# LEONIDAS SATELLITE CONCEPT STUDY REPORT

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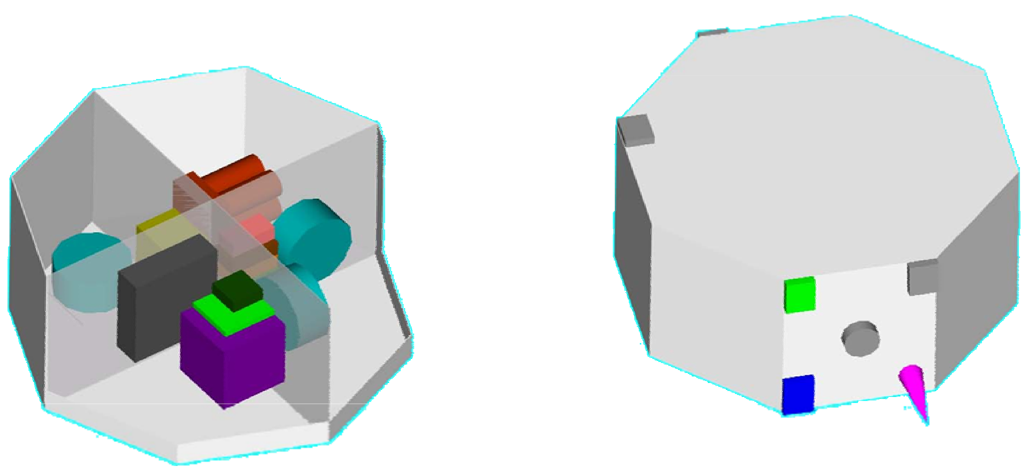

## ABSTRACT

Microsatellites provide a cheap and efficient way to perform experiments, collect data and provide space validation for science technology developments. Hawaii has the capability to complete an entire low Earth orbiting satellite mission which gives it a tremendous advantage in microsatellite market. Our satellite concept report is an outline of how we would construct a satellite with "plug and play" capabilities to allow easy integration of various payloads. We have listed the various components for each subsystem and specified how each will interact with one another. The concept report will serve as a foundation for writing future proposals.

## BACKGROUND

Extreme high cost of manufacturing and launching large satellites has increased need for micro and nanosatellites to take this role. A typical large satellite can have a mass over 10,000 kg when you include the fuel and cost over 300 million dollars to manufacture and launch. However, microsatellites way between 10 and 100 kg and meet the high performance demands but cost approximately 1% of what it takes to build a large satellite. The State of Hawaii is unique in the fact that it is one of the few states with the capability to complete an entire low Earth orbiting satellite mission. Along with an abundance of experience in the engineering and science disciplines necessary to complete Phase A proposals and mission designs, Hawaii also has the facilities to manufacture, integrate and test space flight hardware. Through the Pacific Missile Research Facility (PMRF) on the island of Kauai, Hawaii has the capability to launch small satellites into polar orbit. Also, unlike land locked states, Hawaii does not have the nominal risks of failing debris from stages one and two. Finally, Hawaii has antennas capable of receiving and sending commands and controls to satellites in orbit. All these capabilities give Hawaii a tremendous advantage in the microsatellite market.

In 2002 Dr. Luke Flynn, director of the Hawaii Space Grant Consortium (HSGC), Dr. Wayne Shiroma and Dr. Carlos Combria, professors of the college of Engineering, founded the Low Earth Orbit NanoSatellite Integrated Distributed Alert System (LEONIDAS). LEONIDAS originated as an extension of the CubeSat program created by Dr. Shiroma, in response to the Department of Defense's (DOD) increased interest in developing smaller satellites. LEONIDAS eventually evolved to focus on developing satellites for rapid deployment and repetitive surveillance of a given area. Since the development of LEONIDAS, the University of Hawaii Engineering Department has constructed three student-developed cubesats, one of which is waiting in Russia to be integrated and launched with other universities' cubesats. University of Hawaii gained further experience in the space technology field by funding two engineering students from HSGC, to participate in the Magnetic Field Investigation of Mars by Integrating Consortia (MIMIC) mission proposal hosted at Jet Propulsion Laboratory's (JPL).

Recently Professor Lloyd French, a System Architect for LEONIDAS, sought to create a team of undergraduate students to design and launch a microsatellite. The team is comprised of students from the Department of Engineering and Geophysics & Planetary Sciences that includes Native Hawaiian and local engineering students. The microsatellite will demonstrate Hawaii's capability to complete an entire spacecraft mission and serve as a testbed for experiments and science technology developments.

## OBJECTIVE

The LEONIDAS Mission Concept Study Team (LMCST) objectives are to research and understand the basic concepts for spacecraft design. We will research the various satellite components, payloads, past successful launched buses and how the subsystems will integrate and function with one another. The concept study will be colossal in advancing our education as engineering students through the knowledge and experience we will gain while working on the LMCST. Our report will demonstrate Hawaii's capability to complete a successful spacecraft mission, and from our concept study we will develop future proposals.

## GOALS

Our primary goal is to submit a proposal to be entered into the Air Force's nanosatellite competition. Given funding we wish to start developing the prototype for our satellite. Ultimately, we want to build, launch and operate a fully functional satellite carrying GPS experiment, an antenna experiment, software experiment and an imager for global reconnaissance. A successful mission would place the University of Hawaii's College Engineering program amongst the elite engineering programs in the nation. This would have major implications in enhancing the technology and research at the university. It would also be beneficial in the recruitment of finest professors and the best and brightest students of Hawaii. The mission's success would be a major stepping stone for improving the economic development of Hawaii.

## SYSTEMS ENGINEERING

The baseline design for the LMCST satellite concept was intended to conduct four experiments. The first experiment will conduct remote sensing using UV imaging. The last two experiments will be test flown to demonstrate their effectiveness for future missions. They are an Active Antenna by Dr. Wayne Shiroma's CubeSat team and a GPS unit provided by SSTL. A fourth experiment involves testing a spacecraft housekeeping software provided by JPL. These experiments will operate in a sun synchronous orbit with a period of 96 minutes at about 350 km. Our satellite will orbit the earth approximately 15 times a day in which only three will be adequate to perform experiments, operate the imager or transfer data.

