School of Aerospace Mechanical and Mechatronic Engineering

AERO3760: Space Engineering 2

SnapSat Preliminary Design Report

21 August 2015

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1 Introduction

SnapSat is a design solution as a part of the AERO3760: Space Engineering 2 course at the University of Sydney. The project involves specific design specifications as set out by the course administrator and the CubeSat requirements [?]. This report details the selected mission and preliminary design considerations. Satellite components will be purchased off the shelf, where financially viable. Otherwise, components will be made in-house. Over the course of the project development, components will be tested separately and a series of final testing will be conducted once assembly is completed. SnapSat will undergo vibration testing, vacuum testing, communications and power testing and the attitude determination and control system will be tested on an air-bearing table.

2 Spacecraft Deign Overview

Summarised in table 2.1 below is the outline of all components in the SnapSat proposed design.

Table 2.1: SnapSat Design Overview

Subsystem	Description
Structural	- industrial grade aluminium - laser cut, bent to shape and riveted together
ADCS	- air core magneto-torquers made in-house - Osram SFH203P Photodiodes - IMU: Adafruit 9-DOF
EPS	- Australian Robotics solar panels - battery: LiNiMnCo 26650 rechargeable cell
OBC / OBDH	- Arduino DUE - 4 × PCBs (power, control, EPS and communications)
TT&C	- VHF (Xbee) - UHF - tape measure antennae
Thermal	- thermal tapings and passive coatings (Kapton tape) - selected components will have multi-layer insulation
Payload	- Arducam

2.1 Subsystem Design Schematic

The layout of Snapsat, with the interconnects of power and data lines between the subsystems is shown in the figure below.The Solar panels will be arranged according to Figure 2.1. This design has been chosen to maximize the solar panel area of the SnapSat while still abiding by the volume/size limitations of CubeSats. Increasing the solar panel area has given us the ability to power a higher quality camera, increasing the performance of the payload and usefulness of the SnapSat.

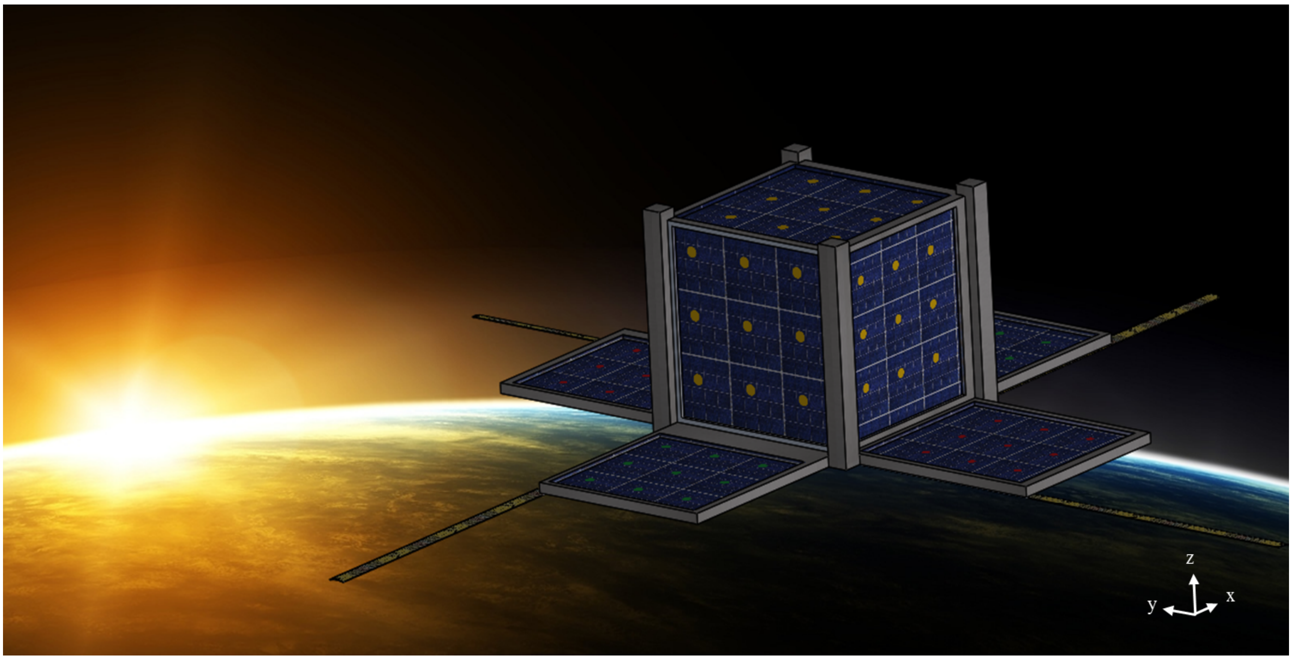


Figure 2.1: Design Schematic for SnapSat - the colours indicated the panels that power the batteries. There are three individual batters and so three sets of solar panels to power them

In the following report yaw will be refereed to as rotation about the z axis, roll about the x axis and pitch about the y axis (although these are relatively interchangeable as the SnapSat is rotationally almost symmetric).

