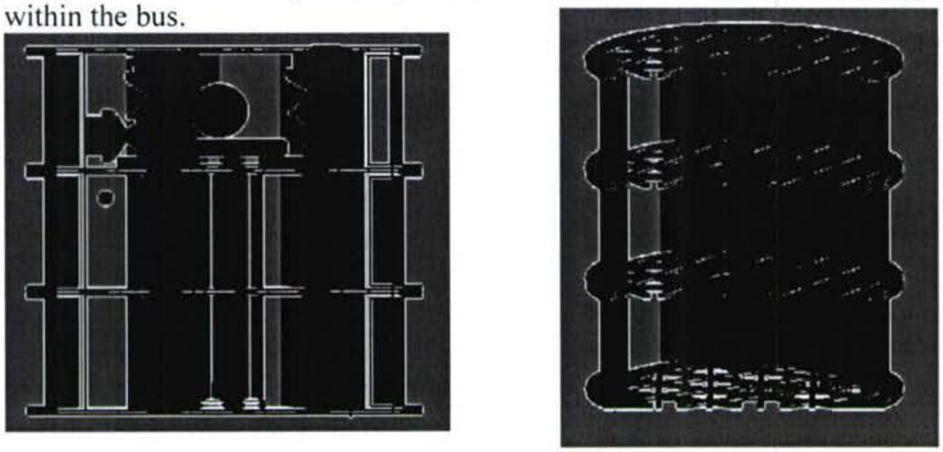


Figure 9: Internal Image of Satellite Bus / Integration

Pressure vessels and pressurized components defined in NASA-STD-5003 are prohibited. Fluid and gas containers or structural compartments that cannot be vented shall meet the definition of a sealed container as specified in NASA-STD-5003. Any sealed container shall be constructed of metal. A sealed container is defined as any single; independent (not part of a pressurized system) container, component, or housing that is sealed to maintain an internal non-hazardous environment and that has a stored energy of less than 14,240 ft-lbs and an internal pressure of less than 100 psia. The Nanosat shall be designed to withstand the launch vehicle shock environment without failure.

3.2.2 Attitude Dynamic Control System

3.2.2.1 Pulse Plasma Thrusters:

There will be six Pulse Plasma Thrusters, PPTs, used; this allots two per axis for redundancy. An additional two could be fitted to the top of the bus to allow for a de-orbiting burn. The placement of PPT's will be based on the moment of inertia of the satellite. The bus needs to interface with PPT's by giving the location of the moment of inertia, as well as supply adequate spacing for the mounting locations. These thrusters need to have the ability to fire in coupled pairs as needed to correctly control the attitude of the satellite. A power requirement of 1000 volts will be necessary to spark the igniter beginning the reaction. No external fuel tanks are required as the fuel is housed within the thruster itself.

The computer control algorithms are also vitally important. Without proper interface between these two systems the PPTs will not have a successful mission. The only other system required interfacing with the PPTs for operation, assuming all ADCS knowledge and orientation is taken care of by the control algorithms for the PPT, is the power systems to ignite the spark and maintain the charged anode and cathode. The PPT systems will require two separate circuits to operate properly; one circuit will utilize large capacitors to deliver the 1000 volt charge to the spark igniter for each firing. The second circuit will maintain a 1-2 watt power source to the cathode and anode, again all these values are per thruster. A pair of thrusters will be positioned on all three axial faces of the satellite. One pair of PPT's for the X axis, one pair for they Y axis, and one pair for the Z axis. The current design will require a support bracket that will need to be developed.

For the PPT subsystem to function properly there must be adequate power provided. Most critically is the capacitor, provided by SB Electronics, which has a discharge of 1000 volts to the spark igniters. Without this discharge ablation cannot be initiated and the required plasma state will not be attained. In this scenario the PPT have no likelihood of success. Secondly if the ablation is successful the charge on the cathode and anode need to be established and maintained. This charge will accelerate the plasma fuel through the throat and out the nozzle of the thruster. Though some thermodynamic thrust will be developed without the cathode anode pair, thrust levels would be greatly reduced. Finally it is critical for the PPT to be properly restrained within the bus.

A rough working prototype of the Pulse Plasma Thrusters already exists, however a slightly altered design with a negatively charged nozzle is favored for additional benefits. A preliminary design can be seen in Figure 10 including the nozzle as mentioned. The nozzle design allows for added thrust due to thermodynamic forces that are in addition to the electromagnetic propulsion already present. Through some initial testing it was also found that the inclusion of a permanent magnet placed behind the fuel allows for more consistent burning of the Delrin. The addition of the magnetic field extending through the fuel bar to the combustion creates a spiraling or helical effect on the exhaust about the axis of the anode. This effect assists in keeping the plume stable and is helpful in the prevention of plume contact with other components allowing for repeated firings with less system degradation. Delrin is aTeflon like plastic that has desirable characteristics for this mission. This material is selected for the fuel for several chemical properties inherent to it. The main benefit over Teflon is the materials ability to burn completely and cleanly allowing for no debris to collect on the camera lens. This is due to a vortex that is created within the nozzle.

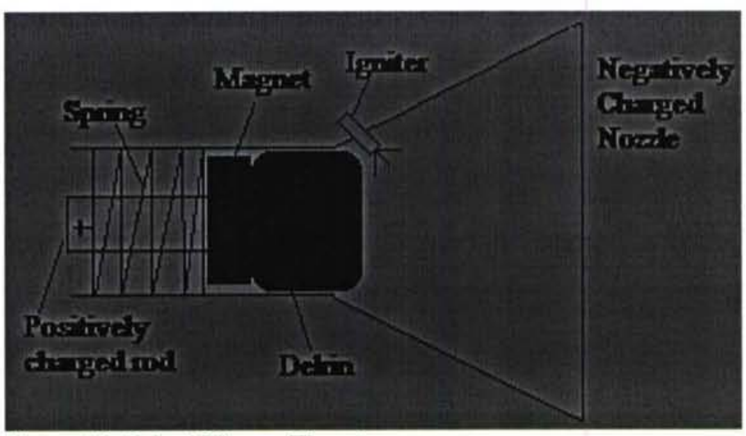


Figure 10: Pulsed Plasma Thruster

A more complete design for the PPT has been completed using the design ideas mentioned above. The design will be cylindrical in shape with a diameter of 47.0 mm and a total length of 80mm. A cross sectional view of this proposed design can be seen in figure 11. The design will be composed of two half cylinder pieces that will be bolted together with a flange. This same flange is to be used to bolt the PPT's to the structure of the bus using one bolt on either side holding the entire system in place. This can be seen below in figure 11.

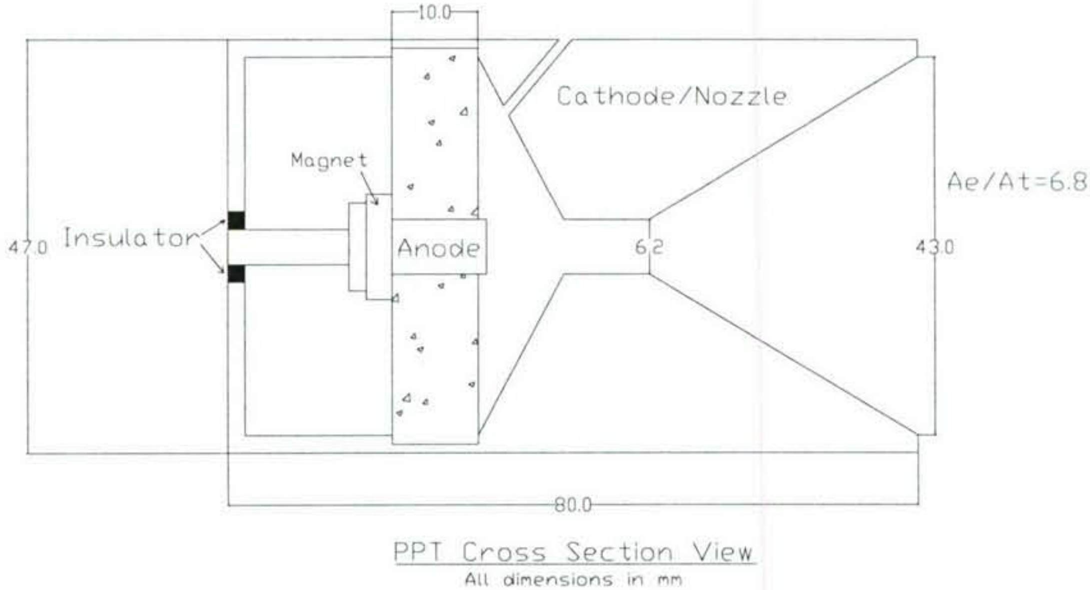


Figure 11: Cross section Cylindrical PPT design

Torque Rods:

Three separate torque rods will be needed since the space craft is able to rotate on all three axis. This is a major requirement because each torque rod has to be able to act separate from the others in order to adjust for each axis position independently.

The power subsystem is a vital link to the functions of the torque rods. There should always be the available power needed to activate and run the torque rods when needed. The structures subsystem will need to allot the amount of space required to house the torque rods. Furthermore the team needs to ensure that they are not positioned too close to any equipment that could be influenced by an induced magnetic field. The torque rods will need to be aware of how much to correction for the position of the space craft is necessary. The last interfacing of the torque rods is with the other ADCS systems. By precisely distributing the attitude control to each attitude control device will ensure that the torque rods are not over stressed as well as keeping them to a reasonable size.

The dimensions and materials for the torque rods have been changed. Each rod will be 12 inches long by 0.5 inches in diameter. There are two bobbins, one at each end to hold the wire on the rod, with the bobbins in place the total winding length of the rod is 11.55 inches. There will be 1408 turns of wire. The core material for the torque rods will be constructed out of EFI 50 nickel alloy. This metal will be annealed in a hydrogen atmosphere at 1200 centigrade for four hours. The bobbins will be made from delrin. Both will have a diameter of 0.37 inches. Bobbin one will be 0.3 inches long and bobbin two will be 0.15 inches long. The wire that will be used is a 26 gauge copper magnet wire. It has a diameter of 0.0159 inches and with 1400 turns the total resistance will be 1.316 ohms. The winding factor on the rod will be 0.9 and a thin layer of epoxy will cover the wire and protect the rod with adding any noticeable weight. Figure 12 shows a torque rod that was built to the same specifications under the same means as the ones that will be used. The only difference is that it is not as long as the ones for this project.

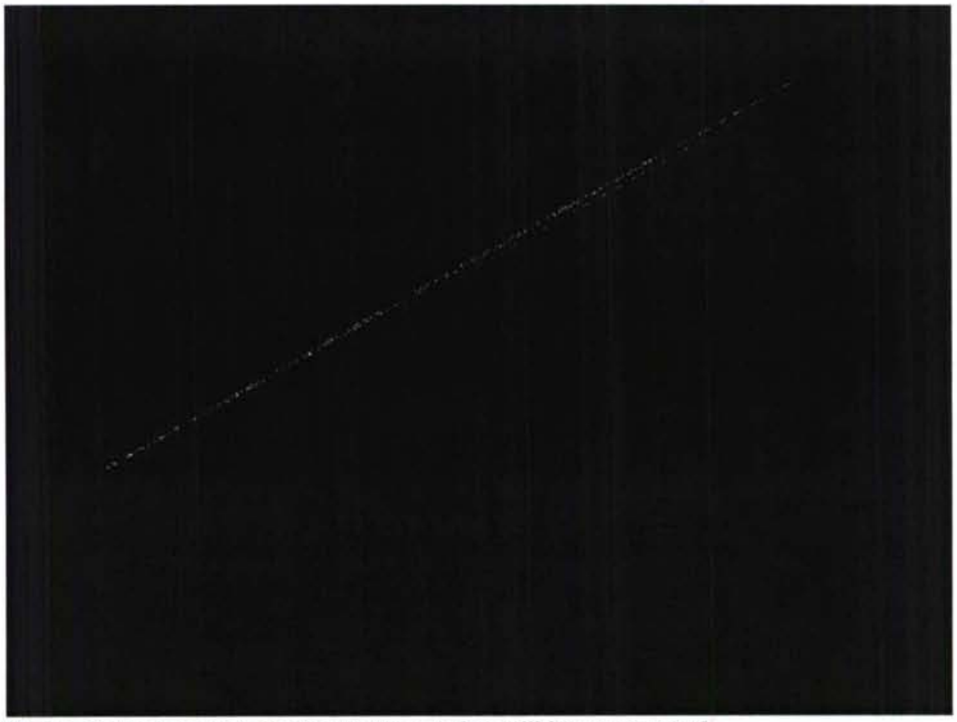


Figure 12: Torque rod similar to the one that will be constructed

Notice in figure 12 the bobbins on each end of the rod. Bobbin one is larger than bobbin two and feeds both ends of the wire through it. Also, this figure shows a metal tube, this tube was used to encase the torque rod, something that will be avoided using the epoxy coating.

3.2.2.3 IMU

The fabrication of a container for the inertial measurement unit, IMU, will have to be created to insure proper storage and safety for the duration of the mission. Without this the component could be damaged or could possibly damage other systems of the craft. The addition of proper magnetic and electrical shielding for the components will also assure proper operation throughout the mission. Proper testing and analysis is also needed to ensure all components are functional in a space environment.

There is primarily one main component of the ADTS system which is necessary to carry out the KnightSat mission appropriately. This component will transmit information to the flight computer where the data can then be interpolated into commands that are sent to different parts of the ADCS to keep the craft oriented accurately. The system is the MicroStrain 3DM-GX1, Figure 13, which is a gyroscope, accelerometer, and magnetometer all together in one system. It has the capability of performing all of these actions to the necessary parameters the satellite requires.

The magnetometer has a sensor range of +/- 1.2 Gauss. In a low earth orbit altitude the most that the magnetometer will be seeing will be between 0 – 1 Gauss. This will provide adequate and accurate data It operates by measuring the earth's magnetic field on all three axis's using the Euler equations which can then be converted into quaternion equation and orientation matrices, Equation 4.0.9, to be transmitted into data which the computer can then send down to the ground station telling its attitude. The Quaternion equation is more beneficial in the field of non-spinning craft and provides accurate results even if there is the presence of magnetic interference or linear acceleration. The Euler equation provides more useful information regarding spinning craft.

The accelerometer has a sensor range of +/- 5 G's. This portion of the control can help verify the crafts velocity in orbit through using a modified version of the equation V= a * t. Only through further simulation will this actually be able to be verified if the craft will produce enough G's to make this possible. Figure 13 shows the circuit board of the Gyroscope.

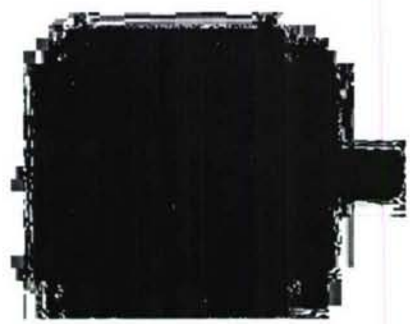


Figure 13: Circuit Board of Gyroscope

Throughout this process determining the exact components needed to make this mission a success has progressed. Finalizing each component has been the main focus point. Each option has been weighed and research has been done to guarantee that these parts are what are necessary to carry out this mission to the best of its ability. The magnetometer previously thought to be most adequate to the systems needs is no longer being used. Through performing further research the team has discovered that the IMU will provide all of the necessary data needed and perform all the functions a separate magnetometer would do. It is also available in one package which cuts down on weight and cost.

3.2.2.4 Gravity Gradient Boom

The Gravity Gradient Boom (GGB) is a mechanism that requires deployment, which needs an opening of some sort in order to perform its intended function. To accommodate this mechanism an opening was created at the top of the bus.

In order for the gravity gradient boom to achieve mission success, it must fulfill its own set of requirements while depending on other subsystems successes. Specifically, the gravity gradient boom must deploy and stabilize the satellite with the camera facing earth. Also, the gravity gradient boom must maintain stabilization to within one degree of vertical.

The boom will require interfacing with several subsystems in order to obtain success. First, the gravity gradient boom must interface with the satellite's bus. It must be rigidly secured to the satellite bus for the extended boom to be effective. Also, the boom system must interface with the computer and communications system which will tell it when to fire the solenoids and deploy the boom after the craft is positioned properly. Finally, the boom must interface with the satellites electrical subsystem to draw the necessary power needed to activate the deployment mechanism.

Shown below, in figure 14, are the manufacturing drawings of the gravity gradient boom system. These drawings will be used for the manufacturing of the boom model. The first drawing shows the exploded view so the internal parts of the system can be seen. The second drawing shows the assembled view and how the pieces fit together in the casing.

Figure 14: Exploded view of the Gravity Gradient Boom Spindle Assembly

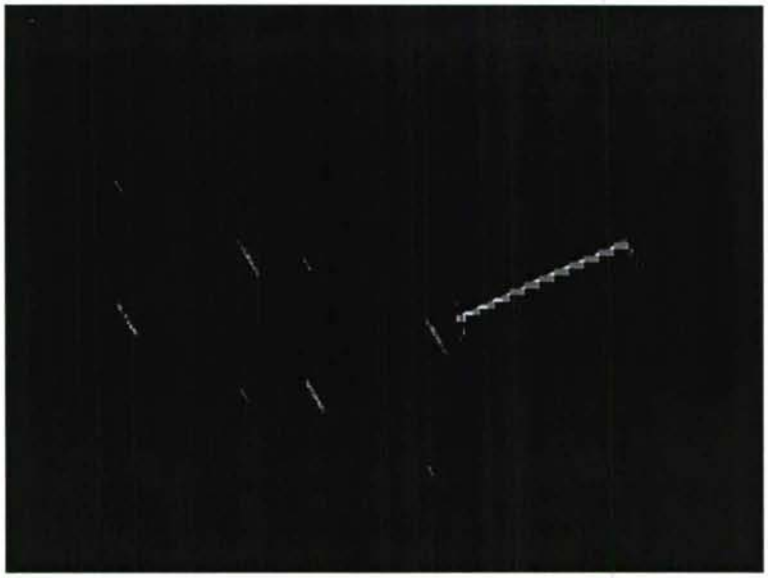


Figure 15: Completed view of the Gravity Gradient Boom Assembly

Shown below, in figure 16, is a schematic of the forces acting on the satellite in orbit.