There are eight different modes for the spacecraft: Camera Experiment, Active Antenna Experiment, SSTL Experiment, JPL Experiment, Communications, Maneuver, Power Save, and Safe Mode. During the Camera Mode the spacecraft will image a specified area. In the Active Antenna Experiment Mode we will attempt to communicate to the ground station by running Active Antenna and the UHF unit to demonstrate its functionality. The SSTL GPS unit will run during SSTL Experiment Mode were it will attain and collect knowledge on our spacecraft's polar orbit. The JPL Experiment Mode will take place after primary mission is complete is a software that monitors the health of the spacecraft. The spacecraft's Communications Mode utilizes the S-band frequency to transmit and receive data to and from the ground station. During the Maneuver Mode of the spacecraft uses the ACS system to adjust its orientation. While in darkness, the spacecraft will conserve energy by entering a Power Save Mode where all systems, except for C&DH will function in standby. In an emergency, the Safe Mode will be utilized where by the spacecraft shuts down all systems excluding the secondary communications and the C&DH systems.

By incorporating these modes into an orbital period we were able to construct the sample operations cycle in Table 1. There will be four different operation cycles for our mission. The operations cycle is used to create a power profile for the spacecraft, which is used to size the power system. The power profile for communications and payloads is shown Figure 1 & 2.

Operation Cycles

| Time (min) | Mode | Power (W) | Time (min) | Mode | Power (W) |
|---|---|---|---|---|---|
| 0 | SSTL Experiment | 42.36 | 0 | Active Experiment | 44.48 |
| 6 | SSTL Experiment | 42.36 | 6 | Active Experiment | 44.48 |
| 12 | Power Save | 14.8 | 12 | Power Save | 14.8 |
| 18 | Power Save | 14.8 | 18 | Power Save | 14.8 |
| 24 | Power Save | 14.8 | 24 | Power Save | 14.8 |
| 30 | Power Save | 14.8 | 30 | Power Save | 14.8 |
| 36 | Power Save | 14.8 | 36 | Power Save | 14.8 |
| 42 | Power Save | 14.8 | 42 | Power Save | 14.8 |
| 48 | Power Save | 14.8 | 48 | Power Save | 14.8 |
| 54 | Power Save | 14.8 | 54 | Power Save | 14.8 |
| 60 | Power Save | 14.8 | 60 | Power Save | 14.8 |
| 66 | Power Save | 14.8 | 66 | Power Save | 14.8 |
| 72 | Power Save | 14.8 | 72 | Power Save | 14.8 |
| 78 | Power Save | 14.8 | 78 | Power Save | 14.8 |
| 84 | Maneuver | 41.36 | 84 | Maneuver | 41.36 |
| 90 | SSTL Experiment | 42.36 | 90 | Active Experiment | 44.48 |
| 96 | SSTL Experiment | 42.36 | 96 | Active Experiment | 44.48 |
| Time (min) | Mode | Power (W) | Time (min) | Mode | Power (W) |
| 0 | Communications | 50.36 | 0 | Camera Experiment | 47.36 |
| 6 | Communications | 50.36 | 6 | Camera Experiment | 47.36 |
| 12 | Power Save | 14.8 | 12 | Power Save | 14.8 |
| 18 | Power Save | 14.8 | 18 | Power Save | 14.8 |
| 24 | Power Save | 14.8 | 24 | Power Save | 14.8 |
| 30 | Power Save | 14.8 | 30 | Power Save | 14.8 |
| 36 | Power Save | 14.8 | 36 | Power Save | 14.8 |
| 42 | Power Save | 14.8 | 42 | Power Save | 14.8 |
| 48 | Power Save | 14.8 | 48 | Power Save | 14.8 |
| 54 | Power Save | 14.8 | 54 | Power Save | 14.8 |
| 60 | Power Save | 14.8 | 60 | Power Save | 14.8 |
| 66 | Power Save | 14.8 | 66 | Power Save | 14.8 |
| 72 | Power Save | 14.8 | 72 | Power Save | 14.8 |
| 78 | Power Save | 14.8 | 78 | Power Save | 14.8 |
| 84 | Maneuver | 41.36 | 84 | Maneuver | 41.36 |
| 90 | Communications | 50.36 | 90 | Camera Experiment | 47.36 |
| 96 | Communications | 50.36 | 96 | Camera Experiment | 47.36 |

Table 1: Operation cycles of satellite while in orbit.

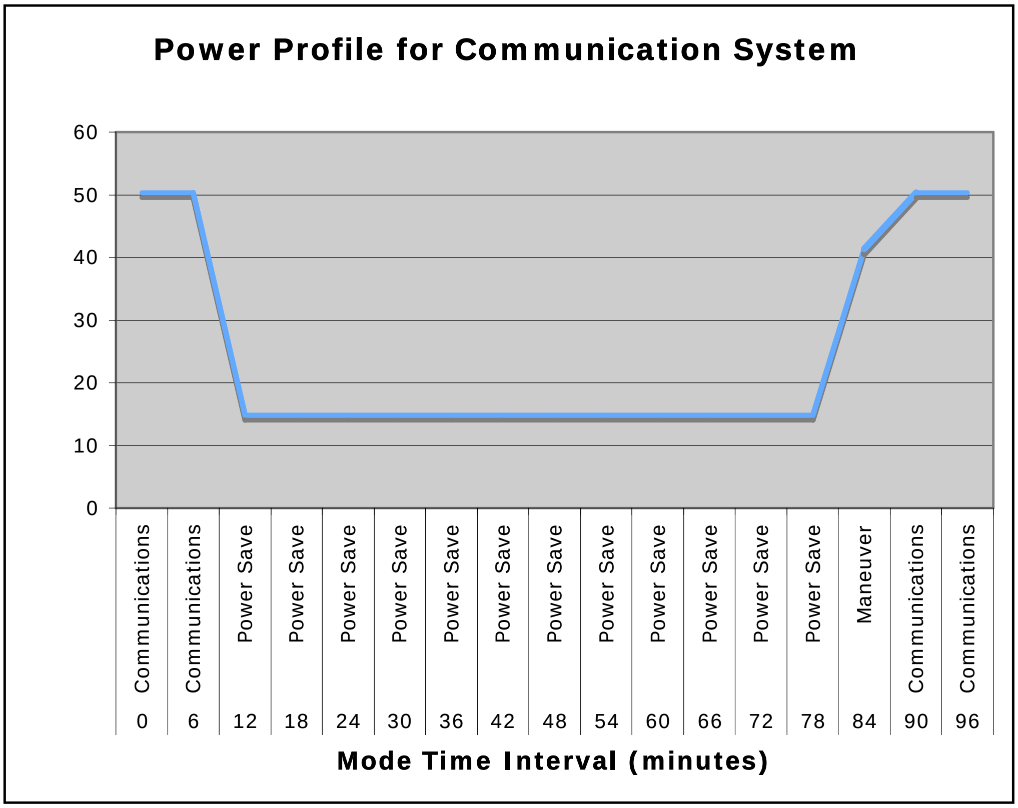


Figure 1: Power profile for Communication System

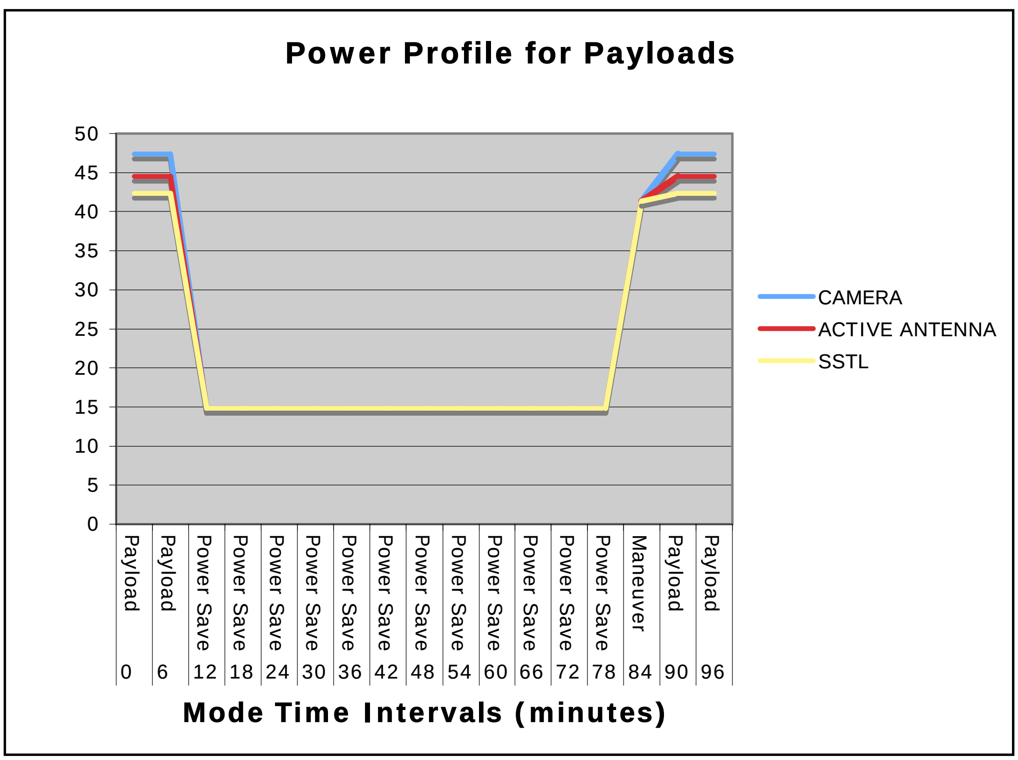


Figure 2: Power profile for Payloads

Components for each subsystem are chosen to fit within the specifications of the above power profile. In the flow chart in Figure 4 is an outline of the six subsystems integrated in our spacecraft, they are:

- Payload
- Command & Data Handling
- Attitude and Control
- Communications
- Power
- Thermal

The components for each subsystem are grouped together and color coordinated for easy readability. A dash line shows power distribution, and the best approximation of power each component needs is listed inside each box. There are four major interfaces that re used to transfer data throughout the spacecraft they are IEEE 1394 and RS232/42/485. The transfer of data is represented by a solid black line and flows in the direction the arrows are pointing. Those subsystems and payloads that must be switched on and off at designated times during the polar orbit are connect to the control board by a solid red line. Each component's specifications and why they were chosen is explained in further detail in the following subsystems report.