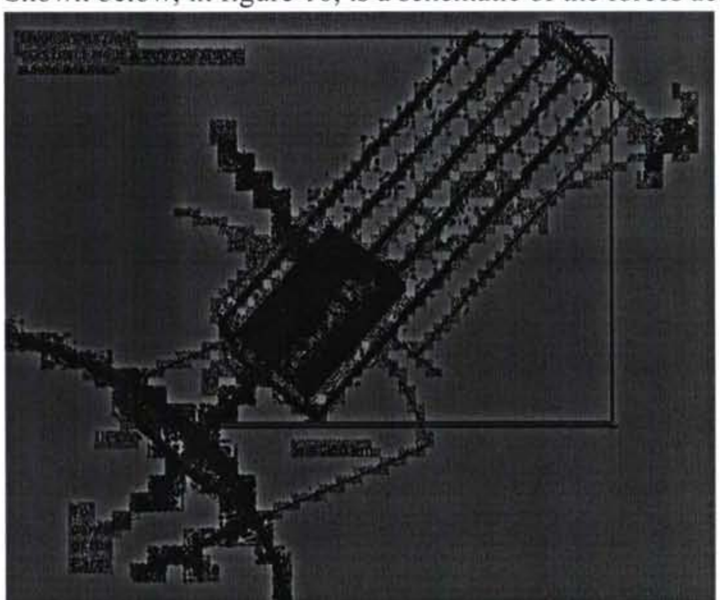


Figure 16: Gravity Gradient Boom

The deployment procedure for this boom is shown below in figure 17. This image displays precisely how the boom will appear before and after the solenoid has been activated. This delicate procedure must operate correctly or the mission could face possible set backs that could jeopardize the success of the mission.

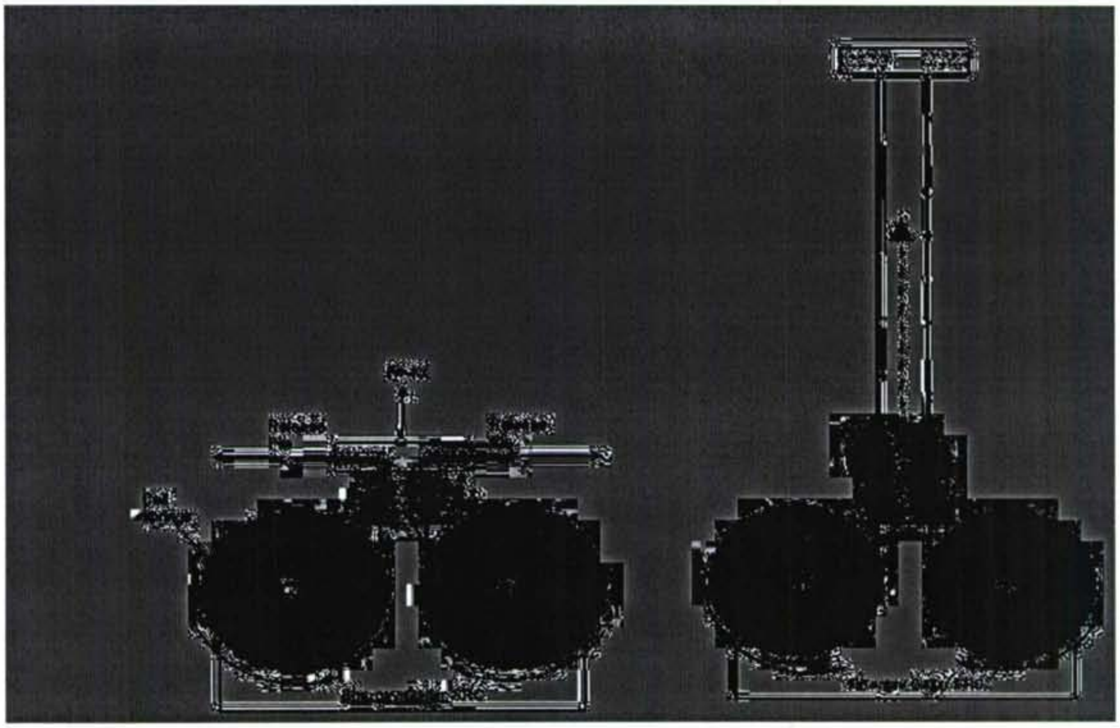


Figure 17: Boom Deployment before and after the Solenoid is fired

According to the current design, the boom and tip mass will weigh under 35 g. The material chosen for the boom is a metal tape and the housing will likely be made of aluminum. At this time the power consumption is a one time charge of 6V to each of two solenoids. Lastly the volume of the system is approximately 48 cubic inches (6 x 4 x 2). The gravity gradient boom will be placed at the top of the satellite as shown in figure 18.

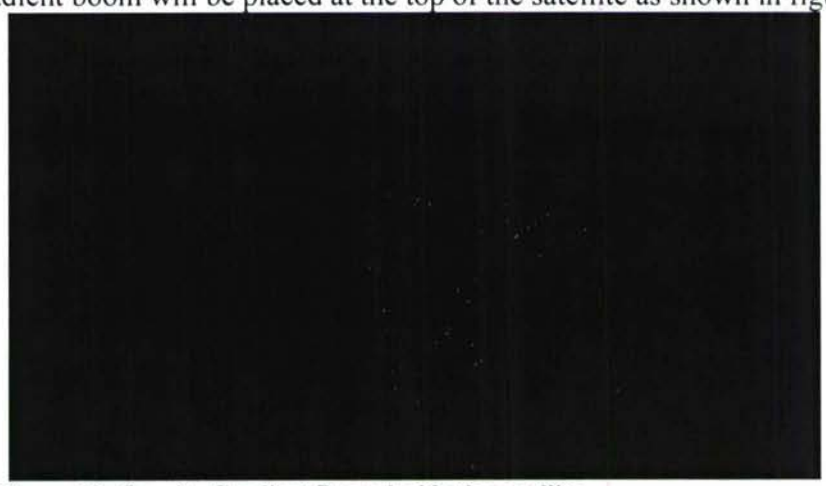


Figure 18: Gravity Gradient Boom inside the satellite

3.2.2.5 Computers

The on board computer is a critical component of not only the ADCS subsystem but all of the subsystems. The computer will be interfaced with sensors as required from each subsystem. The computer will be responsible for reading in and processing data from the sensors. This information will then be used to control the stability of the satellite. All signal processing will be handled by the embedded system without any human interaction. The interaction of each component along with the interaction with the computer is essential in order for the mission to be a success. Development of proper computer algorithms to control each of the systems in the Attitude Determining and Tracking System, ADTS, will be of the highest priority. If the computer does not get an accurate reading from the tracking systems the mission will certainly be a loss.

The hardware and software have been chosen to meet certain requirements. The Gumstix brand of miniature computers has been chosen because of their small size, low power consumption, high performance and the wide range of input/output daughtercards available for expansion. A complete embedded computer system has already been purchased which includes the following Gumstix components: a motherboard, serial expansion board, break out board, and flash memory. All Gumstix hardware comes preloaded with the Linux operating system.

The selection of these components was constrained by certain parameters such as weight, power consumption, data handling rate and storage, operating conditions, and cost. The weight of the hardware is less than .5kg. The power consumption of the embedded system will not exceed 1 watt and must maintain a voltage between 3.5V to 6V. The power consumption ranges from 0.04A when the system is idle to 0.42A when the system is at full computing capacity. In order to properly process and store the sensor data the computer needs to send and receive data at a rate faster than the sensors. The processing speed of the motherboard is 200 MHz, which will be sufficient to accomplish proper data handling. The computer includes 64MB of RAM and 512 MB flash memory. All hardware components will be thermally insulated to ensure that an operating temperature between 0 and 65 degrees Celsius is maintained. The embedded system will also be shielded because any magnetic or electric interference can cause errors in data processing.

The computer must be fully compatible with all sensors and peripheral hardware for accurate signal processing. This will be achieved through the simultaneous use of the different Gumstix boards. The serial expansion board will receive the data from the sensors, uniformly format the data, and then send the data to the motherboard for processing. Next, an analog to digital converter capable of using Pulse Width Modulation (PWM) for conversions is employed. The Gumstix break out board will be used for the PWM conversions. PWM outputs will be calculated by the computer from the input data and then used to control the amount of power sent to the actuators in order to maintain control and stability. The Gumstix will use RS-232 protocol for communication between the computer and peripheral devices, which is compatible with the ADCS sensors. Precise programming of the software is also required in order for the input data to be properly processed and the correct output to be obtained. The accurate translation of input data to output data will be a controlling factor in maintaining the stability of the satellite. Compatibility between hardware and software must be rigorously tested and ensured.

ADCS also has basic stability and control software programs, which will reduce our in house programming needs. The software is completely scalable and only needs to be changed to conform to our particular performance parameters. The programs will be written in C code, which is fully supported by the Gumstix libraries. The programs will be built on a host computer and then transferred to the Gumstix through a cross-compilation tool chain, which is compatible with the Linux operating system. The data is easily transferred through the ethernet or serial ports included on the Gumstix.

Critical to the success of the computer is interfacing and testing. Signal processing can only be precise if all hardware and peripheral devices are able to communicate successfully with the computer. Thorough and extensive testing is also critical to ensure that the correct outputs are derived from specific inputs. The programming will have to be rigorously tested for any and all possible inputs to guarantee that the correct output is obtained. This will be accomplished by placing LED's at the outputs so that the output can be visually verified and confirmed.

Throughout this process determining the exact components needed to make this mission a success has progressed. Finalizing each component has been the main focus point for ADCS. Each option has been weighed and research has been done to guarantee that these parts are what are necessary to carry out this mission to the best of its ability. The magnetometer previously thought to be most adequate to the systems needs is no longer being used. Through performing further research the team has discovered that the IMU will provide all of the necessary data needed and perform all the functions a separate magnetometer would do. It is also available in one package which cuts down on weight and cost.

They can provide the team with the necessary data over the first few orbits from there uploading the computer with that data should provide ADCS enough information to keep the craft stable.
The computer hardware which will be used has changed from the preliminary design. The original hardware consisted of a control area network embedded microcontroller while the current design takes advantage of embedded computer products from Gumstix. The Gumstix products are ideal because they provide full computer functionality on a miniature scale and have the ability to easily add functions through optional expansion boards. The Gumstix embedded systems also meet the requirements needed as stipulated in the preliminary design.

3.2.3 Tracking and Ground Station

A change from the last design phase was the GPS unit. The Surrey Satellite Technology Ltd. SGR-05 receiver was designed specifically for miniature satellites in low earth orbit and has been successfully used on several satellites currently in orbit. Unfortunately this model has proven to be too costly for the team to utilize it, this lead the team to rely on NORAD for most of the tracking. They can provide the team with the necessary data over the first few orbits from there uploading the computer with that data should provide ADCS enough information to keep the craft stable. If there are any problems then further uploading can be completed to fix the errors.

The University pf Central Florida's ground station has been fully constructed and tracking software, PCSat-32 has been implemented and testing has begun on tracking smaller satellites. Images of the ground station can be seen below in Appendix A in Figure 47.

3.3 Electrical / Communications Requirements

Designs must address standard/electrical power system safety hazards including shorting which may lead to fire or ignition sources, routing of wiring, battery hazards such as overcharging, shorting, cell reversal, thermally-induced failures, and inadvertent activation of hazardous subsystems. The NiCd, Nickel Cadmium Batteries, cells will be supplied by the Air Force.

The Satellite shall be designed for electromagnetic compatibility and for mitigation of electromagnetic interference, specifically susceptibility to launch vehicle and range radiation environments. Simple EMI, Electro Magnetic Interface, mitigation techniques will be communicated to the university but the university is not required to generate a detailed EMI analysis, beyond tabular analysis required.

The integrated satellite system shall provide a continuous, electrically conductive path between each major structural component and the launch vehicle interface. All electrical ground support equipment shall meet all safety requirements. All components and/or interfaces that remain attached to the satellite for flight must meet all requirements for flight hardware.

Within the KNIGHTSAT Satellite program at the university, there are a few interfacing requirements that need to be met. Most specifically concerning the KNIGHTSAT program is the gravity gradient boom for the attitude and dynamic control system to be utilized in orbit. The gravity gradient boom needs a one-time supply of power in order to position the satellite into proper position for payload imaging. Attitude Dynamic and Control System also will need power for propulsion in arranging the bus of the satellite in proper position to receive accurate images from the payload. The second interface concern involves the payload imaging system, also known as the high resolution camera. The power and communication group assumes the tasks of supplying power to the high resolution camera as well as communicating image results back to the numerous ground stations on Earth. Another consideration for the power system design is the beginning-of-life and end-of-life requirements. Thermal outputs can be an issue when designing the power system due to the amount of thermal energy radiated by both the power subsystem and other spacecraft subsystems. Power subsystem design should encompass a rough estimate of the power needs of the payload/structure and attitude determination and control subsystems.

The power subsystem of the KNIGHTSAT satellite provides the following resources:

- Power generation by 54 solar arrays (Crystalline solar cells)
- Solar array power and battery energy control / regulation
- Power protection and distribution
- Energy storage using Sanyo KDR-4000 Ni Cad Batteries

It is designed to provide an average of 10W of continuous charging of electrical power for the duration of the mission that requires observatory instruments, pulse plasma thrusters and subsystems. The solar panels and the secondary batteries are designed to provide sufficient power for a period of 5 years.

For the KNIGHTSAT satellite, the power source will be crystalline solar cells that transfer energy in the form of radiation from the sun into usable electric energy. Crystalline sources are most common in today's space satellites because of the abundance of energy given off by the Sun and their ability to provide a cost efficient method of converting the Sun's usable energy into power. The usable electric energy is transmitted from the solar cells to the Nickel Cadmium battery packs and stored for necessary use. In this situation, the batteries are discharged throughout the system to ensure ample power and operation at all times for the mission. There are two types of batteries, primary and secondary; for this mission, secondary batteries will be employed for their ability to discharge and be charged again for later use. The batteries will be discharged through a power distribution system which will be composed of cabling, power converters, and a power bus. The Power will be supplied to requested components via power cables and DC converters. Power regulation must also maintain the required power requirements distributed to each subsystem of the satellite and the discharge and charge of the batteries bus voltage. The battery charging systems are also part of the power regulation system.

The communication system requires the use of two radios to transmit the images and information between the satellite and the ground station. Between the radios and the ground station as a dual band Omni directional antenna capable of sending high resolution data in frequencies that will be captured at the ground station. The antenna can also send data containing correspondence of functions that may be required of the satellite. Compared to other Universities Satellite Programs, the Power and Communications of the KNIGHTSAT satellite compare consistently. The KNIGHTSAT satellite subsystem of Power and Communications is composed of solar energy converted into power that transmits to necessary subcomponents primarily through direct transfer of solar energy and secondly through 20 Sanyo KDR-4000 batteries and DC converters. The communication system uses a modem, antenna, amateur radio waves, and radios to communicate between the satellite and Earth. All of the components are standard in small satellites.

The design of the power and communications subsystem consist of 54 solar array panels surrounding the body of the satellite converting solar rays into stored power energy while in the suns ray direction. The solar cells are attached to the aluminum bus via #6 3/8th allen key screw with locking bolts cemented onto the extruding screw. The #6 stainless steels screws posses lightweight physical properties and tensile strength of 600 psi or greater. The locking screws will be specifically used to connect the solar panels to the satellite. Figure 19 shows the side-view of the schematic.

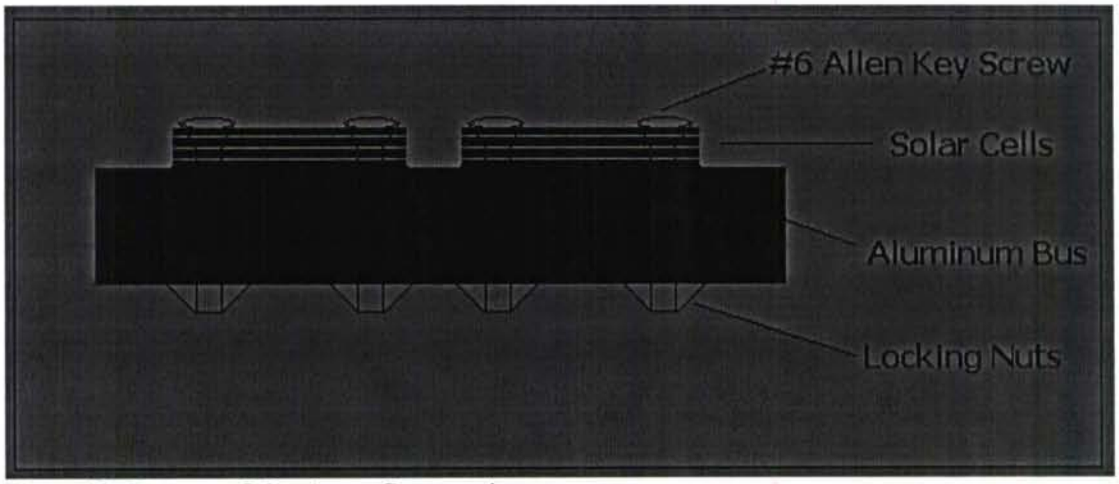


Figure 19: Prototype Solar Array Construction

Figure 20 shows the Kyocera prototype panels have 18 solar cells. There are metallic interconnects and a thin sheet of glass bonded to the top of each panel and are therefore referred to as CICs (Cell, Interconnects, Cover glass). The CICs also come with a diode, a protection device that among other things keeps the cell from permanently "frying" under certain conditions. The interconnects on each cell are of the same polarity and provide redundancy for the connections. They are wired in series to the topside of the Kyocera cell on one of the long edges of the cell. The entire topside of the cell is of negative polarity while the entire bottom side is of positive polarity.

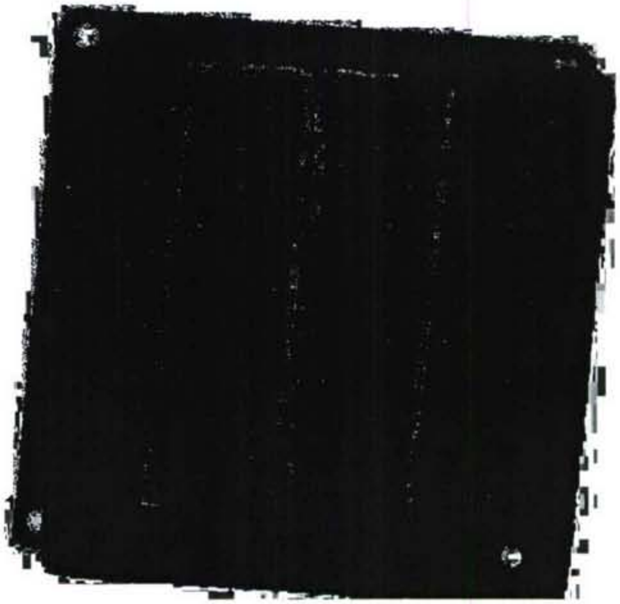


Figure 20: Display of Kyocera Crystalline solar cell

The interconnects are thin strips of metal less than .002" thick made to relieve stress and are used to connect the solar cells in series. Stress relief is important to reduce the possibility of the interconnections separating off of the cell after thermal cycling. Silver plated Kovar is usually used because the CTE (Coefficient of thermal expansion) is fairly well matched to the cell's germanium substrate. Copper is not a good choice since it expands and contracts significantly more than the Germanium. However, solder plated Cu metal strips can be used at the ends of the series strings for termination wires or for connecting strings in parallel. The solar array is composed of solar cells connected in series as strings, which are wired to adjacent strings in parallel on each face of the satellite. Figure 21 displays the satellite solar array.