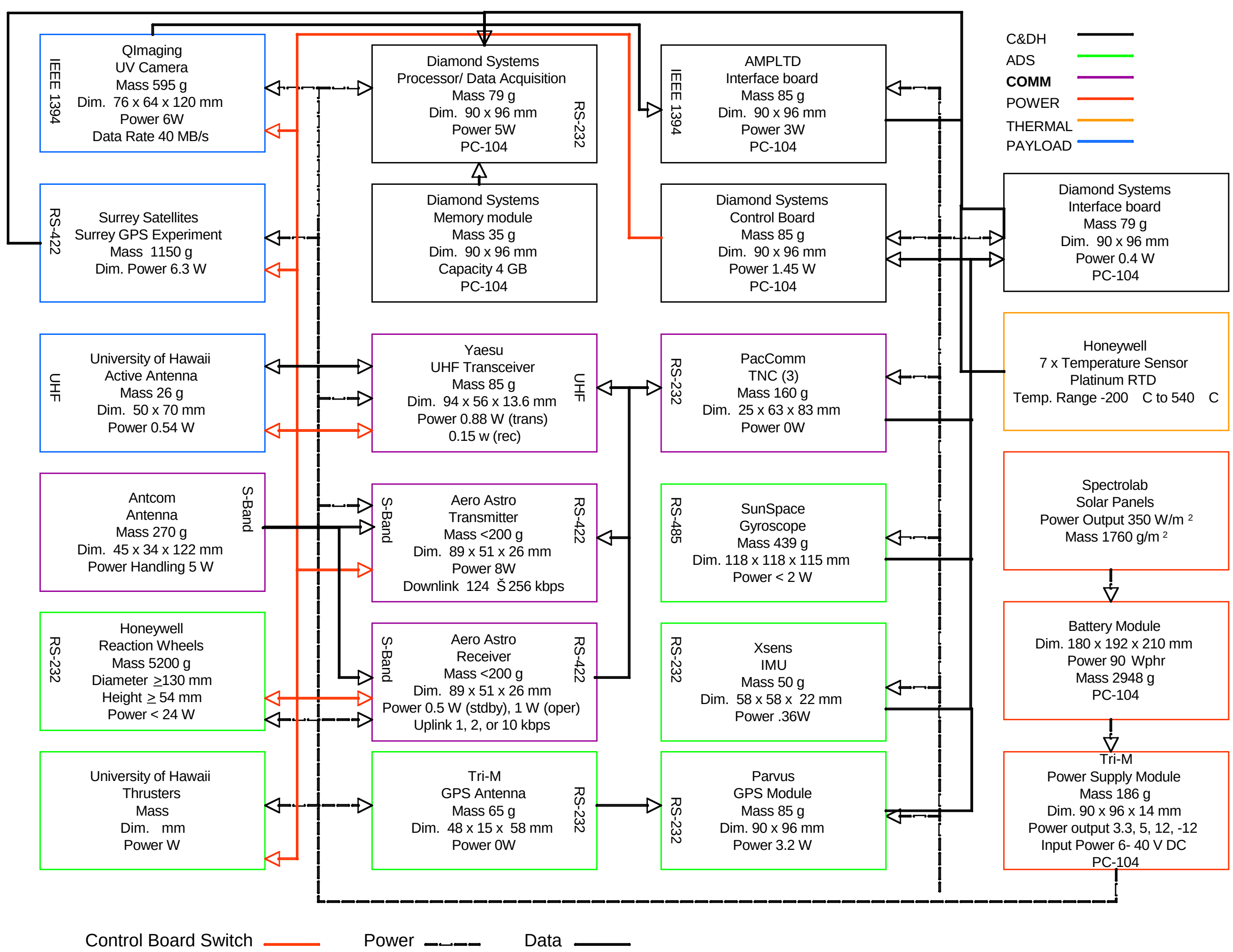


Figure 3: Flowchart of satellite's subsystems

## PAYLOADS

Payloads are merely the instruments, devices, imagers or experiments the spacecraft carries and houses to perform the given tasks for the desire mission and is the main reason for developing the spacecraft initially. Our philosophy when developing our satellite concept design was to create a bus that possesses the capabilities to easily integrate and deploy various payloads. We hope to provide a means to achieve space validation for new technologies and act as a testbed for low earth orbit experiments.

To establish a baseline of what components are needed in order to develop our satellite we have chosen three payloads that mark the furthest capability that the spacecraft can deliver. To scope the capabilities of our satellite we have chosen an imager, a low earth orbit experiment and a science technology development from the various agencies interest in our satellite concept design.

### Ultra-Violet & Visible Camera

The primary payload on our satellite will be an ultra violet camera that will have the ability to take pictures in both visible and ultra-violet light. Pictures taken by the UV camera will provide proof of our launch and can be used as a means to monitor the ecosystem of Hawaii. Picture taken in visible light could help environmental and economic development.

Figure 4: QICAM-UV

Requirements

- IEEE 1394 FireWire Interface
- Power Supply 8-24 V
- Sustain Data Rate 40 MB/s

We will be using the QICAM-UV by QImaging, a digital camera that has high resolution UV and visible range with scientific and industrial imaging applications. QICAM-UV has a spectral range that extends to 200 nm in the UV region and provides a resolution of 1.4 million pixels in a 12-bit digital output. With a high-speed readout it produces linear image data at a maximum frame rate of 205 fps.

| Specifications (UV-Camera) | |
|---|---|
| Dimensions | 76 x 64 x 120 mm |
| Operating Temperature | 0ºC/+35ºC |
| Mass | 595 g |
| Power Consumption | 6 Watts |
| Exposure/Integration Control | 12 $\mu$s to 17.9 min in 1 $\mu$s increments |
| Optical Interface | ½”, C-Mount optical format |
| Light Sensitive Pixels | 1.4 million; 1392 x1040 |
| Pixel Size | 4.65 $\mu$m x 4.65 $\mu$m |
| Doppler Velocity | 0.5 m/s |

Table 2: Specifications for QICAM-UV

## Active Antenna

Dr. Wayne Shiroma and The University of Hawaii CubeSat team developed the Active Antenna known as a grid oscillator. The grid oscillator was developed for application with The University of Hawaii's CubeSat and as an alternative to the usual low frequency range UHF/VHF.

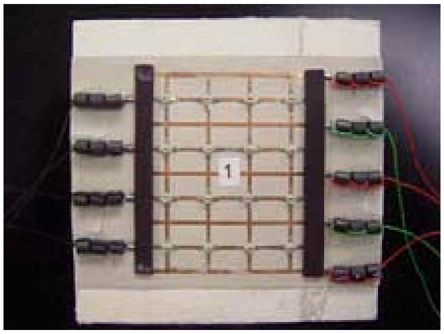

Figure 5: CubeSat Team's Active Antenna

Requirements

- Power Supply 1.5 V
- 4 mm buffer beyond the housing wall

The grid oscillator transmits at higher frequencies and has an efficient power-combining scheme with a compact design. Unlike conventional wire or strip antennas the grid oscillator does not need to be deployed nor does it require additional circuitry. The grid oscillator's built in redundancy makes it tolerant to single point failure, which decreases the chance for component failure. If the Active Antenna is successful it will help lay the groundwork for future data-intensive CubeSat missions.

| Specifications (Active Antenna) | |
|---|---|
| Dimensions | 50 x 70 x 12.7 mm |
| Mass | 26 g |
| Operating Frequency | 5.76 GHz |
| Power Consumption | 0.54 Watts |
| Transmitted Power | 20.33 dBm |

Table 3: Specifications for Active Antenna

## Surrey Satellite Technology Ltd. Space GPS Receiver

Surrey Satellite Technology Ltd. has interest in testing their technology on our satellite. Surrey will aid us in integrating their GPS receiver to our bus.