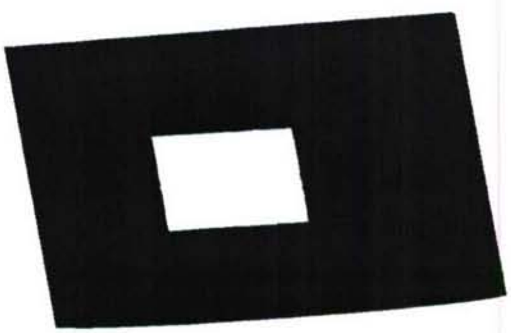


Figure 21: Cad wiring of solar panels

In order to increase the voltage from the cells, two panels are connected electrically in series. This is shown in figure 21 where panels of the same color represent two panels connected in series. From the series connections, 27 panels will be connected electrically in parallel. The simplest way to connect the cells is by using wire between positive and negative leads stemming from the solar panels. This can be done using 26 gage wire which can be soldered from one cell's leads to the next.

After the strings are completed, the current on each side can be combined in parallel. There are two holes on each panel of the satellite to run the positive and negative leads from each solar panel. These leads will connect directly to the power board. Wires are routed from the last cell in each string to a terminator strip (these wires all come from either the front or the rear side of the cells).

The solar cells converted power is stored in the Nickel Cadium battery packs and regulated via a PT-5100 DC converter. Figures 31, 32, 33, and 34 display the circuit design for the power subsystem. The power pins are labeled in the following table:

For the connector

Pin 14	=	Positive input from the solar cells.
Pin 1 – 4	=	Output 12 volts VNQ660SP
Pin 5 – 8	=	Output 9 volts VNQ660SP
Pin 10 – 13	=	Output 5 volts Regulator
Pin 25 – 26	=	Output 3.3 volts Regulator
Pin 21 – 22	=	Output 3.3 volts Regulator
Pin 15 – 19	=	Ground
Pin 23 – 24	=	Ground
Pin 9	=	Open
Pin 20	=	Positive from battery

These components help to create, regulate, and implement the power circuit of the satellite.

For the solar array of the satellite, there are 2 solar panels connected in series with an additional 27 solar cells also connected in parallel. The bus of the satellite has twenty sides divided into three sections. Each section is defined as its own panel. For the power system, there will be two sets of solar panels connected in series. The sets connected in series will then be routed in parallel to each other. Figure 32 shows aforementioned solar array.

This material has a CTE much closer to that of glass/silicon. The silver clad Kovar, however, does not have a solder coating; so it is necessary to add solder to create a joint. The solar cells converted power is stored in the Nickel Cadium battery packs and regulated via a PT-5100 DC converter. Figures 31, 32, 33, and 34 (refer to Appendix A) display the circuit design for the power subsystem. The power pins are labeled as the following table:

For the solar array of the satellite, there are 2 solar panels connected in series with an additional 26 solar cells also connected in parallel. This combines to contribute a 24 solar cell panel connected in series. The bus of the satellite has twenty sides divided into three sections. Each section is defined as its own panel. For the power system, there will be two sets of solar panels connected in series. The sets connected in series will then be routed in parallel to each other. Figure 32 (refer to Appendix A) shows aforementioned solar array.

The DC Controllers (PT5110/PT5101/PT5102/PT5013) decreases the voltage from the main bus of 20V to 9/5/12/3.3V respectively. The VQN660SP line FET switch restricts voltage and current passing through power lines in the system. The FET switch will open a line if the voltage or current increases past the suggested value. The switch also can connect to the command and data handling to ensure that necessary devices receive required power. If the FET fails, it will leave the switch in the closed loop position thereby continuing to deliver power to the components of the satellite.

The inhibit design for the KNIGHTSAT mission allows for the satellite to be flown with completely discharged batteries. Once launched from the payload position, the normally open, inhibited circuits become closed, and the satellite can begin to charge its batteries from power collected by the solar cells. This design is two-fault tolerant in that if any two inhibits fail, the satellite will still be prevented from exchanging current between the solar panels, the batteries, and the loads. The redundant electrical DPDT switches will be utilized for the satellite's inhibit design. These switches will be placed in parallel with each of the mechanical inhibits, and once the satellites charge their batteries and becomes operational, the onboard software will automatically close these switches. Figure 22 describes the electrical inhibiting design.

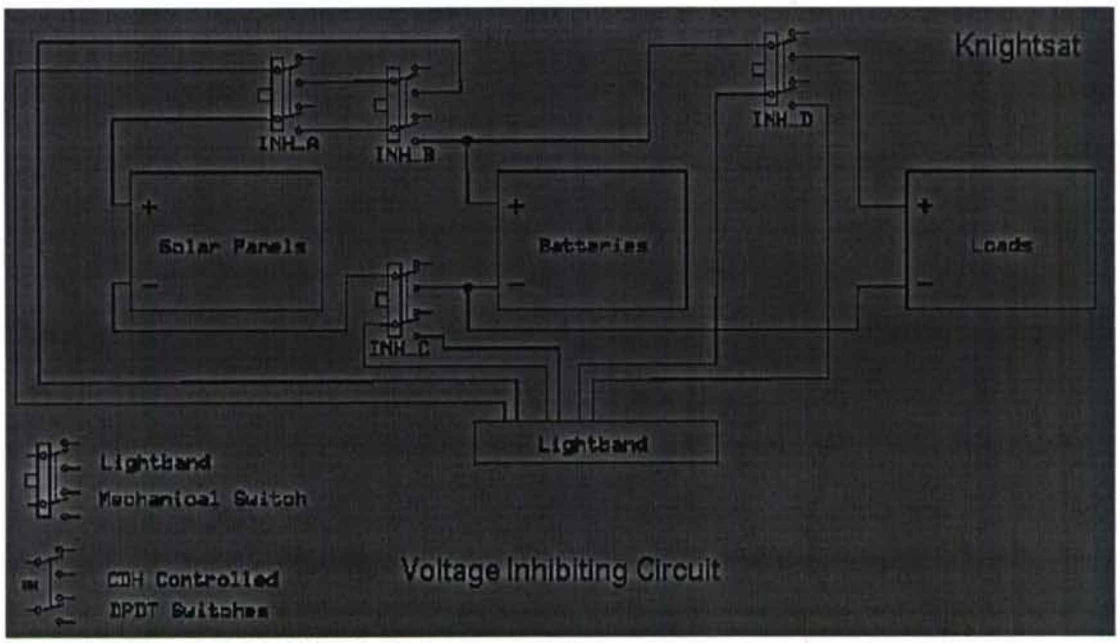


Figure 22: Electrical Inhibiting Design

The power is distributed to the subsystems of the satellite as required. Insulators will surround the battery packs as well as any other thermally discharging component to prevent the satellite's systems from melting, exploding, and/or thermally affecting the performance of any other subsystem of the satellite. The average power for the satellite on a 24 hour period is displayed in Table 1.

Table 1: Average Power during a 24 hour Period

Average Power Budget for Satellite (over 24 hours)							
Subsystem	Component	Budget (Volts)	Amperage (mA)	Power (Watts)	Time of operation	W Hrs	% of Budget
Max Power				9.072	24	217.728	
ADCS							
	Gyros	12	65	0.78	0.3	0.234	0.23
	Torque Rods	6	200	1.2	0.666	0.7992	0.79
	PPT	6	2000	12	0.1	1.2	1.19
Structures/ Payload							
	Camera	5	150	0.75	0.3	0.225	0.22
	Boom			1.8	0.05	0.09	0.09
Power and Communications							
	Kenwood TH-7 radios (Receive)	13.5	50	0.675	0.3	0.2025	0.20
	Kenwood TH-7 radios (Transmit)	13.5	420	5.67	0.3	1.701	1.69
	Computer	5	800	4	24	96	95.49
	KPC-9612 Modem	6	45	0.27	0.3	0.081	0.08
	Total	67	3730	15.145		100.533	100.00
					Excess Power	117.195	Whr
					Average Power	4.18336	W

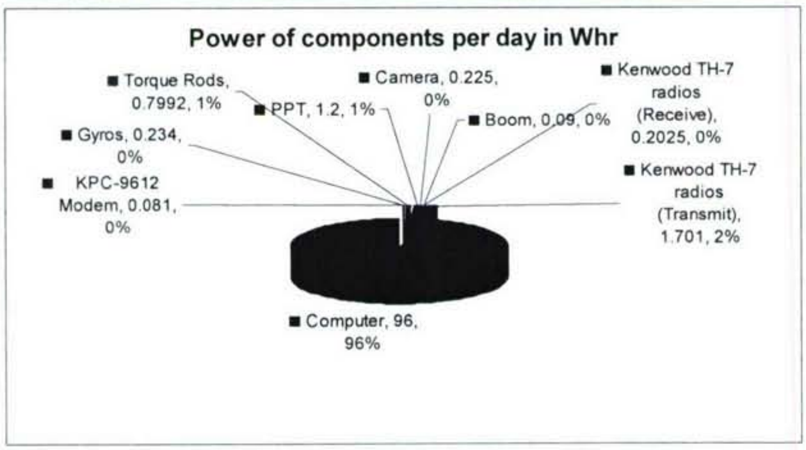
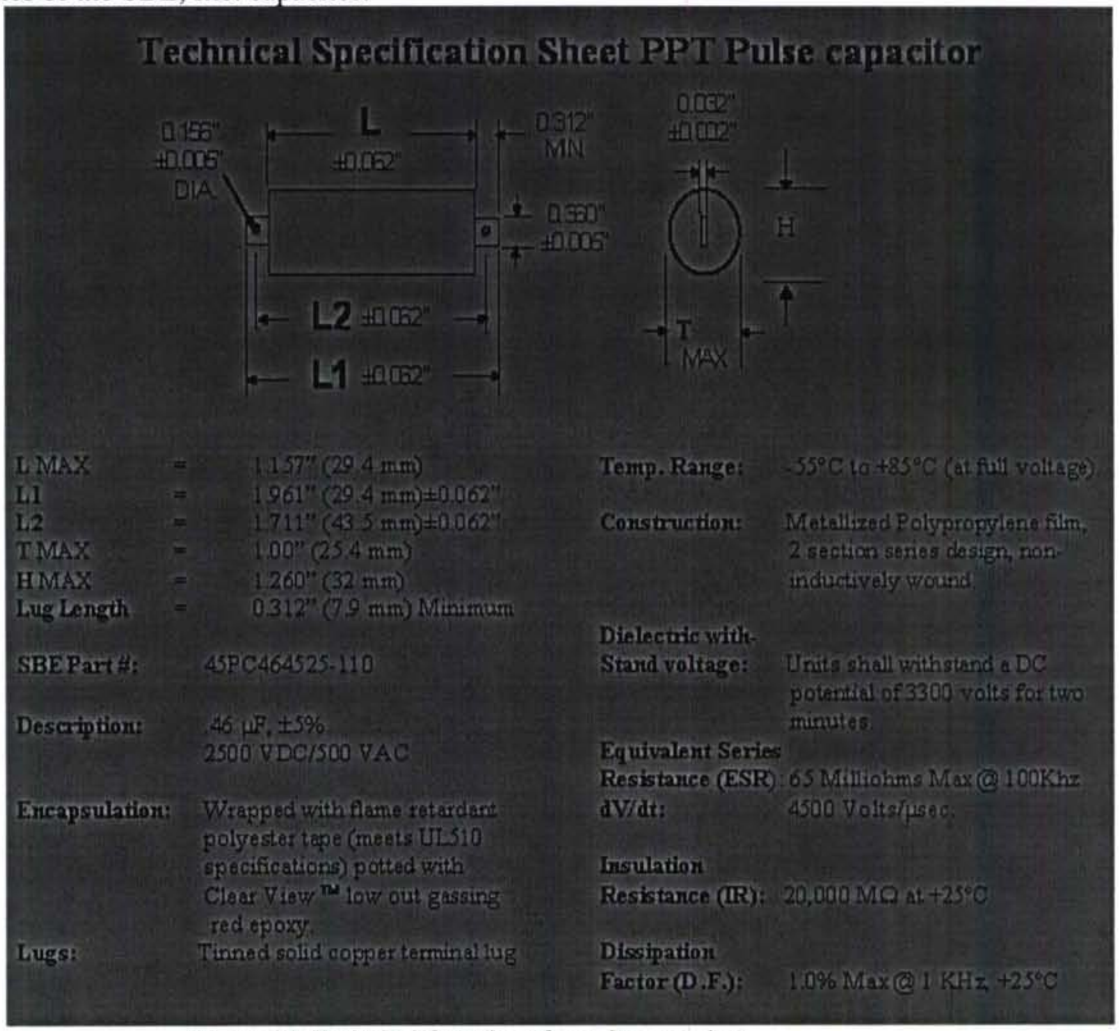


Figure 23: Average Power Budget for Satellite

Table 1 represents the consumed power by the components of the satellite organized into subsystems. Similarly, Figure 23 shows average power used on a 24 hour period for the satellite in orbit. Since most components operate for 40 minutes or less on cycle, it is acceptable for the computer, which is functional 24 hours a day, to use 96 percent of the average power. For a given twenty-four hour day, the average power will be 4.18 Watts. In analyzing the chart, there is an excess power of approximately 117.2 Watt-hour. While this number is significantly large for the estimated 20 Watts of power, it assumes accurate information when the power required to initialize and support daily operations for the boom and the pulse plasma thrusters are refined and implemented.

A substantial performance objective for the KNIGHTSAT satellite is to extend the gravity gradient boom in order to position the satellite in camera ready mode. The boom will be extended by two power solenoid and pulse plasma thrusters that require 1000 volts and .6 amps. Since the batteries and solar cells can not perform this operation, a capacitor made by SBE, Inc will be charged and later discharged across the pulse plasma thrusters and the power solenoid. Figure 24, below, shows the electrical properties of the SBE, Inc. capacitor.



L MAX	=	1.157" (29.4 mm)
L1	=	1.961" (29.4 mm)±0.062"
L2	=	1.711" (43.5 mm)±0.062"
T MAX	=	1.00" (25.4 mm)
H MAX	=	1.260" (32 mm)
Lug Length	=	0.312" (7.9 mm) Minimum

SBE Part #: 45PC464525-110

Description: .46 μF, ±5% 2500 VDC/500 VAC

Encapsulation: Wrapped with flame retardant polyester tape (meets UL510 specifications) potted with Clear View™ low out gassing red epoxy.

Lugs: Tinned solid copper terminal lug

Temp. Range: -55°C to +85°C (at full voltage)

Construction: Metallized Polypropylene film, 2 section series design, non-inductively wound.

Dielectric with-Stand voltage: Units shall withstand a DC potential of 3300 volts for two minutes.

Equivalent Series Resistance (ESR): 65 Milliohms Max @ 100Khz

dV/dt: 4500 Volts/μsec.

Insulation Resistance (IR): 20,000 MΩ at +25°C

Dissipation Factor (D.F.): 1.0% Max @ 1 KHz, +25°C

Figure 24: Technical data sheet for pulse capacitor

The pulse capacitor can distribute .46 μF and 2500 volts in DC current, which is more than what the KNIGHTSAT satellite will be operating on. The capacitor intended in the design will be charged from the solar cells and after ample power is stored, it will be discharged throughout the pulse plasma system.

There will be two Kenwood TH-7 radios on board the satellite, one to receive data and instruction while the other will transmit data from the satellite's high resolution camera and subsystems back to the ground station on Earth. Figure 25 shows the Kenwood TH-7 radios that comprise the communication system.

Figure 25: Kenwood Radio to be used on KNIGHTSAT satellite.

Attached to the Kenwood TH-7 will be a Kantronics 9612+ modem that will interface as a way to communicate between the radio and the ground station. As means of data travel, the Icom dual band Omni directional antennas will be used to send and receive data. As in figure 26, the Kenwood TH-7 radio, Kantronics 9612+ modem, and Icom Omni directional antenna will provide the communication system to the KNIGHTSAT satellite.

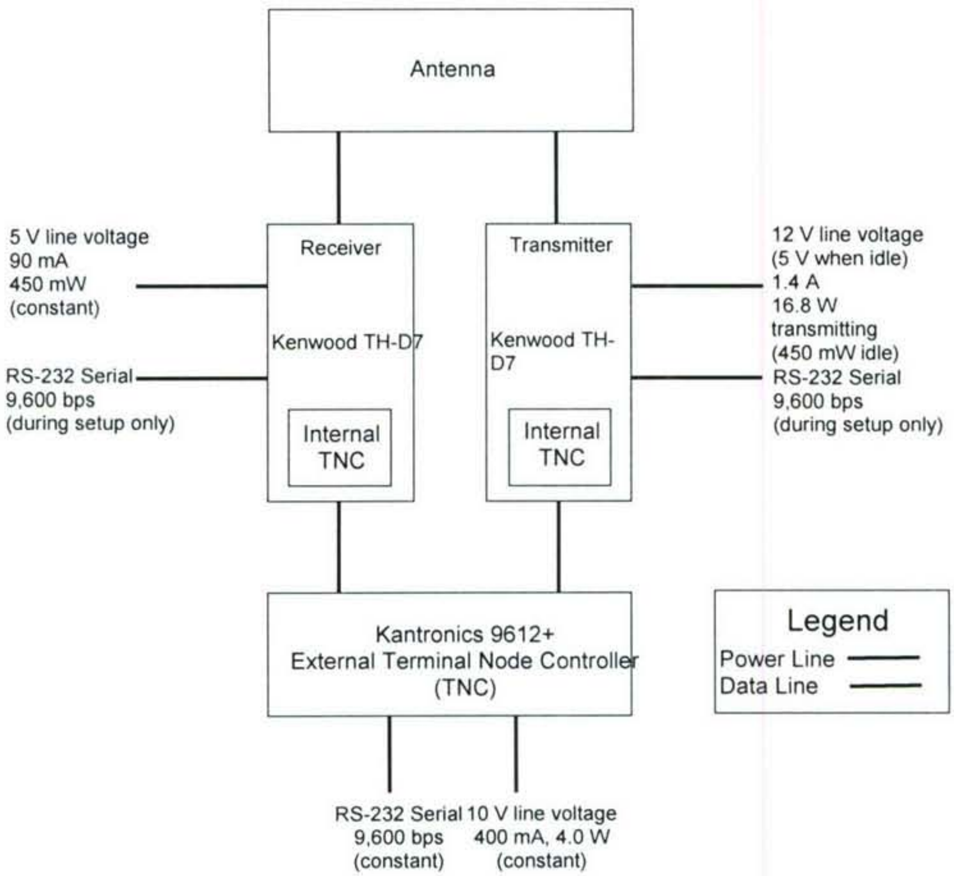


Figure 26: Communication Set-up

3.4 Ground Operation Requirements

Nanosat requires mechanical ground support equipment for use in Nanosat-4 integration operations. Mechanical ground support equipment shall be designed using a factor of safety of 5.0 for ultimate failure and shall be designed to lift the Nanosat from a single point above its center of gravity. Mechanical ground support equipment that remains attached to the Nanosat for flight must meet all requirements for flight hardware. Lifting equipment is to be designed such that it will not contact the NSS during integration and ground handling operations. Factors of safety applied to mechanism operation and holding loads may be based on either test or analysis, depending on whether the mechanism is subjected to the necessary battery of tests. Ground support must not only exceed factors of safety but must also maintain a simple loading process.

The Surrey Satellite Technology Ltd. SGR-05 receiver was designed specifically for miniature satellites in low earth orbit and has been successfully used on several satellites currently in orbit. Unfortunately this model has proven to be too costly for the team to utilize it. This resulted in the decision for the team to rely on NORAD for all of the tracking. They can provide the team with the necessary data over the first few orbits from there uploading the computer with that data should provide ADCS enough information to keep the craft stable. If there are any problems then further uploading can be completed to fix the errors.

Also, refer to the Tacking and Ground station requirements in 3.2.3.

4.0 Verification

For testing purposes, Pyrex or glass primary mirrors were purchased from Science Kit and Boreal Laboratories. Continued attempts to purchase the metal mirrors were made, however unsuccessful, due to the low availability of mirrors of the desired size on the market. However, several manufacturers are currently being perused, including the original manufacturer; Aspheric Techologies. The design calls for three, aluminum, 6061-T6 mirrors, coated in nickel, and polished to the highest degree of accuracy. Metal mirrors provide the simplest design and will be pursued until deemed unpractical.

For the optical system verification Z-Max, an optical simulation/measurement program, was used to verify that the calculations and measurements were accurate.

Recently, the optical system has been designed so that the mission objectives, obtaining an image with a 10 meter per pixel resolution, would be met. This updated design included a third mirror. Also, new calculations were used to ensure that these measurements were correct. It was found that by adding the third mirror, and performing the calculations at 500 km in altitude, that an image less than 10 meters per pixel will be obtained. If the space craft is launch closer to earth, at 350 km, an even better image will be obtained. Below are sample images obtained from the testing Celestron Cassegrain with the CCD Camera purchased from XL Imaging.

The stated mission for the Knightsat, requires that upon receiving the images, and fixing the distortions, that the images would be combined into one "stereo" image. Currently the Knightsat payload team in testing the ANNA program, which is produced by Photo-Illusion Inc. The ANNA program can arrange the images through real-time observations, as well as depth paint, and render several files at a time. The requirements are as follows: 256 MB of Ram, 10G hard drive, and 2GHz Processor. All of these requirements have currently been met by the Knightsat ground station, located at the Kennedy Space Center in Titusville, Florida.

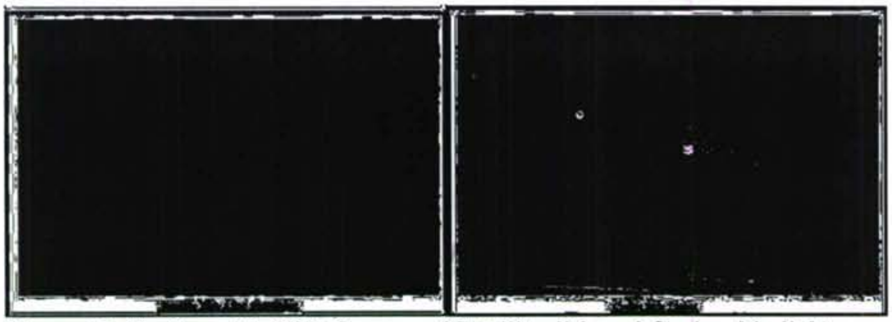


Figure 27: Testing with CCD Camera of Satellite dish and Cruise ship lights

Figure 27 above, demonstrates some of the testing that has been done with the CCD camera. The Knightsat optics team was made aware of the camera's sensitivity and the effect of vibration and, as a result, blurring. The Matlab Image Processing Toolbox will be implemented into Knightsat's ground station, in order to clarify any consequential blurring. Below are figures provided by the Matlab Image Processing Toolbox, demonstrating the clarifying effects of this program.

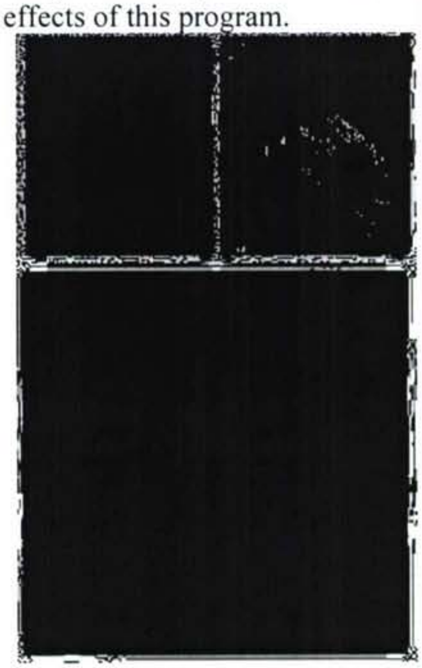


Figure 28: Examples of Image Processing Toolbox

In order to ensure the payload is operating properly, a health monitoring system was also integrated into the payload sub-system. Below, in figure 29, a visual representation is shown, presenting the health monitoring system.

The health monitoring system for the payload camera will be base on basic ping and pingback program. While the camera remains connected and powered on, the ping and pingback times will be recorded in a data file to be described in the following pages. This file will be transmitted in conjunction with any photos taken.

From the flow chart, if the pingback is not received, the program will call the health monitoring systems for the battery and solar panels to run and respond, as well as send a second ping to the camera. This will allow a consolidated report on these three subsystems, in case of camera malfunction. The second camera ping will either confirm the lack of connection between the camera and computer, or give a response, meaning the first ping attempt may have simply been timed-out. If any subsystem does not respond favorably, an error will show in the data file. Regardless of errors, the program will repeat itself at set intervals.

If the camera experiences a lasting malfunction (i.e. went 'dead'), the data file will show the time of this malfunction. This will help in problem solving for future missions. For instance, if the camera goes down around the time of a solar flare, the new design will need to shield the payload better.

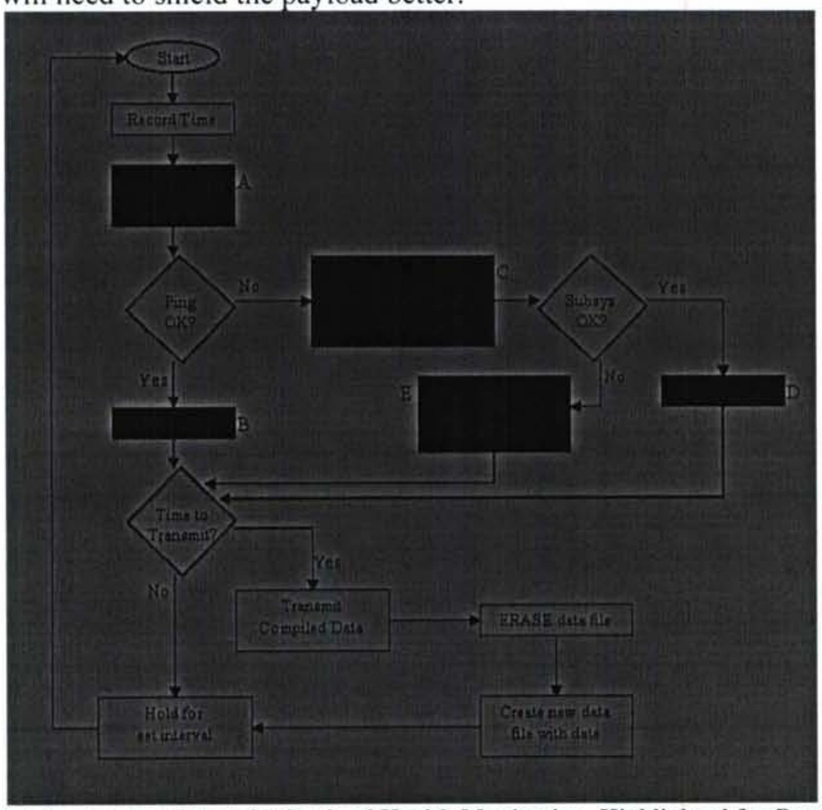


Figure 29: Logic Flow Chart for Payload Health Monitoring, Highlighted for Data File Explanation

The data file will read as follows:

Data File		Corresponding Flow Chart Item	
Date: 05.25.07		@ Start	Camera Working Properly
Ping	01:03:19.3	A	
CAM	01:03:20.4	B	
Ping	01:38:54.2	A	
CAM	01:38:54.9	B	
Ping	02:13:12.8	A->C	Camera Malfunction, not powered on. Sol and Bat Health Monitoring Response Pos.
CAM	00:00:00.0	D	
SOL	02:15:37.6	D	
BAT	02:15:23.9	D	
ERROR IN CAM		E	
Ping	02:53:13.6	A	Camera Reset, Working Properly
CAM	02:53:14.2	B	

The file opens with the date. From (A), the first ping is recorded with a timestamp. While the camera is operational, the file is updated from (B) to read "CAM" along with a pingback timestamp. This loop will be repeated at intervals to be determined, depending on whether the camera will remain powered on continuously or only as needed.

From the data file, the ping that took place at 02:13 shows that the camera did not respond. Following a null response on the flow chart, a second ping is sent to the camera, and the file calls on the health monitoring system(s) for the solar panels and battery. This example shows that the health monitoring provided a positive response at the time indicated, however the camera still did not respond (timestamp 00:00:00.0) so we see the "ERROR IN CAM" stamp directly below these values. This procedure gives us somewhat of a redundancy in the health monitoring; we are able to see if the camera quit working in conjunction with another malfunction, all within one loop and file. Regardless of error(s), the health monitoring loop will repeat to show if the error is constant (camera is dead) or varying (battery charge runs out before the satellite is out of Earth's shadow, causing camera to go on and off intermittently).

However, for this example, the camera does respond favorably to the next ping attempt. This could occur if the camera was set to be powered off during the last ping, and now power has been restored.

→ Things to note in the programming ←

- Max allowable time for pingback (A, C) (4 seconds)
- Max allowable time for subsystem health monitoring response (C) (3 minutes)
  <This depends on other health monitoring loops and the time to run them>

- Hold time before restarting loop <How often per orbit / per hour? >
- How often to erase data file <every transmittance, or save for 48 hours>
- In case of dead camera, only run the (C) loop every 10^{th} negative ping response, up to 100 negative responses in a row. At this point, slow health monitoring to once per orbit and only run (A-B) loop.

The preliminary design of the bus was a cylindrical structure manufactured out of Aluminum 6061-T6. The required factory of safety (FOS) for the bus is 2.0. When the force analysis was preformed on the preliminary design, utilizing COMOS from Solid Works, the FOS was 1.98. This did not meet the requirements for the factor of safety set forth by the Air Force. To compensate for the failure in FOS the structure was recreated as a solid structure, without a honeycomb pattern on the walls. The structure was chosen to be cylindrical to reduce the stress levels during launch. The more cylindrical the satellite interface plane (SIP) the less stress the structure will encounter from the light band when launched. Originally, flexible solar cells were going to be utilized to compensate for the cylindrical structure. Upon further investigation it was discovered that flexible solar cells had never been tested in space. As a result this information it was a decided that non-flexible solar cells would be a more reliable option. The use of non-flexible solar cells required the bus to have flat surfaces in order to be able to mount the solar cells to the structure. The preliminary structure was modified to have a cylindrical interior and isocagon exterior. This modification enabled the solar cells to be mounted to the exterior surfaces.

Attitude Dynamic Control Systems:

For the Pulse Plasma Thrusters, the specific impulse will be calculated using equation 4.0.1, found in Appendix B, as some of these values are not yet known. A reliable number may not be calculated until a prototype can be tested. Other calculations of interest include the ablation rate determined using equation 4.0.2 resulting in $3.6\frac{\mu g}{cm^2}$. This, in conjunction with simulations, will be used to determine the amount of fuel that will be available for ablation. With a Delrin density of $1.41{}^{g}\!/\!_{cm^3}$ the mass burn per pulse can be calculated using equation 4.0.3 giving $6.876\frac{\mu g}{pulse}$, from this the energy per pulse can also be found using equation 4.0.4, being $3J$.

The placement of the individual PPT's is critical as certain coupled forces are required to produce desired motion. As there is virtually no reaction forces acting on the structure as thrusters are fired all forces put into the satellite must be part of a coupled pair so as to produce motion purely in one direction.

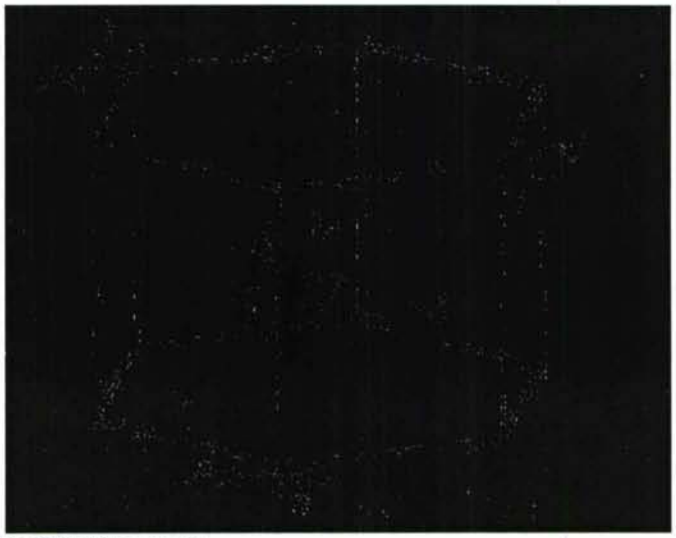


Figure 30: PPT Placement

The figure above illustrates proper placement for the PPT's. Each numbered triangle represents a nozzle direction with two PPT's at each location oriented to fire at 45° to each other. These thrusters will be fired in conjunction to produce the desired motion. The table below describes the firing patterns which will create motions in the indicated directions.

Table 2: Firing Sequence

Motion	X-axis	Y-axis	Z-axis
Translation	(2,3) (6,7)	(4,5) (1,8)	- -
Rotation	(1,4) (5,8)	(2,7) (3,6)	(1,5)(2,6) (3,7)(4,8)

To test the torque rod two methods will be deployed; a bench test method and a hanging test method. The bench test method is currently being built, as at this stage in development it is the more important test. The bench test uses a magnetometer to read the magnetic field created by the torque rod. Figure 31 shows a schematic of the bench test.

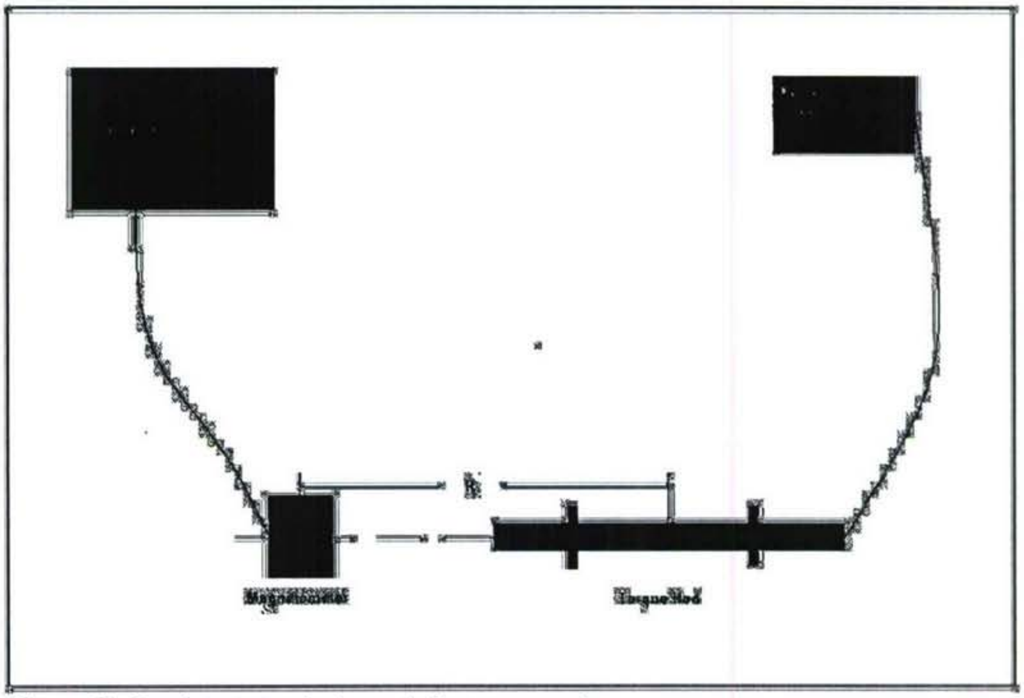


Figure 31: Bench test stand schematic for torque rod

The torque rod and magnetometer will be secured to the test stand so that they cannot move. The magnetometer will lie directly in line along the x- direction with the torque rod. This is so that it is easiest to read the magnetic field of the torque rod, since the magnetic field will be projected along that direction. The magnetometer is connected to a computer and the torque rod is connected to a power supply of 6 volts. An important feature to note is that this entire bench test stand must be rid of all metal so that no false readings can be taken. The tests will be held out doors away from any metallic or magnetic objects. The only objects containing metal that will be in close proximity will be the computer and power supply.

The hang test will be done once the torque rods are fully developed. A mock up version of the satellite will be constructed to hold each torque rod on its axis. The entire unit will be suspended and set into a spin. The torque rods will then be used to stabilize it. Of course, since it is suspended along one axis, only tow torque rods will be needed at a time. To make sure each works fine it will be suspended from two different axes to ensure each axis can be stabilized.