Requirements

- Power Supply 18-38 V
- RS422 Interface
- Minimum three antennas to determine altitude

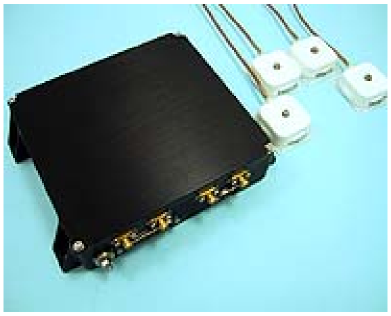

Figure 6: SSTL GPS Receiver

Surrey GPS receivers are known for providing GPS standard time, position and velocity in a compact unit. From 24 hours of data Surrey's GPS receivers have the ability to deliver onboard orbit knowledge to within several meters. Surrey's SGR-05/10/20 GPS receivers each decode and receive L-Band signals, and have the ability to calculate the position within 10 meters. The code is stored in Flash memory which enables the receiver to boot rapidly and gives it the ability to upgrade while in orbit. In addition, Surrey GPS receivers have a separate TTC node that gives it telemetry and telecommand from the primary processors.

| Specifications (SSTL GPS Receiver) | |
|---|---|
| Dimensions | 160 x 160 x 50 mm or 295 x 160 x 35 mm |
| Operating Temperature | -20ºC/+50ºC |
| Mass | 950 g (Unit), 50 g (antenna) |
| Power Consumption | 6.3 Watts (28 V) |
| Orbital Position (3-D) | 10 - 20 m |
| Orbital Velocity (3-D) | 0.15 - 0.25 m/s |
| Attitude Determination | 0.5º - 1º |
| Pseudorange | 0.9 m |
| Doppler Velocity | 0.5 m/s |

Table 4: Specifications for SSTL GPS Receiver

## COMMAND & DATA HANDLING (C&DH)

The Command & Data Handling subsystem collects data and telemetry of the satellite and manipulates data sets from payload. It runs and stores all necessary software for the spacecraft and controls all functions carried out by each subsystems. The C&DH implements all commands and direction received by the ground station and relays the data to the correct subsystem. It directs the modes of the spacecraft while it orbits the Earth and ensures each one is operating at the proper time. Each board in our C&DH subsystem will use the PC-104 form factor, which will make assembly of the subsystem easier and thus eliminate possible error created through integration of different buses.

### Processor Board

The processor board is the brain of the entire spacecraft. It processes all data and runs all software for each subsystem. Limited by the total mass and power of spacecraft, it was important to have a small, low power consumption processor. We also want our board to use the PC-104 bus because it will simplify the assembly when integrating each subsystem.

Figure 7: Prometheus Processing Board

Requirements

- Mass ≤ 100 g
- Power Consumption ≤ 5 W
- PC-104 Bus Interface
- Processing Speed ≥ 100 MHz
- Be able to withstand temperatures -40ºC to 85ºC

Our spacecraft uses a Prometheus Z32-E-ST, manufactured by Diamond Systems Corporation, a two in one processor and data acquisition board to control each task. The built in data acquisition saves space which we are limited on and prevents us from needing to incorporate an additional board. Prometheus processor uses a ZFx86 microprocessor chip, by ZF Micro Solutions Company, with the ZF FailSafe System that has the ability to recover from software crashes or operating system corruption without human intervention. The Prometheus is a low power cost processor with industrial temperature rating. Besides ensuring that all subsystems are running properly the processor will process pictures taken from the UV-camera and run the software for Dr. Shiroma's and Surrey's experiment. The Prometheus' speed of 100MHz with only 5 watts of power is plenty. The Diamond Systems Corp. offers the option of adding a flashdisk that will allow us to store up to 4GB of data between downlinks.

| Specifications (Prometheus) | |
|---|---|
| Processor | ZF Micro ZFx86 |
| Speed | 100 MHz |
| Power Consumption | 5 Watts |
| Serial | 4 RS-232 |
| Ethernet | 10/100 |
| USB | 2 X 1.1 |
| # Inputs | 16 SE , 8 D/I |
| # Outputs | 4 |
| # Digital I/O | 24 I/O |
| Operating Temperature | -40ºC /+85ºC |
| Dimensions | 90 X 96 mm |

Table 5: Specifications for Prometheus Processing Board

Figure 8: Flashdisk memory

## Interface Boards

The interface board provides a way for all the subsystems to communicate to the processor board. It acts as translator of data received from the Payloads, Attitude Control and Communication System. The interface boards are key elements in our mission concept which will demonstrate the ability to produce a microsatellite that has "plug and play" capabilities i.e., fully operational without the need build or design special interfaces to make components compatible. The "plug and play" capability will allow us to maneuver components around in the spacecraft to accommodate payloads and experiments.

Requirements

- Mass $\leq$ 100 g
- Power Consumption $\leq$ 3 W
- PC-104 Bus Interface
- IEEE 1394 FireWire with data rate of 400 MB/sec
- 4 $\leq$ RS-232 Ports
- 2 $\leq$ RS-422 Ports
- 2 $\leq$ RS-485 Ports
- Be able to withstand temperatures -40ºC to 85ºC

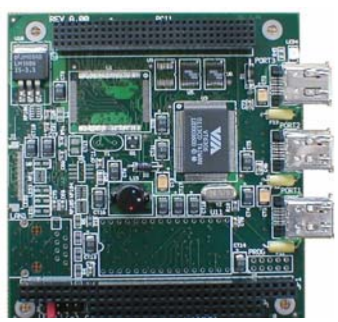

Figure 9: FireSpeed2000 Interface Board

The C&DH subsystem contains two interface boards the Emerald-MM and the FireSpeed2000. The interface boards consist of IEEE 1394 and RS-232/422/485 ports, which makes integration of the UV-Camera and the other subsystems possible. The Surrey Satellite Experiment has a TBD interface, but we have approximately 9 watts of power and 5kg to accommodate any necessary boards that need to be integrate for the experiment.

Figure 10: Emerald-MM Interface Board

| Specifications (Emerald-MM) | |
|---|---|
| Interfaces | RS-232/422/485 |
| PC-104 Capable | Yes |
| Power Consumption | 0.4 Watts |
| # RS-232 ports | 4 |
| # RS-422 ports | 2 |
| # RS-485 ports | 2 |
| Operating Temperature | -40ºC /+85ºC |
| Dimensions | 90 X 96 mm |

Table 6: Specifications for Emerald-MM Interface Board

| Specifications (FireSpeed 2000) | |
|---|---|
| Interfaces | IEEE 1394 FireWire |
| Transfer rate | 100/200/400 MBits/sec |
| PC-104 Capable | Yes |
| Power Consumption | 3 Watts |
| Operating Temperature | -40ºC /+85ºC |
| Dimensions | 90 X 96 mm |

Table 7: Specifications for FireSpeed 2000 Interface Board

## Control Board

The control board acts as the operations director it ensures that only the proper subsystems are running and those not needed are off. The control board is more specifically a counter/timer and digital input/output module.

Figure 11: Quartz-MM-10 control board

Requirements

- Mass $\leq$ 100 g
- Power Consumption $\leq$ 2 W
- PC-104 Bus Interface
- 7 < Counters
- Be able to withstand temperatures -40ºC to 85ºC

We will use a Quartz-MM-10 (QMM-10) to control the operations of the spacecraft. The QMM-10 has 10 counters each 16 bits wide, which is enough for each payload and every subsystem. It is also made by Diamond Systems Corporations so integration with the other C&DH boards should be no problem.

| Specifications (Quartz-MM-10) | |
|---|---|
| # Inputs/ Outputs | 8 in, 8 out |
| PC-104 Capable | Yes |
| Power Consumption | 1.45 Watts |
| #Counters | 10 |
| Resolution | 16 bits |
| Max clock input rate | 20 MHz |
| Operating Temperature | -40ºC /+85ºC |
| Dimensions | 90 X 96 mm |

Table 8: Specifications for Quartz-MM-10 control board

## ATTITUDE CONTROL SYSTEM (ACS)

The Attitude Control System determines and controls the spacecraft's location in space and its orientation relative to the Earth. The ACS is vital for placing the spacecraft in the proper position to carry out experiments, pointing for taking pictures and downlinking and uplinking data.

### Global Positioning System (GPS) Receiver

Figure 12: COMM-1288 GPS Receiver

The GPS subsystem allows us to compute the spacecraft's position and velocity. Like the C&DH subsystem we want the GPS module to have the PC-104 architecture to allow for easy integration with C&DH subsystem.

Requirements

- Mass $\leq$ 100 g
- Power Consumption $\leq$ 5 W
- PC-104 Bus Interface
- Be able to withstand temperatures -40 °C to 85°C

We will use the COMM-1288 GPS module by Parvus for our spacecraft. The COMM-1288 integrates on a single PC-104 board, a high-speed Triband 900/1800/1900MHz GSM/GPRS modem and a low power 12-channel parallel tracking GPS receiver.

| Specifications (Comm-1288) | |
|---|---|
| GPS Receiver | iTrax02 |
| Power Consumption | 1.1 Watts (idle)<br>3.2 Watts (peak) |
| # Serial Ports | 4 x RS-232 |
| Mass | 85 g |
| Operating Temperature | -40 ºC/+85 ºC |
| Dimensions | 90 x 96 x 15 mm |

Table 9: Specifications for COMM-1288 GPS Receiver

## GPS Antenna

We will use the Big Brother GPS Antenna by Tri-M Systems and Engineering. The GPS Antenna allows us to amplify the GPS signal to the GPS receiver. We chose the Big Brother antenna for its durability and for its built in low noise amplifier.


Figure 13: GPS Antenna

| Specifications (GPS Antenna) | |
|---|---|
| Center Frequency | 1575.42 MHz |
| Noise Figure | 2.0 max |
| Bandwidth | 2MHz min. |
| Output Impedance | 50 ohm |
| Mass | 65 g |
| Operating Temperature | -40 ºC/+85 ºC |
| Dimensions | 48 X 15 x 58 mm |

Table 10: Specifications for GPS Antenna

## Orientation System

The GPS system will give the spacecraft's location in space, but the orientation to Earth will be determined using the Inertial Measurement Unit (IMU) and a gyroscope. The IMU tracks rapidly changing orientations in 3D and measures the directions of gravity and magnetic north to provide a stable reference, and the gyroscope will provide additional 3-axis inertial measurements. We will use an IMU by Xsens and a gyroscope by SunSpace. The Sunspace Gyroscope has flight heritage on TUBSAT.

Figure 14: IMU

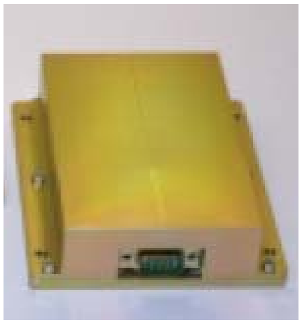

Figure 15: Gyroscope

| Specifications (IMU) | |
|---|---|
| Dynamic Range | All angles in 3D |
| Angular Resolution | 0.05 deg |
| Power Consumption | 0.36 Watts |
| Digital Interface | RS-232 |
| Mass | 50 g |
| Operating Temperature | 0 ºC/+55 ºC |
| Dimensions | 58 x 58 x 22 mm |

Table 11: Specifications for IMU

| Specifications (Gyroscope) | |
|---|---|
| Measurement Range | 80 deg/s |
| Power Consumption | < 2 Watts |
| Data Interfaces | RS-232/485 |
| Mass | 439 g |
| Dimensions | 99 x 117 x 31 mm |

Table 12: Specifications for Gyroscope

## Reaction Wheels

Reaction wheels allow us to adjust the spacecraft's orientation and inertial rates. The spacecraft will have three reaction wheels one for each axis of motion.

Requirements

- Mass $\leq$ 2000 g
- Power Consumption $\leq$ 7 W
- Data Interface RS-232/422/485
- Be able to withstand temperatures -25 °C to 55°C

We chose to use Honeywell's miniature reaction wheels to control our spacecraft's momentum. The Honeywell gyroscope is built based on commercial and military aircraft gyroscope technology. It has a compact design with high momentum-to-mass efficiency and uses a DC brushless-spin motor.

Figure 16: Reaction Wheel

| Specifications (Reaction Wheels) | |
|---|---|
| Angular Momentum | 0.2 to 1.0 Nms |
| Wheel Torque Output | > 28 mNm |
| Wheel Speed Range | $\pm$ 9000 rpm |
| Data Interfaces | RS-422 |
| Mass | 1300 g |
| Operating Temperature | -25 ºC/+60 ºC |
| Dimensions | 130 mm (diameter) 54 mm (height) |
| Power Consumption | < 6 Watts |

Table 13: Specifications for Reaction Wheel

## COMMUNICATION SYSTEM (COMM)

The COMM subsystem transmits and receives all data and commands between the spacecraft and the receiving ground station. We will use S-Band frequency to send information, in part because X-Band, though faster, is a frequency that is over used, therefore, making it difficult to communicate to our spacecraft due to "cross talking".

### Receiver (S-Band)

The receiver's main objective is to collect commands and directions sent from the command station on earth and relay them to the processor.

Requirements

- Mass $\leq$ 500 g
- Power Consumption $\leq$ 8 W
- Uses Interfaces RS-232/422/485
- Capable to receive S-Band frequency
- Uplink rate $\geq$ 10 kbps

For our spacecraft we would use the S-Band receiver by AeroAstro Company. The AeroAstro receiver is one of the lowest power cost receiver; with only one watt of power, it uplinks data at a rate of 10 kbps. The receiver module is compact in size at 89 mm x 51 mm x 25 mm, making it no longer than a PC-104 board.