Finally, a gauss meter will be used to measure the exact magnetic field of the rod as well as the retained magnetic field once the current is not running through it. The amount of torque that each rod will produce is dependent upon the amount of current that is passed through it. Figure 32 shows a plot of torque versus current where the current goes up to 0.4 mA.

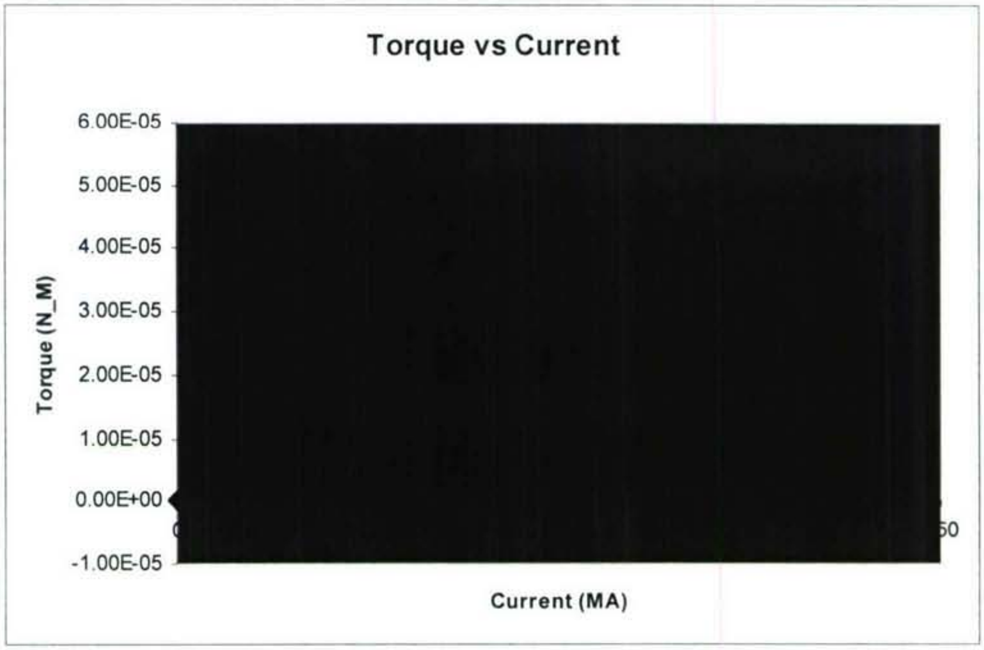


Figure 32: Toque vs. Current for each torque rod

The two main changes made to the torque rod design come from the material for the core and the dimensions of the core. The EFI 50 material was selected because it has a very high permeability as well as a low retained magnetic field. The dimensions of the torque rod were made based on the largest amount of available space in the satellite. 12 inch torque rods fit in the bus very well and the bigger the rod the more torque that can be produced. The copper wire was changed to 26 gauge because it will allow for more turns resulting in a larger magnetic field, also there are two layers of windings which will also allow for more amount of turns for the length of the rod.

In equation 4.0.8 Lg is the gravity gradient torque applied to the spacecraft. This equation is a function of the mass of Earth, the gravitational constant, the center of mass vector, and the inertial matrix. The gradient torque can be changed by varying the values of the inertial matrix, which is the purpose of extending the boom.

Preliminary designs of the gravity gradient boom included a boom no shorter than five meters. The boom integrity was questioned at this length, and with new information, the boom is now between two and four meters in length. This new length will put less lateral stress on the boom. Also, this will reduce the spindle profile and save space in the satellite. Figure 33 shows the restoring force caused by the boom in a simulation performed by Bhargav Gajjar.

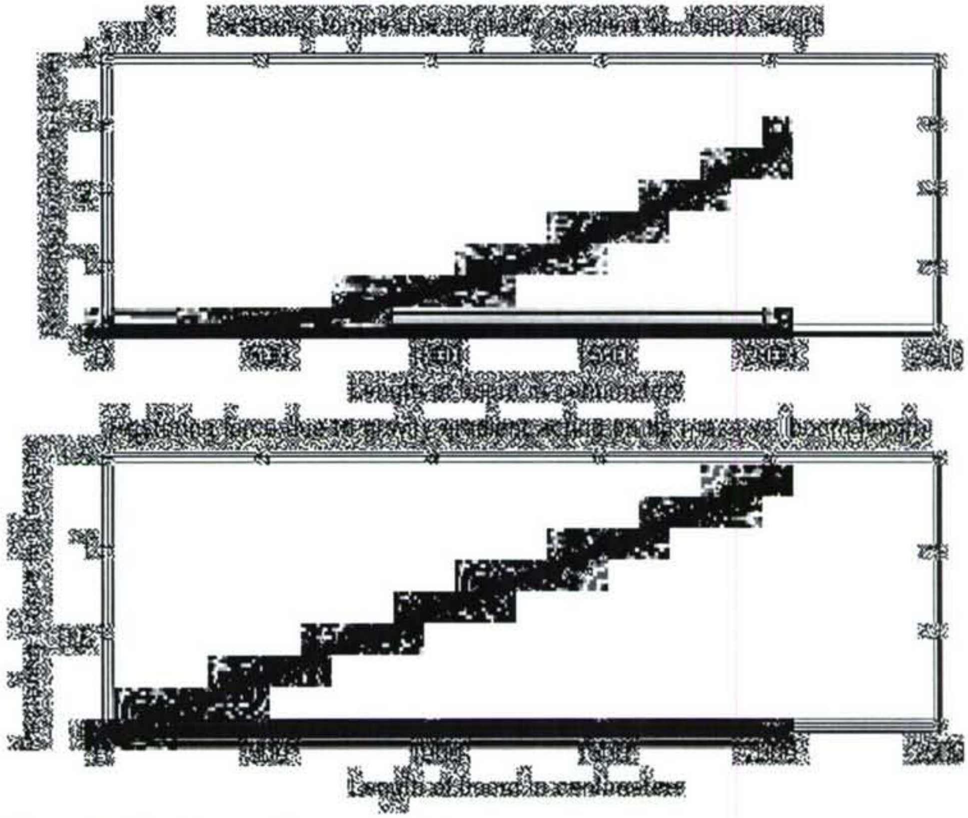


Figure 33: Results found from simulation

Power / Communications

In order to determine the life and capacity of the batteries, solar cells, and power source, engineering analysis was used to accurately portray the life of the space system. The first characteristic of the satellite involves the sun incident angle between the normal vector of the solar cells and the sun rays. For this spacecraft, $\Theta = 1$ deg. Using data from the Kyocera Crystalline Solar Cells, the cell efficiency was .140, the max operating Temperature was 301 Kelvin, the End of Life degradation efficiency was .097, the solar intensity was 1353 Watts per meter squared, the Silicon packing factor was .91, The desired power was 20 Watts, the batter capacity was 4 amp hours, and the area of the solar area was 2.65×10^{-4} meters squared. The bus voltage and array voltage is centered on the number of batteries used in the system. The power and communications group estimated the number of batteries to be 16. Using Equation 4.3.1 and Equation 4.3.2, refer to the Appendix for listed equations, the bus voltage was 20 Volts and the array voltage was 24 Volts. Next, the charge capacity was calculated using Equation 4.3.3, where BC is the battery capacity and N is the number of batteries. The charge capacity was 64 amp hours. The charge voltage was the same as the array voltage; therefore, the Power due to charge time was calculated using Equation 4.3.4. The charge power calculated was 102.4 Watts.

For the End of Life power, the charge power was added to the desired power of 20 Watts. The End of life power yielded 122.4 Watts. Using equation 4.3.5, the thermal efficiency of the thermal cells was computed using the max operating temperature of 302

Kelvin. The thermal efficiency of the solar cells was 99.5%. The efficiency degradation due to radiation exposure was next calculated using the End of Life efficiency of .097%. Equation 4.3.6 yields degradation due to radiation exposure of .903%.

The beginning of life power was then calculated using equation 4.3.7, where the efficiency of the angle is equal to the cosine of the worst case incident angle. The beginning of life power was calculated to be 233.77 Watts. The beginning of life power was then divided by the beginning of life efficiency multiplied by the solar intensity. This process is outlined in equation 4.3.8. This yielded the Area of the cells which was 1.23 m^2. The area of the array was then calculated using the cell area divided by the silicon packing factor. Equation 4.3.9 shows the equation used to obtain the area of the array which was 1.36 m^2. The number of solar cells was then calculated by dividing the cell area by the area of one solar panel. The equation is outlined in Equation 4.3.10. The total number of solar cells required is 972.

The uplink and downlink budgets were calculated. Table 8 and Table 9 show our links for a dipole antenna.

5.1 Testing Requirements Considerations

The satellite prototype will be thoroughly tested by the KnightSat team using University of Central Florida resources. Testing and verification of the power and communications subsystems will guarantee mission success. Primarily, the power and communications subsystems are concerned with acceptable completion of the following test: a Bake Out Test, Electro-Magnetic Interference Test, and Electrical System Aliveness and Functional Test. In the event the subsystems do not meet outlined specifications from the university or AFRL, problem reports will aim to document and analyze the problem before a rational solution is produced.

5.2 Project Milestones and Evaluation Criteria

The milestone tasks designed to produce a fully-functional satellite are outlined below in order by the date the task is to be achieved. In between the timeslots below, there are tasks scheduled to help meet milestones but the tasks themselves are not the milestone to be achieved, shown below in Table 3.

Table 3: Power / Communications Milestones

Milestone I.D. Number	Milestone Description	Intended Date of Achievement
1	Amateur Radio License	January 24, 2007
2	Avionic Chassis Development	January 20, 2007
3	PC Board Completion	January 20, 2007
4	Initial Testing and Specifications of Components	January 31, 2007
5	Intermediate Level Testing	February 17, 2007
6	System Integration	March 18, 2007
7	Final Integration Report and Documentation Sign-Off	April 19, 2007

The milestones described above are monumental requirements to ensure the successful completion of the mission. The milestones above are important guidelines to help reach the end satellite product and are subject to change based upon circumstances and availability.

5.3 Project Tracking Procedures

To ensure a successful project and functional product, all members of the power and communications team must be well informed of tasks, schedules, and deadlines. This process will be completed by weekly briefings outlining work for the week and the group's tasks manager, otherwise known as a Gantt chart. By these measures, individual team members can inform the group of progress and modify the master schedule accordingly to accommodate any work that may be ahead or off schedule.

5.4 Statement of Work

The following sections outline the work completed by the power and communications group.

5.4.A Amateur Radio Licensing

Selected members of the group were given the tasks to acquire amateur radio licenses. Licenses are required to track and communicate with the satellite once in orbit. After preparing and taking the amateur radio test, the mission to communicate pictures from the satellite can be carried out by using amateur radio waves.

5.4.C.1 Battery Box

In order to gain a definite components list, individual components were measured and weighed to find the specifics for the satellite. The batteries were weighed and measured to specification. The weight of each battery is 160 grams. The dimensions of the batteries are 1.30 inches in diameter by 2.34 inches in length.

5.4.C.2 Modem Box

The modem was measured and weighed according to specification. The modem weighed 100 grams. The dimensions of the modem were 6 inches wide by 6 inches long by .25 inches high.

5.4.C.3 Dual Radio Box

The radio were disassembled and measured according to specification and intent on the satellite. The radios weighed 100 grams each. The dimension of each radio is 4 inches in length by 2 inches in width by .5 inches thick.

5.4.C.4 PC Board

Local power regulation of the PC Board has been completed. The components of the PT5110, PT5101, PT5102, PT5103, VNQ660, and L6370 regulate and deliver power from the solar cells or batteries to the systems that require current and voltage. The currents and voltages are separated and regulated via these components into 12 volts, 9 volts, 5 volts, and 3.3 volts.

5.4.D.1 Testing of Components

Aliveness and functional test of ordered components are immediately prepared upon arrival to verify parts meet specifications outlined by the manufacturer. Functional test ensure that devices will work properly once in orbit. Aliveness and functional test range from communication checks to voltage regulations.

5.4.D.1.1 Radio Operations Check

Radio operations check was completed by sending data packets and voice signals throughout the ground building. The radios are Kenwood TH-D7 radios capable of sending and receiving voice and data packets over a large range. Data packets were also sent through the computer to simulate the capability of the radios sending large data packets.

5.4.D.1.2 Modem Operations Check

Modem operations check was completed simultaneously with the Kenwood TH-D7 radios. The modem was used as a transmitter of data from the computer to the radios. Results from the test indicate the modem is fully-functional and capable of transmitting data through computer software.

5.4.D.1.3 Solar Testing

The solar cells have been checked out and ordered. The solar cells are Kyocera 1.12 Watt mini modules. The panels have been analyzed to provide power to the Knightsat satellite. The solar cells will be bonded to the duo decagonal surface of the satellite via silicone adhesive. From the solar cells, the power will draw into the charging circuit.

5.4.D.1.4 Battery Testing

The battery box in the circuit has been properly tested and evaluated in the charging circuit. The batteries are to be used for power dispersion should the solar cells enter a period of eclipse. The voltage regulators then assess the power needs of the satellite to route power to demanding systems.

5.4.D.1.5 Power Components Testing

The power components of the satellite have been tested and are per spec. The power components includes the integrated power circuit that holds the PT-51 series and the TPS. After all testing has been finalized and confirmed, integration of the satellite will be put into action.

5.4.D.2 Intermediate Level Testing

Intermediate level testing has begun by evaluating levels of vibration, EMF interference, frequency matching, thermal and vacuum. By interpreting information provided in these test, proper implementation of the satellite is capable of being assembled. Testing these specific areas allow the structure to be stable and fully-functional while in orbit.

5.4.D.3 Integration

The mission of the KnightSat satellite is to take a picture of the earth from orbit. The integration team is drafting an integrated system test plan which will address requirements that need to be validated or verified by system tests to achieve this mission. To that end, the various subsystems of the team have designed and built components capable of completing this mission successfully. First, however, each subsystem must pass a qualification test, including functional tests after each exposure to vibration, shock, and thermal vacuum environments. Integrating the subsystems is the final step before flight evaluation and launch.

Cabling Procedure:

Materials and Equipment:

1. 1 male 9-pin DB connector and cover
2. 1 female 9-pin DB connector and cover
3. 1 male 15-pin DB connector and cover
4. 1 male 26-pin DB connector and cover
5. 2 3.5 mm microphone jacks
6. 2 2.5 mm speaker plug jacks
7. Soldering Iron
8. Soldering Sucker
9. Solder Wire
10. Magnifying Lens
11. Soldering Workbench clap
12. 2 3-foot length 5 conductor shielded cable
13. 1 3-foor length 10 conductor shielded cable

Procedure:

1. First we began constructing the 232 serial modem cables that connects the modem to the computer. One end of the wire will be equipped the 26-pin male DB connector which attaches to the modem. The other end of the cable contains a 9-pin female DB connector that attaches to the computer serial port.
   a. Below is a picture of the cable that we constructed.

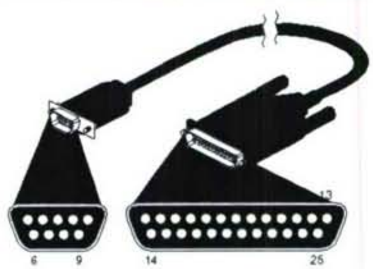


Figure 34: Cables
Picture courtesy of: techpubs.sgi.com/.../db9.db25.pin.cable.gif

b. We used the wire scheme shown in the table below to connect the pins for proper communication between modem and computer.

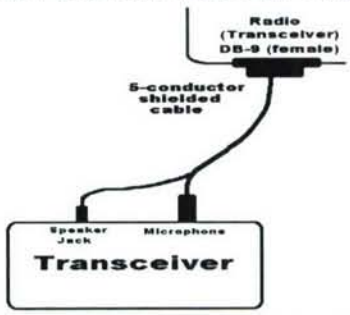
Table 4: Pin Locations

Kantronics 9612+ (25-pin male connector)	Computer (9-pin female connector)
Pin 2	Pin 3 TXD
Pin 3	Pin 2 RXD
Pin 4	Pin 7 RTS
Pin 5	Pin 8 CTS
Pin 6	Pin 6 DSR
Pin 7	Pin 5 SG
Pin 8	Pin 1 DCD
Pin 20	Pin 4 DTR
Shield	Shield

c. We were sure to use the proper precautions when soldering.

d. We checked to see if the pin-out layout was correct by testing conductivity using a voltmeter.

2. Next we constructed the cable that connects the modem to the high speed radio. One end of the cable will contain a 15-pin male DB connector which attaches to the modem. The other end of the cable will be connected to a microphone plug and a speaker plug. Below shows the cable for Port 1, the difference is a DB-9.



Picture courtesy of: Kantronics User Manual page 25

Figure 35: Transceiver Used in Testing

a. We used the wire scheme shown in the table below to connect the pins for proper communication between Port 2 of the modem and transceiver using the supplied 3-foot 5 conductor shielded cable.

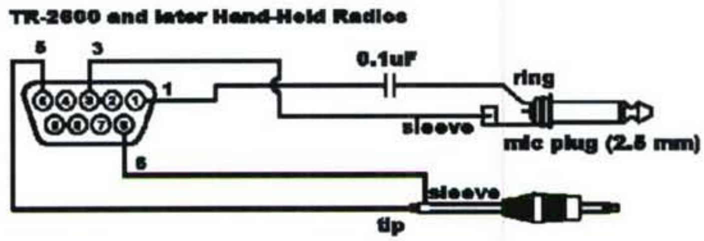
Table 5: Secondary Pin Locations

Kantronics 9612+ (15-pin male connector)	Kenwood TH-D7 Connectors
Pin 1	3.5 mm Microphone sleeve
Pin 2	2.5 mm Speaker tip
Pin 3	3.5 mm Microphone tip
Pin 9	2.5 mm Speaker sleeve

b. We made sure to use the proper precautions when soldering.

c. We checked to see if the pin-out layout was correct by testing conductivity using a voltmeter.

3. Lastly, we constructed our final wire, which is the wire that connects Port 1 of the Kantronics modem which operates at 1200 baud and is connected to our other transceiver.
    a. Using the other supplied 3-foot 5 conductor shielded wire we followed the table below and constructed our wire.