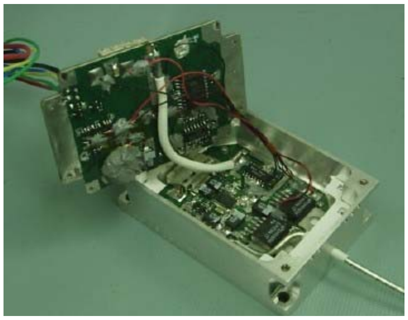

Figure 17: Inside an AeroAstro communication module

| Specifications (Receiver) | |
|---|---|
| *Receive Frequency* | *1760 – 1840 MHz* |
| *Interface* | *RS-422* |
| *Power Consumption* | *0.5 Watts (stdby)*<br>*1 Watts (operational)* |
| *Input Data Rate* | *10 kbps* |
| *Operating Temperature* | *-30 ºC/+60 ºC* |
| *Dimensions* | *89 X 51 x 25 mm* |
| *Mass* | *< 200 g* |

Table 14: Specifications for Receiver

### Transmitter (S-Band)

The transmitter relays all data received from the payloads and information concerning the status of each subsystem to the command station. The transmitter can also act as a Doppler range finder to locate the spacecraft in orbit by measuring the time a signal is sent out and returns back.

Requirements

- Mass $\leq$ 500 g
- Power Consumption < 35 W
- Uses Interfaces RS-232/422/485
- Capable to receive S-Band frequency
- Uplink rate $\geq$ 125 kbps

Figure 18:  AeroAstro Receiver/Transmitter

We will also being using an AeroAstro S-Band transmitter for our spacecraft.  The AeroAstro transmitters have been flight-proven and were used in the Canadian Space Agency's MOST mission in which it is still fully functional.  Once again the AeroAstro module was the lowest power cost module we discovered.  It has the ability to downlink at 124-256 kbps with only 8 W DC.

| Specifications (Transmitter) | |
|---|---|
| Transmit Frequency | 2200 – 2300 MHz |
| Interface | RS-422 |
| Power Consumption | 8 Watts |
| Output data rate | 125 – 256 kbps |
| Operating Temp. | -30 ºC/+60 ºC |
| Dimensions | 89 X 51 x 25 mm |
| Mass | < 200 g |

Table 15:  Specifications for Transmitter

## Antenna (S-Band)

The S-Band antenna acts as a beacon for the transmitter and receiver. Data is received and release through the antenna.  On our spacecraft we will have two S-band antennas for the transmitter and receiver.   When searching for an S-Band antenna our main concern was to find the smallest antenna possible in order attain the most surface area possible for the solar panels.

Figure 19:  S-Band Antenna

Requirements

- Mass $\leq$ 500 g
- Capable to receive S-Band frequency
- Operational altitude > 4.83 km
- Dimensions $\leq$ 45 x 140 mm

We will use Antcom's S-Band antenna for its small size and its durability.  It has the ability to withstand altitudes up to 21.3 km and vibrations up to10 G's.

| Specifications (S-Band Antenna) | |
|---|---|
| Frequency | S-Band |
| Power Handling | 5 Watts |
| Vibration | 10 G's |
| Altitude | 21.336 km |
| Material | 6061-T6 Al alloy base |
| Operating Temperature | -55 ºC/+85 ºC |
| Mass | 270 g |
| Dimensions | 44.5 x 133.4 x 121.2 mm |

Table 16:  Specifications for S-Band Antenna

## UHF Transceiver

The UHF Transceiver will collect and release data during Active Antenna experiment. It will also act as a backup for the S-Band transmitter and receiver if they should happen to fail.

Requirements

- Mass $\leq$ 500 g
- Capable to receive UHF frequency
- Uses Interfaces RS-232/422/485

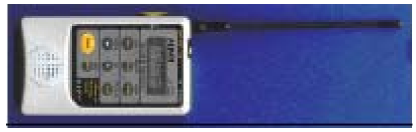

Figure 20: Yaesu Transceiver

We will use the same transceiver Dr. Shiroma's team picked for their cubesat which is an Alinco DJ-CST transceiver built by Yaesu Company. Dr. Shiroma's team chose the transceiver due to its relatively low power consumption, price and compact size.

| Specifications (UHF Transceiver) | |
|---|---|
| ***Dimensions*** | ***94 x 56 x 13.6 mm*** |
| ***Operating Temperature*** | ***-10 ºC/+60ºC*** |
| ***Power Consumption*** | ***0.15 Watts (rec)<br>0.88 Watts (trans)*** |
| ***Mass*** | ***85 g*** |

Table 17: Specifications for Yaesu Transceiver

## Terminal Node Controller (TNC)

We will use three PicoPacket TNCs by PacComm to assist the Communication modules. The TNC is a modem that runs packet firmware that allows you to send a message containing a long status report, such as listing various types of equipment, service data, and current condition. The firmware arranges the message into "packets" each having specific length and formatted into bits. The TNC then sends the formatted data to the transmitter where it is transmitted. The receiving station will be running the same TNC protocol and knows that each packet should be a certain length. If interference causes the TNC not to receive the expected message, the receiving TNC automatically tells the transmitting TNC what it received and what it did not. The transmitting TNC re-transmits the missed packets. This occurs at the speed of light and results in perfect data transmission.


Figure 21: TNC

| Specifications (TNC) | |
|---|---|
| ***Power Consumption*** | ***0 Watts*** |
| ***Data Interfaces*** | ***RS-232*** |
| ***Mass*** | ***160 g*** |
| ***Dimensions*** | ***25 x 63 x 83 mm*** |

Table 18: Specifications for TNC

## POWER SYSTEM

The Power subsystem consists of solar panels and batteries that provide all the necessary power for each subsystem and payload to run properly.

### Solar Panels

Solar panels are vital when making an autonomous spacecraft. They are responsible for powering the subsystems and payloads and recharging the battery packs. Our main objective when choosing solar panels is to attain the most efficient and least dense solar panel. To eliminate mechanical error we have chose not to deploy the solar panels. Solar panels will be attached both magnetically and non-magnetically.