Picture courtesy of: Kantronics User Manual page 39

Figure 36: TR-2600 Hand Held Radio

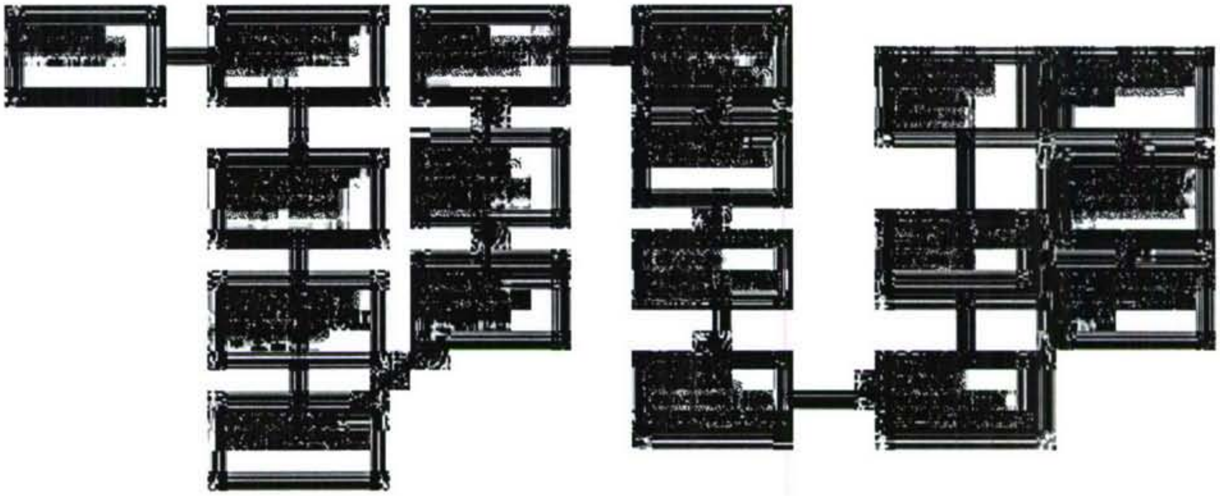
Table 6: Pin Locations

Kantronics 9612+ (9-pin male connector)	Kenwood TH-D7 Connectors
Pin 1	3.5 mm Microphone tip
Pin 3	3.5 mm Microphone sleeve
Pin 5	2.5 mm Speaker tip
Pin 6	2.5 mm Speaker sleeve

    b. We made sure to use the proper precautions when soldering.
    c. We checked to see if the pin-out layout is correct by testing conductivity using a voltmeter.

4. After performing the cabling procedure described above we successfully completed the cable assembly needed for radio to modem to computer communication.

4.2 Mission Operational Sequence



5.0 Organization

		KnightSat Group Leader		
		Rebecca Kendall		
		Structures/Payload		
		Danielle Grant-Group Leader		
Power Box	**Computer Box**	**Battery Box/Bus Structure**	**Imaging/Optics**	**Imaging/Optics**
Bobby Shanahmi	Lashanda Oliver	Danielle Grant	Chad Moody	Rebecca Kendall
		Attitude Dynamic Control System		
		Daniel Stuhr- Group Leader		
Torque Rods	**PPTs**	**Gravity Gradient Boom**	**ADTS**	**Power/Comm**
Jason Dunn	Ruben Salles	Keegan Ford	Daniel Stuhr	Jessica Vega
		Power and Communications		
		Timothy Young – Group Leader		
Power Distribution	**Solar Cells / Energy Storage**	**Antenna Transmittance**	**Power Regulation**	**Energy Storage/Power/ Comm Subsystem**
Max Adsit	Jerome Dilworth	Arthur Hayes	Joe Wiley	Timothy Young
		Mechatrionics Integration Team		
		Randal Allen - Team Leader		
Power/Communication	**ADCS**	**Payload**		
Luis Readdy	Ed Daughtery	Emily Afifi		

6.0 Conclusion

The KnightSat project has passed through two reviews during this reporting period, a Critical Design Review, CDR, and Proto-Qualification Review, PQR. Given that primary mission has been reformed and a new secondary mission has been selected, a reevaluation of the mission and system requirements is taking place. An effort is being made to integrate the satellite subsystems into a cohesive satellite system. The necessary student and design and build the various satellite systems and payload has been acquired the organizational structure for the KnightSat teams has been formed.

A web based database has been setup to centralize the storage and distribution of documents to the team and the PIs. Also, a web based configuration management database has been established in order to produce and centralize the necessary documentation easing the burden of tracking and storing the required documentation. Efforts are now complete in establishing the satellite ground station, (SGCC) at the Florida Space Institute facilities. Facilities and resources that will be used to design, fabricate, test and store the engineering and flight models of the spacecraft, the payload and its subsystems have been identified and allocated for the KnightSat spacecraft.

7.0 References

References

Dr Rolland, Jannick of CREOL, The University of Central Florida

Griffin, Michael D. and James R. French. Space Vehicle Design. Reston, VA: American Institute of Aeronautics and Astronautics, 2004.

Larson, Wiley J. and James Wertz. Space Mission Analysis and Design. Torrence, California: Microcosm, Inc, 1992

Shilling, Tim, Anthony Lowery and Grayson McArthur. "DINO PDR". 2004. University of Colorado at Boulder. 12 Oct. 2006. http://www.sgc.colorado.edu/dino

Brown-Webster, Brandon, Rogan Flowers, Peter Grossman, Richard Park, Stephan Pratt, Joeseph Sanderson, Kristopher Velez, and Michael Weronski. "Power, Thermal and Environment". November 18, 2004. Virginia Polytechnic Institute and State University. October 12, 2006. http://www.aoe.vt.edu/~cdhall/courses/aoe4065/FDReports/PTE04.pdf

Lee J., Ng A., Jobanputra R. "On Determining Dipole Moments of a Magnetic Torquer Rod - Experiments and Discussion." Canadian Space Agency, 2002.

Burton, R.L., and Turchi, P.J., "Pulsed Plasma Thruster,"*Journal of Propulsion and Power,* Vol. 14, No. 5, 1998, pp. 716-735

Schaub, Hanspeter, and Junkins, John L., "Analytical Mechanics of Space Systems" American Institute of Aeronautics and Astronautics, Inc. 2003

Piscayne, Vincent L., "Fundamentals of Space Systems," Oxford University Press, Pages 245-328; New York, New York 2005

3DM GX1 Informational and Spec pages
http://www.microstrain.com/3dm-gx1.aspx

Honeywell HMR2300r 3-Axis Strapdown Magnetometer Information and Spec pages
http://www.ssec.honeywell.com/magnetic/products.html

Surrey Satellite Technologies Ltd SGR-05 GPS receiver
www.sstl.co.uk/documents/Subsys_**SGR05**_HQ%5B1%5D.pdf

8.0 Appendices

8.1 Appendix A
Mechanical Interface Control Drawings:

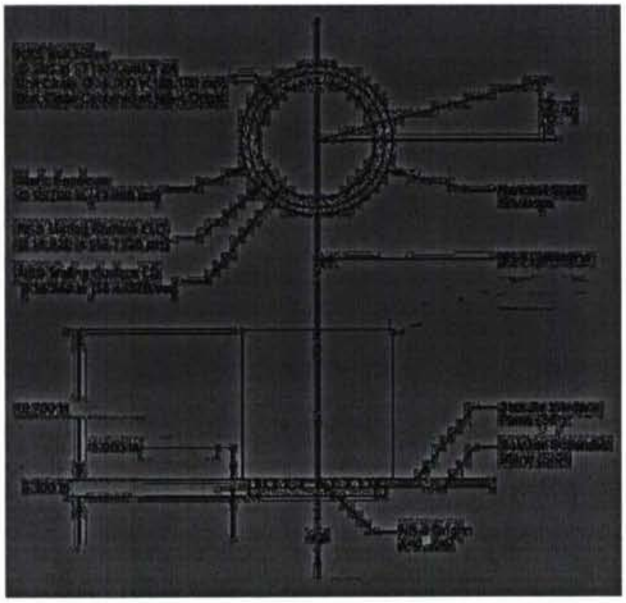


Figure 37: Lightband Schematic

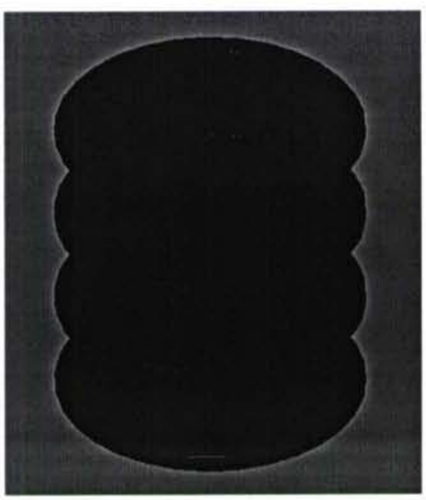


Figure 38: Final and Preliminary Designs

Figure 39: New Imaging System (after obtaining desired aluminum mirrors) Nasmyth

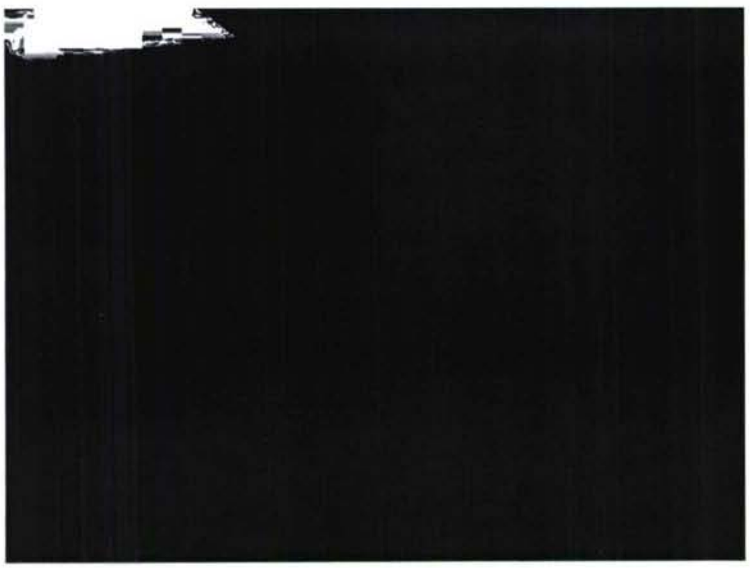


Figure 40: Ground Station Being Assembled

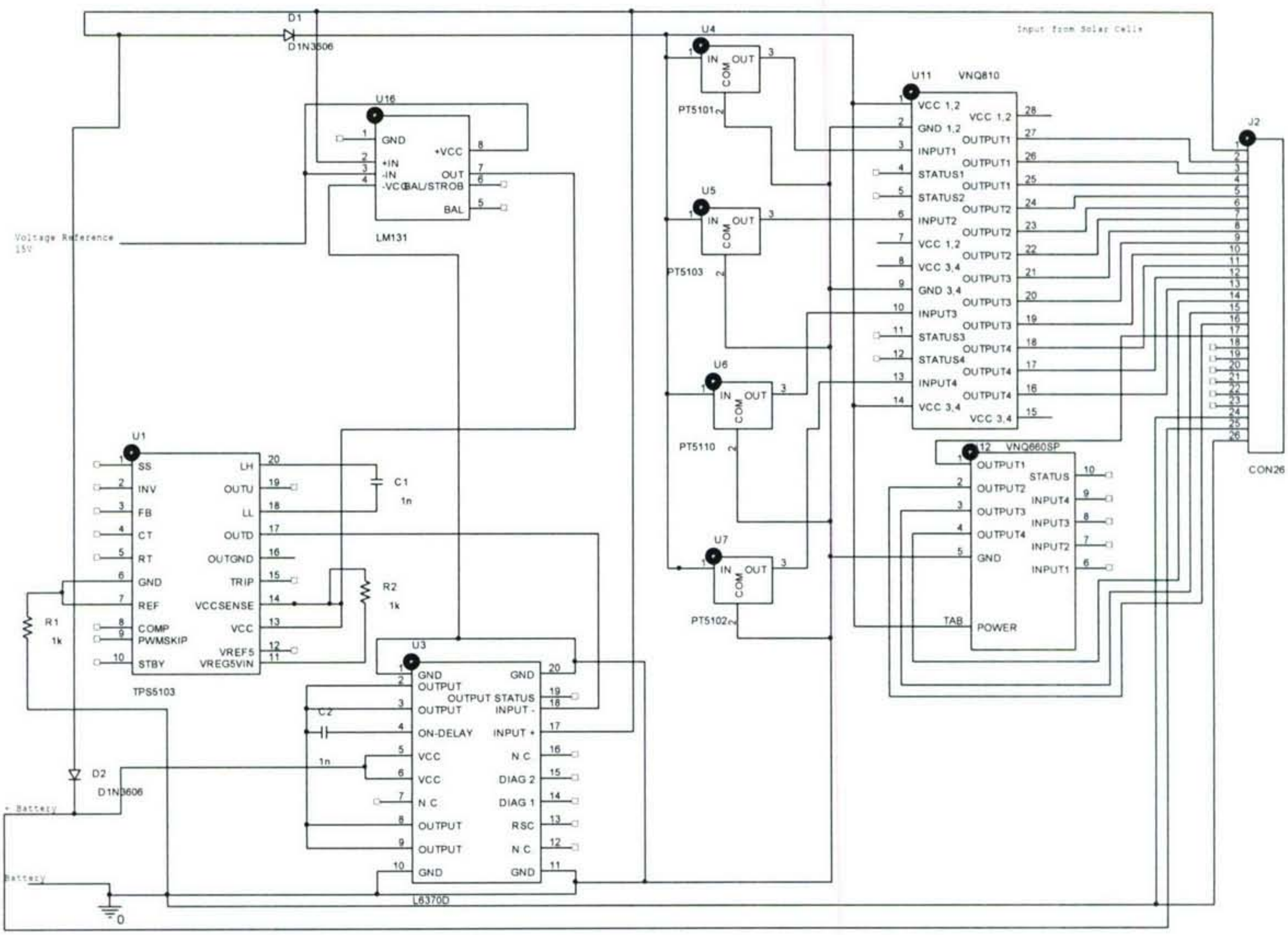


Figure 41: General overview of Power System Schematic

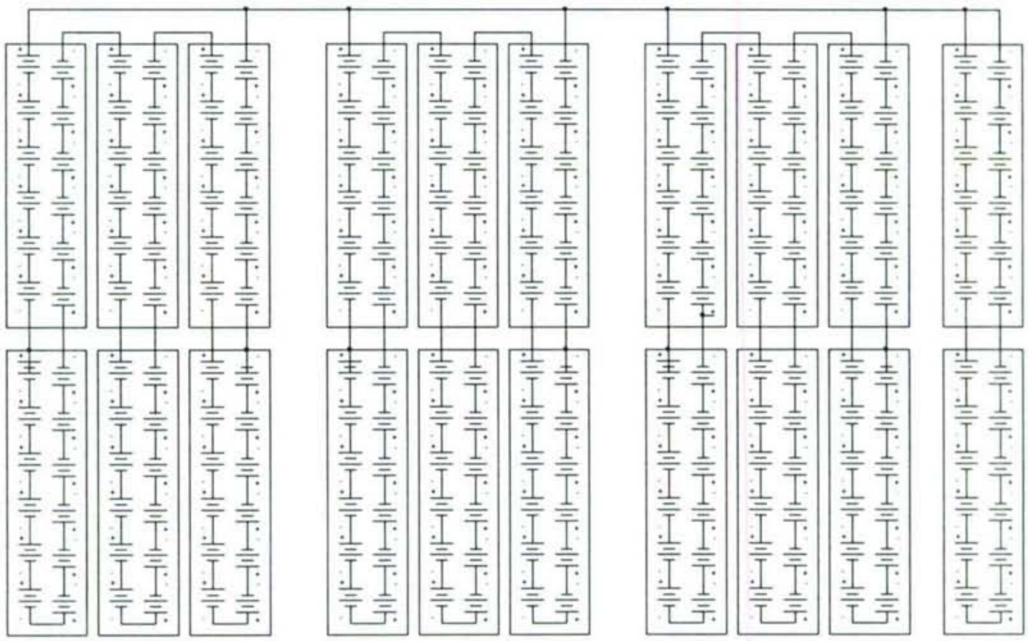


Figure 42: Solar Panel Array

The local power regulation component of the power system distributes required watts, amps, and voltages to systems requiring stable power as shown in Figure 43.

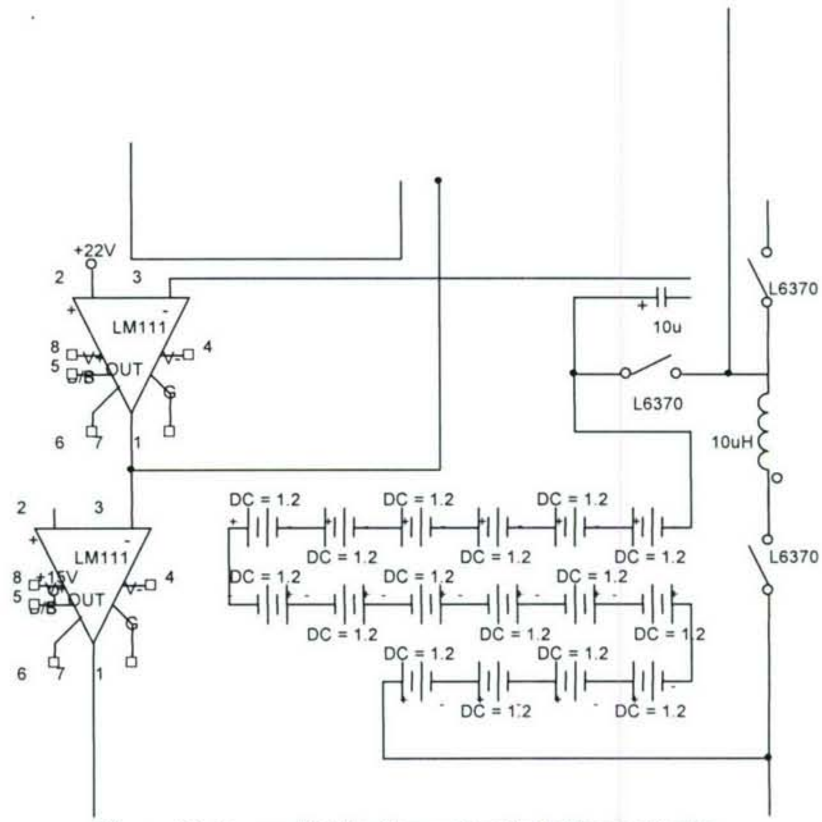


Figure 43: Power distribution system including batteries

The comparators are the control for the Pulse Width Modulation Controllers (TPS5103). If the line voltage were to fall below 20V, then the system activates the Pulse Width Modulation Controller (PWM) to change the frequency of the switches. This ensures that the batteries continue to provide ample power to the system power line should the solar cells fall deficient. If the line is above the 20V, then the comparator will change the frequency of the switches to control and transfer power from the solar cells. The FET switch and the 4 channel VNQ660SP are for frequency switching.

The local power regulation component shown in Figure 44 transforms the main bus voltage and converts it to a usable voltage for the subsystem devices.

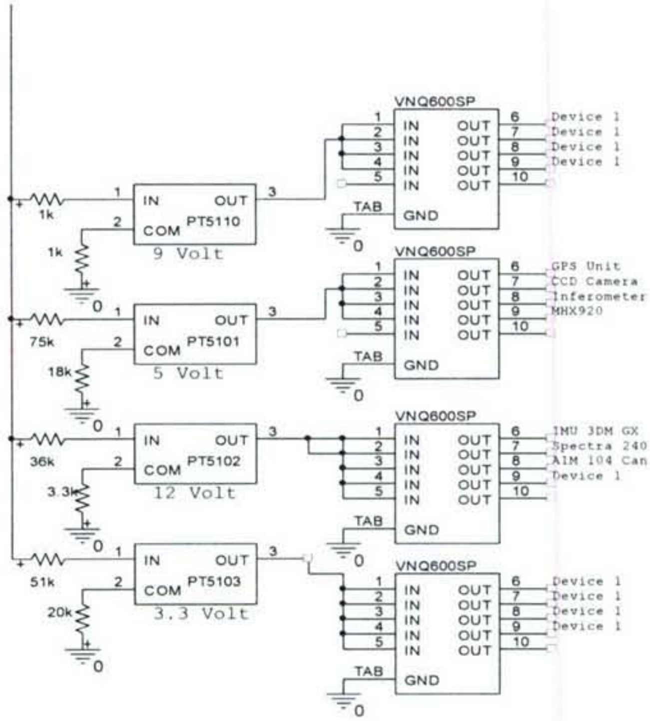


Figure 44: Local Power Regulation

Table 7: Mass and Cost Budgets

[redacted]		
[redacted]	[redacted]	[redacted]
[redacted]	0.2163	$475
[redacted]	25.00	$2000
[redacted]	6.0	$1,576
[redacted]	5.45	$3,500
[redacted]	[redacted]	[redacted]

Figure 45: Inventory Log: Power & Communications

[redacted]					
1	PT2103 A Voltage Regulator	Arrow Electronics, Inc.	3.15	1	3.15
3	TPS5103IDB DC/DC Controller	Arrow Electronics, Inc.	6.75	3	20.25
5	PT5102A Voltage Regulator	Arrow Electronics, Inc.	3.15	1	3.15
7	LM111JG Comparators	Arrow Electronics, Inc.	5.60	2	11.2
9	Kenwood TH-D7 Radios	Amateur Electronic Supply LLC	339.94	2	679.88
11	20 Pin Straight Female Header	Futurlec	0.15	10	1.5
13	VNQ660SP Quad Channel Relay	Digkey	6.56	10	65.6
15	PT5101A Voltage Regulator	Digkey	12.25	2	24.5
17	L63700 Current Regulation Switches	Digkey	10.35	6	62.1
19	VNQ810 Quad Channel Driver	Digkey	6.62	3	19.86
21	PT6103A Voltage Regulator	Digkey	12.25	3	36.75
23	Inductor 10 µH	Digkey	1.72	3	5.16
25	Capacitor 10 µF	Digkey	2.53	10	25.3
27	Ribbon Cable 20 ft.	Digkey	15.72	1	15.72
29	LED Holder	Radio Shack	1.30	1	1.30
31	Multipurpose PC Board 417 holes	Radio Shack	1.79	1	1.79
33	Set of four 2-position board terminals	Radio Shack	2.29	1	2.29
35	10µF 35V 20% Radial-lead electrolytic compacitor	Radio Shack	0.99	1	0.99
37	Three 30-ft. spools of solid, 22-gauge wire	Radio Shack	5.99	1	5.99
39	Standard mono phone plugs	Radio Shack	3.99	1	3.99
41	25-Position Male Solder D-Sub Connector	Radio Shack	1.89	2	3.78
43	15-Position HD male solder D-Sub connector	Radio Shack	1.89	1	1.89
45	Communications Components	Tedco Electronics	24.89	1	24.89
47	Adhesive CV-2289	NuSil	157.00	2	314

Telemetry Budgets

Table 8: Downlink Telemetry Budgets

[redacted]

Spacecraft Transmitter Power Output:	[redacted]	watts
In dBW:	4.8	dBW
In dBm:	34.8	dBm
Spacecraft Transmission Line Losses:	-1.0	dB
S/C Connector, Filter or In-Line Switch Losses:	0.0	dB
Spacecraft Antenna Gain:	[redacted]	dBiC
Spacecraft EIRP:	[redacted]	dBW
[redacted]		
Spacecraft Antenna Pointing Loss:	-1.0	dB
Antenna Polarization Loss:	-1.5	dB
Path Loss:	-135.6	dB
Atmospheric Loss:	-2.2	dB
Ionospheric Loss:	-0.2	dB
Rain Loss:	0.0	dB
Isotropic Signal Level at Ground Station:	[redacted]	dBW
[redacted]		
Ground Station Antenna Pointing Loss:	-2.0	dB
Ground Station Antenna Gain:	[redacted]	dBiC
Ground Station Transmission Line Losses:	-1	dB
Ground Station LNA Noise Temperature:	[redacted]	K
Ground Station Transmission Line Temp.:	290	K
Ground Station Sky Temperature:	450	K
G.S. Transmission Line Coefficient:	0.7943	
Ground Station Effective Noise Temperature:	542	K
Ground Station Figure of Merrit (G/T):	-14.8	dB/K
G.S. Signal-to-Noise Power Density (S/No):	[redacted]	dBHz
System Desired Data Rate:	[redacted]	bps
In dBHz:	39.8	dBHz
Telemetry System Eb/No:	[redacted]	dB
Telemetry System Required Bit Error Rate:	1.00E-06	
Telemetry System Required Eb/No:	18	dB
System Link Margin:	[redacted]	dB

Ground Station Antenna Pointing Loss:	-2.0	dB
Ground Station Antenna Gain:	13.5	dBiC
Ground Station Transmission Line Losses:	-1	dB
Ground Station LNA Noise Temperature:	125	K

Ground Station Transmission Line Temp.:	290	K
Ground Station Sky Temperature:	450	K
G.S. Transmission Line Coefficient:	0.7943	
Ground Station Effective Noise Temperature:	542	K
Ground Station Figure of Merrit (G/T):	-14.8	dB/K
Signal Power at Ground Station LNA Input:	[redacted]	dBW
Ground Station Receiver Bandwidth:	[redacted]	Hz
G.S. Receiver Noise Power (Pn = kTB)	[redacted]	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:	[redacted]	dB
System Desired Data Rate:	9600	bps
	39.8	dBHz
Telemetry System Required SNR:	18	dB
System Link Margin:	[redacted]	dB

Table 9: Uplink Telemetry Budget

[redacted]

Transmitter Power Output:		[redacted]	watts
	In dBW:	20.0	dBW
	In dBm:	50.0	dBm
Transmission Line Losses:		-3.0	dB
Connector, Filter or In-Line Switch Losses:		-1.0	dB
Antenna Gain:		13.5	dBiC
Ground Station EIRP:		[redacted]	dBW
[redacted]			
Ground Station Antenna Pointing Loss:		-1.0	dB
Antenna Polarization Losses:		-4.0	dB
Path Loss:		-145.3	dB
Atmospheric Losses:		-3.0	dB
Ionospheric Losses:		-1.0	dB
Rain Losses:		0.0	dB
Isotropic Signal Level at Ground Station:		[redacted]	dBW
[redacted]			
Spacecraft Antenna Pointing Loss:		0.0	dB
Spacecraft Antenna Gain:		[redacted]	dBiC
Spacecraft Transmission Line Losses:		-1.0	dB
Spacecraft LNA Noise Temperature:		[redacted]	K
Spacecraft Transmission Line Temp.:		270	K
Spacecraft Sky Temperature:		290	K
S/C Transmission Line Coefficient:		0.7943	
Spacecraft Effective Noise Temperature:		786	K

Spacecraft Figure of Merrit (G/T):	-28.0	dB/K
S/C Signal-to-Noise Power Density (S/No):	[redacted]	dBHz
System Desired Data Rate:	[redacted]	bps
In dBHz:	39.8	dBHz
Telemetry System Eb/No:	[redacted]	dB
Telemetry System Required Bit Error Rate:	1.00E-06	
Telemetry System Required Eb/No:	17.0	dB
System Link Margin:	[redacted]	dB
Spacecraft Antenna Pointing Loss:	[redacted]	dB
Spacecraft Antenna Gain:	[redacted]	dBiC
Spacecraft Transmission Line Losses:	[redacted]	dB
Spacecraft LNA Noise Temperature:	[redacted]	K
Spacecraft Transmission Line Temp.:	[redacted]	K
Spacecraft Sky Temperature:	[redacted]	K
S/C Transmission Line Coefficient:	[redacted]	
Spacecraft Effective Noise Temperature:	[redacted]	K
Spacecraft Figure of Merrit (G/T):	[redacted]	dB/K
Signal Power at Spacecraft LNA Input:	[redacted]	dBW
Spacecraft Receiver Bandwidth:	[redacted]	Hz
G.S. Receiver Noise Power (Pn = kTB)	[redacted]	dBW
Signal-to-Noise Power Ratio at S/C Rcvr:	[redacted]	dB
System Desired Data Rate:	[redacted]	bps
	[redacted]	dBHz
Telemetry System Required SNR:	[redacted]	dB
System Link Margin:	[redacted]	dB

8.2: Appendix B

Equation 4.3.1 $$V_{bus} := 1.25 \cdot N \cdot V$$

Equation 4.3.2 $$V_{array} := 1.20 V_{bus}$$

Equation 4.3.3 $$C_{chg} := N \cdot BC$$

Equation 4.3.4 $$P_{charge} := \frac{V_{chg} \cdot C_{chg}}{15 \cdot hr}$$

Equation 4.3.5 $$\eta_{temp} := 1 - \left(\frac{.005}{K}\right)(302 - MOT)$$

Equation 4.3.6 $$\eta_{rad} := 1 - \eta_{EOL}$$

Equation 4.3.7 $$P_{BOL} := \frac{P_{EOL}}{\eta_{rad} \cdot \eta_{temp} \cdot \eta_{angle}}$$

Equation 4.3.8 $$A_{cell} := \frac{P_{BOL}}{\eta_{BOL} \cdot SI}$$

Equation 4.3.9 $$A_{array} := \frac{A_{cell}}{PF}$$

Equation 4.3.10 $$N_{cells} := \frac{A_{cell}}{A_{cellactual}}$$

Equation 4.0.1 $$I_{SP} \approx \left(\frac{E}{A}\right)^{0.62}$$

Equation 4.0.2 Ablation rate: $\dfrac{\Delta m}{A}$

Equation 4.0.3 Mass Burn per Pulse: $\dfrac{\Delta m}{pulse}$

Equation 4.0.4 Energy per Pulse: $\dfrac{E}{Pulse}$

Equation 4.0.5 Magnetic Moment Vector: $\vec{M} = iNA\hat{n}$

Equation 4.0.6 Magnetic Flux Density:

$$B = \frac{\mu_o}{4\pi} M \left[\frac{\frac{R}{L} - \frac{1}{2}}{\left(R^2 - RL + \frac{L^2}{4}\right)^{3/2}} - \frac{\frac{R}{L} + \frac{1}{2}}{\left(R^2 + RL + \frac{L^2}{4}\right)^{3/2}} \right]$$

Equation 4.0.7 Magnetic Torque: $\vec{T}_M = \vec{M} \times \vec{B}$

Equation 4.0.8 Gravity Gradient Torque:

$$L_g = \frac{3 \cdot G \cdot M_e}{R_c^{\,5}} \cdot I \cdot R_c$$

Equation 4.0.9 Quaternion Equation

$$R := \begin{bmatrix} (q_1)^2 - (q_2)^2 - (q_3)^2 - (q_4)^2 & 2(q_1 q_2 + q_4 q_3) & 2(q_1 q_3 + q_4 q_2) \\ 2(q_2 q_1 + q_4 q_3) & (q_1)^2 - (q_2)^2 - (q_3)^2 - (q_4)^2 & 2(q_2 q_3 + q_4 q_1) \\ 2(q_3 q_1 + q_4 q_2) & 2(q_3 q_2 + q_4 q_1) & (q_1)^2 - (q_2)^2 - (q_3)^2 - (q_4)^2 \end{bmatrix}$$

Table 10: ADCS Mass and Power Budgets

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	[illegible]			[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]		[illegible]				[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	

8.3 Appendix C: Knightsat Requirements Document

KnightSat Requirements Document

Revision History

Date	Name	Version	Change Reference
13-Apr-07	Randy Allen	1.0	First Draft
22-May-07	Rebecca Kendall	2.0	Payload / Structures Revision
1-Aug-07	Rebecca Kendall	2.1	Draft (Entire) Revision

Document Reviewers

Date	Title	Name
	Payload lead	Rebecca Kendall
	ADCS lead	Daniel Stuhr
	Structures lead	Danielle Grant
	Power/Comm lead	Timothy Young

Document Approval

Date	Title	Name
	Project Director	Dr. Roger Johnson
	Propulsion Advisor	Dr. J. Brandenburg
	Mission Advisor	Dr. R. Crabbs
	Mission Advisor	D. Hand

Table of Contents

1 Team – University of Central Florida

1.1 Professors

1.1.1 Dr. R. Johnson – Project Director
1.1.2 Dr. J. Brandenburg – Propulsion Advisor
1.1.3 Dr. R. Crabbs – Mission Advisor
1.1.4 Dr. D. Hand – Mission Advisor

1.2 Students

1.2.1 Rebecca Kendall – Student Leader, Payload / Optics

1.2.2 Danielle Grant – Team Lead, Structures / Payload
1.2.2.1 Lashanda Oliver
1.2.2.2 Robert Shanami
1.2.2.3 Chad Moody
1.2.2.4 Rebecca Kendall

1.2.3 Bhargov Gajjar – Gravity Gradient Boom Design

1.2.4 Daniel Stuhr – Team Lead, Attitude Determination and Control System
1.2.4.1 Jason Dunn
1.2.4.2 Ruben Salles
1.2.4.3 Jessica Vega
1.2.4.4 Keegan Ford
1.2.4.5 Mohammed Elzooghby

1.2.5 Randal Allen – Team Lead, Mechatronics
1.2.5.1 Ed Daughtery
1.2.5.2 Luis Readdy
1.2.5.3 Emily Afifi

1.2.6 Tim Young – Team Lead, Power / Communications
1.2.6.1 Arthur Hayes
1.2.6.2 Max Adsit
1.2.6.3 Joe Wiley
1.2.6.4 Jerome Dilworth

2 Mission Needs

2.1 Operations Concept

2.1.1 Primary objective

2.1.1.1 Provide two images of a point on the earth at better than 10 meters per pixel resolution of an identifiable area over land.

2.1.2 Secondary objectives

2.1.2.1 Demonstrate the University of Central Florida's gravity gradient.

2.1.2.2 Utilize pulse plasma thrusters to de-orbit the spacecraft.

2.2 Spacecraft Life and Reliability

2.2.1 Mission duration

2.2.1.1 Official Mission Duration will be determined by Air Force; however Planning is based on a six month time period.

2.2.2 Success criteria

2.2.2.1 Success criteria is based upon the requirements of the launch provider being met, and the requirements set by the Knightsat Team be met, i.e. primary mission be successful as well as all subsystems correlate properly.

2.3 Communication Architecture

2.3.1 See section 13

2.4 Programmatic Constraints

2.4.1 See section 3

The requirements for the Nanosat-4 as stated by the Air Force and the supervising director are to first meet all program deadlines and for each student to participate in all aspects of the design and implementation process. Furthermore, the design team is encouraged to meet the requirements of the launch provider, which include the system meeting the strength and material requirements; the system utilizing the materials and fasteners recommended; the system being built considering assembly and testing requirements; and the avoidance of all Phillips or flat-head fasteners in the system. The system must not contain sealed or pressured containers and must be launched in a "dead" (no battery charge) state. The system is also required to use NiCad (Nickel Cadmium) batteries and should make use of machined metal primary structures.

3 System Drivers and Associated Risks

3.1 Performance

3.1.1 Performance is the middle constraint

3.1.2 Associated performance risks

3.1.2.1 Risk 1: Structural Stability (withstanding 20 G's)

3.1.2.2 Risk 2: Payload Optics Withstanding Launch Environment

3.1.2.3 Risk 3: ADCS Accuracy

3.2 Cost

3.2.1 Cost is the driver constraint

3.2.2 Associated cost risks

3.2.2.1 Risk 1: Payload Costs; Optics

3.2.2.2 Risk 2: Accidental loss of Electronic Devices

3.2.2.3 Risk 3: Testing Facilities

3.3 Schedule

3.3.1 Schedule is the weak constraint

3.3.2 Associated schedule risks

3.3.2.1 Risk 1:Not Meeting AFRL Deadlines

3.3.2.2 Risk 2: Not Meeting UCF Deadlines

3.4 Other constraints

3.4.1 Air Force Research Laboratory
3.4.2 Maintaining Student Involvement

4 Functional Requirements

4.1 Production

4.1.1 Requirements

4.1.1.1 Operational Mission Requirements (User / System Engineering)

4.1.1.2 System Specification (System Engineering)

4.1.1.3 System Standard (System Engineering)

4.1.1.4 Transition Plan (System Engineering)

4.1.1.5 External System Interfaces (System Engineering)

4.1.1.6 Integrated System Test Plan (System Engineering)

4.1.1.7 Segment Specifications (System Engineering / Segment System Engineering)

4.1.1.8 Inter-segment Interface Requirements, Control documents (System Engineering / Segment System Engineering)

4.1.1.9 Facility Requirements (System Engineering / Segment System Engineering)

4.1.2 Design

4.1.2.1 Segment Design Specs (Segment System Engineering / Subsystem Engineering)

4.1.2.2 Subsystem Design Specs (Segment System Engineering / Subsystem Engineering)

4.1.2.3 Intra-segment Interface Control Documents (Segment System Engineering / Subsystem Engineering)

4.1.2.4 Segment Test Plans (Segment System Engineering / Subsystem Engineering)

4.1.2.5 Development Plans (Segment System Engineering / Subsystem Engineering)

4.1.2.6 Other – e.g. Database Specs (Segment System Engineering / Subsystem Engineering)

4.1.2.7 Component Specs (Subsystem Engineering / Component Engineering)

4.1.2.8 As-built Specs (Subsystem Engineering / Component Engineering)

4.1.2.9 Subsystem & Component-Level Test Plans (Subsystem Engineering / Component Engineering)

4.2 Integration and Test

4.2.1 Integration and test plan

4.3 Launch and Deployment

4.3.1 Detach from launch vehicle

4.3.2 Enable power management computer

4.3.3 Enable thermal management system

4.3.4 Power ADCS and payload computers

4.3.5 Begin momentum dumping with torque rods

4.3.6 Gravity gradient boom deployment

4.3.7 Signal ground station that Knightsat is operational

4.4 Conduct Mission Operations

4.4.1 Receive imaging coordinates from ground station

4.4.2 ADCS positions spacecraft

4.4.3 Spacecraft obtains ground image

4.4.4 Spacecraft transmits image to ground station

4.4.5 Payload in stand-by mode with subsystem health monitoring

4.4.6 Eventual de-orbit burn

4.5 Conduct Training

4.5.1 Develop courseware

4.5.2 Provide classroom training

4.5.3 Provide simulator training

4.5.4 Provide on-the-job training

4.5.5 Certify team members

4.5.6 Provide continuation training

4.6 Conduct Contingency Operations

4.6.1 Contingency plan

To be reviewed...