Requirements

- Mass $\leq$ 2 kg/m$^2$
- Power $\geq$ 290 W/m$^2$

Figure 22: Spectrolab Solar Panels

We will use ultra triple junction (GaInP2/GaAs/Ge) solar cells, made by Spectrolab that will provide 350 W/m$^2$ with a mass of 1.76 kg/ m$^2$ to our spacecraft. While directly in the sun our spacecraft will have three faces fully exposed. Each face has a surface area of 0.07 m$^2$ that will give our spacecraft a minimum 73.5 watts of power for the entire spacecraft. That gives us a margin of 10 watts from total power, computed for the entire spacecraft.

| Specifications (Solar Panels) | |
|---|---|
| ***Power*** | ***350 W/m$^2$*** |
| ***Mass*** | ***1.76 kg/m$^2$*** |
| ***Material*** | ***GaInP$_2$/GaAS/Ge*** |
| ***Low Earth Orbit life Expectancy*** | ***5 years*** |

Table 19: Specifications for Spectrolab Solar Panels

### Battery packs

Battery Packs provide a way to supply energy to the spacecraft while it is eclipsed by the Earth and store power while directly in the sun.

Figure 23: BAT104-SLA45

Requirements
- Mass ≤ 2 kg
- Output voltage ≥ 8.4 V
- Operational Temperature ≥ -40 to 65ºC

Energy on the spacecraft will be stored using Sealed Lead Acid BAT104-SLA45 battery packs manufactured by Diamond Systems Corporation. The BAT104-SLA45 battery packs are built in PC-104 form so they are easily mounted and assembled with other PC-104 boards. The pre-built battery pack eliminates need for assembly of our own battery module and lessens the chances for error. There will be two battery packs each with a max output voltage of 10 V and a storage capacity of 4.5 A-Hr.

| *Specifications (Battery Packs)* | |
|---|---|
| *Max output voltage* | *10 V* |
| *Capacity* | *4.5Ah* |
| *Dimensions* | *90 x 96 x 106 mm* |
| *Mass* | *1474 g* |
| *Temperature range* | *-65 to 65ºC* |

Table 20: Specifications for BAT104-SLA45

### Power Supply Board

The power supply board distributes power received from battery packs and solar panels to the proper payloads and subsystems.

Requirements
- Mass ≤ 200 g
- Power Input range 6 to 40 V
- PC-104 Bus Interface
- Output Voltage range of 3 to 12 V
- Operational temperatures -40 to 85ºC

Our C&DH system contains three power supply boards that are capable of distributing power of 3.3, 5, 12, -12 V and each has the ability to receive power of 6-40 V DC.


Figure 24: Power Supply Module

## THERMAL SYSTEM

The Thermal subsystem will monitor and regulate the temperature on each payload and subsystem. Temperature regulation is vital for the success of the mission to ensure that all subsystems and payloads stay operational.

### Temperature Sensors

The temperature of each payload and subsystem will be monitored through temperature sensors placed on every board and payload.

Requirements

- Temperature Range -60 ºC to 90ºC
- Accuracy < 1 ºC

We will use Honeywell's HEL-700 Platinum RTD to provide linearity accuracy, stability and interchangeability. Resistance on HEL-700 changes linearly with temperature. The HEL-700 has a large temperature sensing range of -200 to 540 ºC.

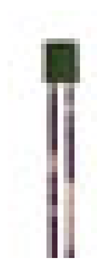

Figure 25: RTD Temperature Sensor

| Specifications (Temperature Sensors) | |
|---|---|
| *Sensor Type* | *100 Ohm Platinum RTD* |
| *Temperature Range* | *-200 ºC to 540ºC* |
| *Temperature Coefficient* | *.00375 Ohm/Ohm/ºC* |
| *Packaging Type* | *Radial chip, SMT axial flip chip* |
| *Self Heating* | *> 0.3 mW/ºC* |

Table 21: Specifications for Temperature Sensor

### Heaters

Thermofoil$^{TM}$ heaters by Minco will be controlled by the C&DH systems to ensure that none of the subsystems or payloads freezes while the Earth eclipses the spacecraft during its orbit. The Thermofoil$^{TM}$ heaters can safely run at wattages twice those of their wire-wound equivalents. Thermofoil$^{TM}$ heaters transfer heat more efficiently, over a larger surface area, than round wires.

| *Specifications* Thermofoil$^{TM}$ | |
|---|---|
| *Material* | *Kaplan/FEP* |
| *Temperature Range* | *-200 to 200ºC* |
| *Max Resistance Density* | *8-70 ohms/cm²* |
| *Thickness* | *.25 mm* |
| *Mass* | *0.04 g/cm²* |

Table 22: Specifications for Thermofoil$^{TM}$

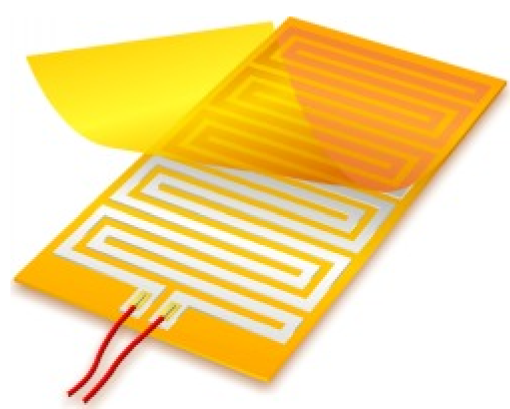

Figure 26: Thermofoil$^{TM}$

### Dissipation of Heat

We will dissipate heat from the spacecraft by placing radiator panels, conductive strapping and shear panels in the spacecraft. Conductive strapping and shear panels will transfer heat from the subsystems to the radiator panels where it will then be released into space. Our orientation in orbit will also assist in protecting the subsystems thermally. Our spacecraft will orbit the Earth with its octagon side facing space, which allows us more surface area to dissipate heat and ensure that the subsystems stay cool.

## SPACECRAFT BUS

The initial conceptual design of the satellite can be seen in Appendix A. The satellite is divided into four sections by shear panels, in which the components for each subsystem are mounted to. Components are mounted to shear panels to help disperse heat, minimize interface issues and eliminate the need for extra wiring.

## CONCLUSION

Our next step with the completion of our satellite concept study is to finish our Pre-Phase A analysis. Afterwards we will do a cost analysis of our spacecraft structure and its components. We plan to begin construction of our prototype spacecraft in mid August and anticipate launching in 2008.

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