5 Payload Requirements

5.1 Physical Parameters

5.1.1 Parameters: 12.5 inches long; 4.255 inches in diameter

5.1.2 Mass: 5 kg

5.1.3 Shape: Cylindrical

5.1.4 Power Required: 0.75 watts

5.2 Operations

5.2.1 Six minute operating time

5.2.2 Data rates

5.2.3 Field of view (FOV): @ 1000 yards— 52.5 feet

5.3 Pointing

5.3.1 Reference: Earth

5.3.2 Accuracy: within 1 degree

5.3.3 Stability: reliant upon ADCS

5.4 Slewing

5.4.1 Magnitude: N/A

5.4.2 Frequency: N/A

5.5 Environment

5.5.1 Maximum and minimum temperatures: Maximum: 325K

Refer to consolidated report within Structures

6 Coverage, Support, and Survival Requirements

6.1 Coverage

6.1.1 Orbit parameters

6.1.1.1 Altitude: Between 300 and 700 km (LEO)

6.1.1.2 Inclination: TBA by AFRL

6.1.1.3 Eccentricity: TBA by AFRL

6.1.2 Eclipses: TBD by AFRL

6.1.2.1 Maximum duration

6.1.2.2 Frequency: 90 minutes

6.1.3 Lighting conditions

6.1.3.1 Sun angle and viewing conditions: Depending on Orbit

6.1.4 Maneuvers

6.1.4.1 Size: N/A

6.1.4.2 Frequency: N/A

6.2 Engineering Support

6.2.1 Design and Test Failure Workarounds

6.2.2 Plan orbit maneuvers: ADCS: Momentum dump utilizing PPTs and Torque Rods

6.2.3 Manage subsystems: Respective team leaders

6.2.4 Manage payload: Chad Moody

6.2.5 Manage consumables: Individual per team, and submitting to Knightsat Team leader

6.2.6 Resolve anomalies: Performed by respective teams

6.2.7 Maintain flight software: ADCS: Computers

6.2.8 Maintain simulator: N/A

6.2.9 Maintain ground-system database: Power/Communications at SpaceHab, Titusville, FL

6.2.10 Analyze spacecraft trends: Dr R. Johnson, R. Crabbs, and team

6.3 Survival Requirements

While the satellite is not subjected to as hostile environment, per se, various components will be hardened for radiation shielding.

6.3.1 Radiation dosage

6.3.1.1 Average: Electron energies < 5 MeV

6.3.1.2 Peak: Electron and proton fluxes peak at $1.5R_E$ *to* $2.0R_E$

6.3.2 Particles and meteoroids

6.3.2.1 Size: Relatively small: < 30 cm

6.3.2.2 Density: Dependent upon the individual

6.3.3 Space debris

6.3.3.1 Density Dependent: upon the individual

6.3.3.2 Probability of impact: small particles: Definate

7 Spacecraft Bus Requirements

7.1 Propulsion Subsystem

The propulsion subsystem provides thrust to adjust orbit and attitude, and to manage angular momentum.

7.1.1 Number of pulse plasma thrusters: 8

7.1.2 Orientation of pulse plasma thrusters: two for each axis (x, y), and 4 (one on top of the bus, one on the bottom of the bus) on the z-axis

7.1.3 Location of pulse plasma thrusters: refer to 7.1.2

7.1.4 Thrust level requirement: 1mN of Force

7.1.5 Impulse requirement: 1000 volt

7.1.6 Heaters? Power requirement: Two requirements: one 1000 volt capacitor and one 1-2 watt power source (per PPT)

7.2 Guidance, Navigation, and Control Subsystem

The guidance, navigation, and control subsystem provides determination and control of orbit and attitude, plus pointing of spacecraft and appendages. Attitude control requirements are based on those for payload pointing and the spacecraft bus pointing.

7.2.1 Pointing requirements

The payload requires earth pointing. Therefore, the two horizontal axes can be used to control the spacecraft's attitude. Since the payload is fixed to the spacecraft's body, the two axes are two of the three body axes. Thus we can use the third axis – rotation about the earth pointing axis – to satisfy the second sun pointing requirement.

7.2.2 Three-axis control requirement

Because of the multiple pointing requirements, three-axis control is implied.

7.2.3 Control requirement during thrusting

Three-axis control requires attitude to be sensed with sensors whose output is used to control torquers. Torquers include thrusters operated off-on to control thrust direction.

7.2.4 Control requirement while not thrusting

Three-axis control is achieved using sensors and torquers.

7.2.5 Coarse control: 1° requirement

Coarse control implies passive control using a gravity gradient.

7.2.6 Low (<1 kW) power requirement

The low power requirement allows for a cylindrical array.

7.2.7 Guidance and Navigation

The guidance and navigation function will use ground tracking to measure the flight path, ground computing of desired velocity corrections, and command of the correction through the communications and command subsystems. The direction of the velocity correction is governed by the attitude control of the spacecraft body and the magnitude is controlled by the engine firing time.

7.2.8 Attitude Control
Attitude control depends on the attitude sensors.

7.2.8.1 Gyroscope
Normal use involves periodically resetting the reference position. Sensor range $\pm 300°/s$

7.2.8.2 Magnetometer
The attitude is measured relative to the earth's local magnetic field. Magnetic field uncertainties and variability dominate accuracy. Sensor range ± 1.2Ga

7.2.8.3 Accelerometer
Sensor range ± 5g

7.2.9 Torquing
Torquing methods for three-axis controlled spacecraft include gravity gradient.

7.2.9.1 Gravity Gradient
Gravity gradient and magnetic torquing are clean and simple but do not

7.3 Communications Subsystem
The communications subsystem receives and demodulates uplink signals and modulates and transmits downlink signals. The subsystem also allows us to track spacecraft by retransmitting received range tones or by providing coherence between received and transmitted signals, so we can measure Doppler shift.

7.3.1 Data rate requirement
The data rate of 9600 bit/s sets the communications subsystem's bandwidth at 19.2dB which establishes the received power required to detect signals.

7.3.2 Frequency requirement
The frequency will be S-band (2GHz), X-band (8GHz), or Ku-band (12GHz).

7.3.3 RF power budget
RF power out is 6W for VHF, 5.5W for UHF, and the sensitivity is less than $0.18\mu V$.

7.3.4 Equipment
The communications equipment consists of a dipole VHF antenna, a directional UHF antenna, and receivers and transmitters based on Kenwood's model TH-7.

7.4 Command and Data Handling Subsystem
The command and data handling subsystem receives and distributes commands and collects, formats, and delivers telemetry for standard spacecraft operations (housekeeping) and payload operations.

7.4.1 Command list
The command list is a complete list of commands for the payload and each spacecraft bus subsystem, including commands for each redundancy option and each commandable operation.

7.4.2 Telemetry list

The telemetry list involves analyzing spacecraft operation to select telemetry measurement points that completely characterize it, including signals to identify redundancy configuration and command receipt.

7.4.3 Timing

Timing involves analyzing spacecraft operation to identify time-critical operations and timeliness needed for telemetry data.

7.4.4 Data rates

Data rates are selected that support command and telemetry requirements and time-critical operations.

7.4.5 Processing requirement

Identifying processing requirements involves examining the need for encryption, decryption, sequencing, and processing of commands and telemetry.

7.4.6 Storage requirement

The computer is required to hold 256 Mb of Images. The downlink of the payload images are dependant on the separate ground station availability.

7.4.7 Equipment

Select components to meet the requirements.

7.5 Thermal Subsystem

The thermal design of a spacecraft involves identifying the sources of heat and designing paths for transporting and rejecting heat, so components will stay within required temperatures.

7.5.1 Heat sources

Heat sources include internal dissipation, such as electrical components, and external radiation such as solar radiation, earth-reflection, and infrared radiation. The Heat sources for Knightsat include the following: the CCD camera, PPT's, batteries, and thermistors.

7.5.2 Location of radiating panels

Radiating panels usually face away from the sun and earth. The radiating panels on the Knightsat will be constantly changing, based on the ADCS and the payload requirements.

7.6 Power Subsystem

The power subsystem generates power, conditions and regulates it, stores it for periods of peak demand or eclipse operation, and distributes it to end users. The subsystem may also need to convert ands regulate voltage levels or supply multiple voltage levels. It frequently switches equipment on or off and, for increased reliability, protects against short circuits and isolates faults. Power subsystem design is also influenced by space radiation, which degrades the performance of solar cells. Finally, battery life often limits the spacecraft's lifetime.

7.6.1 Power subsystem sizing

7.6.1.1 ADCS: 2.2332 W Hrs; Payload/Structures: 0.31 W Hrs; Power/Comm: 97.9645 W Hrs

7.6.1.2 Storage requirement: N/A

7.6.2 Solar arrays, batteries, and components

7.6.2.1 Sizing solar arrays

The required area of a planar solar array is related to the required power, P, the solar constant (1351 W/m^2), and the conversion efficiency of the solar-cell system. Although cells have had efficiencies as high as 30%, practical array designs range from 5% to 15% when taking into operating conditions and degradation at end of life. An array with an efficiency of 7% would have a required area of:

$$A_a = \frac{P}{0.07 \times 1351} \approx 0.01P$$

Note: A cylindrical array will be used. Because the projected area of a cylinder is $1/\pi$ times the total area, the cylindrical array should have approximately π times as many cells as a planar array with the same power rating. Temperature effects slightly favor the cylindrical array, so the actual ratio is closer to 1/2.5.

7.6.2.2 Sizing batteries

Battery sizing is determined from the energy it must produce (discharge power times discharge duration) and from its depth of discharge. For nickel-cadmium (NiCd) with a cycle life less than 1000 cycles, the depth of discharge is 80%; with a cycle life of 10,000 cycles, the depth of discharge is 30%.

To compute the battery's capacity, divide discharge energy (W•hr) by the depth of discharge. The ratio of battery weight to battery capacity is 30W•hr/kg for NiCd batteries.

Temperature range, rate of charge, rate of discharge, degree of overcharge are still to be determined

7.6.2.3 Controlling charging and distributing / converting power

The spacecraft's primary power produced by the solar arrays and batteries is not well regulated. Furthermore, we must match the solar array's electrical output to the battery's charging requirements and provide switching equipment that allows the battery to supply power when needed.

Limiting the battery's charging rate, limiting overcharge, providing for low-impedance discharge, providing for reconditioning are all still to be determined.

Components for switching and fault isolation are still to be determined.

7.7 Structures and Mechanisms

The spacecraft structure carries and protects the spacecraft and payload equipment through the launch environment and deploys the spacecraft after orbit injection. The load-carrying structure of a spacecraft is the *primary structure*, whereas brackets, closeout panels and most deployable components are *secondary structure*.

7.7.1 Primary structure

The size of the primary structure is based on launch loads, with strength and stiffness dominating the design. The structure is fabricated from 7075-T6 anodized aluminum.

7.7.1.1 Load factor

The maximum acceleration in specified direction - $N_x = \pm 20.0$, $N_y = \pm 20.0$, $N_z = \pm 20.0$

7.7.1.2 Limit load

The maximum expected acceleration: 20 g's

7.7.1.3 Yield load

The load at which the structural member suffers permanent deformation: N/A

7.7.1.4 Ultimate load

The load at which the structural member fails: TBD

7.7.1.5 Safety factor

2.0 For yielding strength when 20g's is applied to each axis independently.

7.7.1.6 Safety margin

(ultimate load / limit load) - 1: TBD

7.7.1.7 Yield factor

yield load / limit load: TBD

7.7.1.8 Yield margin

(yield load / limit load) - 1: TBD

7.7.1.9 Uncertainty factor

design load / limit load: TBD

7.7.2 Secondary structure (based on orbit factors)

The secondary structure only has to survive but not function during boost, and we can usually cage and protect deployables throughout this phase

7.7.3 Mass budget

The requirement is for the mass to be less than 30kg.

8 Deployment and Replenishment Requirements

8.1 Deployment

8.1.1 Detach from launch vehicle

8.1.2 Enable power management computer

8.1.3 Enable thermal management system

8.1.4 Power ADCS and payload computers

8.1.5 Begin momentum dumping with torque rods

8.1.6 Gravity gradient boom deployment

8.1.7 Signal ground station that Knight Sat is operational

8.2 Replenishment Requirements

8.2.1 Not applicable

Also needs to be reviewed...

9 System Modes / Failure Modes and Effects

9.1 System Modes

9.1.1 Command the spacecraft

9.1.2 Monitor subsystems

9.1.3 Manage payloads

9.1.4 Manage recorders

9.1.5 Recover payload data

9.1.6 Resolve anomalies

9.1.7 Analyze spacecraft trends

9.2 Failure Modes and effects

9.2.1 Design (refer to consolidated report for further details)

9.2.1.1 Propulsion: PPTs could mis-fire, capacitor could be damaged during launch

9.2.1.2 Guidance, navigation, and control: radios not transmitting, ground station antenna not receiving signal

9.2.1.3 Communication: refer to 9.2.1.2

9.2.1.4 Command and data handling: Gumstix computer damaged during the launch environment

9.2.1.5 Thermal: thermistors not respond to commandsl

9.2.1.6 Power: batteries/solar panels not reponding

9.2.1.7 Structures and mechanisms

9.2.2 Environment

9.2.2.1 Radiation, etc.

9.2.3 Operations

9.2.3.1 Improper commands, etc.

10 Spacecraft Ground Control Requirements

10.1 Ground System Design Process

10.1.1 Establish number and location of ground stations: one (1)

10.1.2 Establish space-to-ground data rates: Uplink: 100 watts; Downlink: 3.0 watts; refer to the consolidated report in Appendix A

10.1.3 Determine required Gain-to-noise Temperature ratio (G/T): 75.9 dBHz

10.1.4 Determine required Effective Isotropic Radiated Power (EIRP): TBD

10.1.5 Determine required data handling: 39.8 dBHz

10.1.6 Establish data handling location: SpaceHab: Titusville, FL

10.1.7 Decide location of Spacecraft Operations Control Center (SOCC): SpaceHab: Titusville, FL

10.1.8 Decide location of Payload Operations Control Center (POCC): SpaceHab: Titusville, FL

10.1.9 Decide location of Mission Control Center (MCC): SpaceHab: Titusville, FL

10.1.10 Determine and select communication links: TBD

10.1.11 Evaluate complete or partial use of service-provided ground systems: SpaceHab: Titusville, FL

11 Data Processing and Distribution Requirements

11.1 Computer Systems development Process

11.1.1 Define requirements

11.1.1.1 Evaluate mission objectives: Requires transmitting images from the CCD Camera to the radios on board the Knightsat.

11.1.1.2 Perform functional partitioning: TBD

11.1.2 Develop system baseline

11.1.2.1 Evaluate candidate architectures: Refer to the Knightsat Consolidated report: beginning on Figure 30

11.1.2.2

11.1.2.3 Perform functional flow analysis: Refer to the Knightsat Consolidated report: beginning on Figure 30

11.1.2.4

11.1.2.5 Establish block diagram for system: Refer to the Knightsat Consolidated report: beginning on Figure 30

11.1.3 Expand baseline concepts

11.1.3.1 Evaluate and select hardware instruction set architecture: TBD

11.1.3.2 Evaluate and select software language: TBD

11.1.3.3 Define processing tasks: TBD

11.1.3.4 Establish computer size and throughput estimate: TBD s

11.1.4 Evaluate system effectiveness

11.1.4.1 Verify requirements traceability: TBD

11.1.4.2 Evaluate baseline against design drivers: TBD

11.1.4.3 Assess development issues for system baseline: TBD

11.1.4.4 Evaluate system testability: TBD

12 System Interfaces

12.1 Ground-System Interface

12.1.1 Degree of autonomy

12.1.1.1 Required autonomous operations: TBD

12.1.2 Ground stations

12.1.2.1 Number: one (1)

12.1.2.2 Locations: SpaceHab, Titusville, FL

12.1.2.3 Performance: Receive Images, and health monitoring links, Image Processing, Stereo image development

12.1.3 Space links

12.1.3.1 Space-to-space links

12.1.3.2 Performance

12.2 Other Interfaces...

13 Communications Architecture Requirements

13.1 Specifying a Communications Architecture

13.1.1 Identify communication links

13.1.1.1 Define mission objectives: To deliver adequate communication uplinks and downlinks between the Knightsat and the Ground station.

13.1.1.2 Define mission requirements: To transmit links between the ground station including the images obtained by the Knightsat payload.

13.1.1.3 Determine the architecture:(TBD)

13.1.2 Determine data rates for each link

13.1.2.1 Specify frequency band:(TBD)

13.1.2.2 Determine sampling rates, quantization levels: 9600 bps (desired)

13.1.3 Design each link

13.1.3.1 Select frequency band(TBD)

13.1.3.2 Select modulation, coding: (TBD)

13.1.3.3 Determine antenna size, beamwidth constraints(Refer to consolidated report)

13.1.3.4 Determine transmitter power constraints: 100 watts

13.1.3.5 Estimate atmospheric, rain absorption:-3dB

13.1.3.6 Estimate received noise, interference powers: 37.0 dB

13.1.3.7 Calculate required antenna gains and transmitter power: (TBD)

13.1.4 Size the payload

13.1.4.1 Select payload antenna configuration: N/A

13.1.4.2 Calculate antenna size: N/A

13.1.4.3 Calculate antenna mass: N/A

13.1.4.4 Estimate transmitter mass: N/A