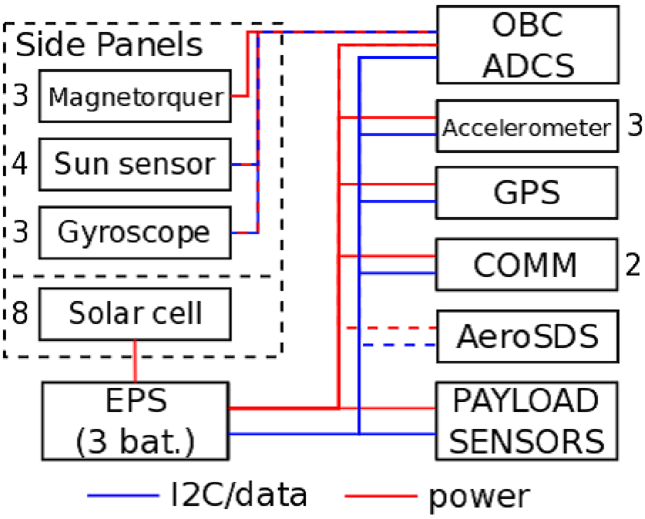


Figure 2.2: Design Schematic

3 Payload Design Overview

The CubeSat had been pivotal to the space research and development industry and had generally increased our accessibility to the cosmos. The cubesat platform uniquely offers an extremely low construction and launch cost in comparison to major satellite manufacturers. This has spurred on many educational bodies and small research groups to collect and analyse their own data, especially in developing nations [1]. Universities have pioneered the build of the smallest of satellites (nano- and pico-), this has been assisted with the miniaturisation in many technological fields such as electronics, materials ans sensors. These small sizes enable the cubesat to 'piggy-back' on the launch of much larger satellites, because of this they are able to get to very high (and expensive) orbits of a fraction of the price. Many big aerospace companies have also made use of the tiny platform such as Orbital Sciences (2006 [2]) and Boeing (2009 [3]) along with the United Nations, who have formally recognised the developmental benefits of small satellites [4].

Despite the quick growth of this industry in the aerosapce and related academia fields, we are only now seeing the cubesat break into early STEM education. Currently, only science and technology scholars and graduates have the full accessibility to the design of cubesats, our ease of access is not well known amongst the general public. Snapsat hopes to change this, by bringing space to social media via beautiful photographs and Twitter. Snapsat is a nano-satellite designed for outreach and space accessibility for educational bodies and the general public. In a sun-synchronous orbit at an altitude at 350km, Snapsat will be in the prime position of Earth observation. Users and sponsors can send a message to the cubesat, which will take a low resolution image of the Earth and tweet it to the world.

The camera we will be making use a TTL (Through The Lens) camera, which can take snapshots and transmit over a TTL serial link. The image output is a pre-compressed JPEG, making memory storage and transmission easier. This module was also chosen for its ease of integration - requiring only two digital pins, there is an extensive arduino library also available.

Currently, the final build will result in a balloon launch test. Following the success of this, SnapSat will be launched on a sounding rocket, where is will be placed in a low Earth orbit for a maximum lifetime of three months.

4 Spacecraft Modes of Operation

The spacecraft will experience the following modes during its lifetime. A different configuration of system operations and instructions will be executed by *SnapSat* in each case. These are summarised in below.

Launch Mode: This turns the satellite off for launch to comply with CubeSat Design Specification 2.3.1. During launch the deployment switch is tripped which will turn the satellite on and transfer it into Establish Contact Mode.

Startup Mode: This mode is entered only when the CubeSat is first launched. In this mode the satellite will immediately turn on the ADCS to detumble. Once the CubeSat is sufficiently stable (not tumbling in the pitch or roll axis) or 30 minutes has elapsed (CubeSat Design Specification 2.4.2) the antennas and solar panels will be deployed. It will then move into Standby Mode.

Standby Mode In this mode, only essential satellite systems are kept ON. This includes the OBC, EPS, VHF receiver,the GPS and intermittently the IMU. It can transition out of Standby mode via an alert from the GPS, IMU, EPS or ground station orders.

Payload Operation Mode: This mode is used only when taking a picture and is entered through a location alert or ground station orders from Standby Mode. The camera module is booted up, the camera takes a picture, stores it is RAM/ROM and then the camera is powered town again to conserve power. After finishing these tasks it will return to Standby Mode.

Transmissions Mode: This mode is entered once communications is established with the ground station or if the GPS acknowledges that a ground station is in range. It will relay basic telemetry and if a picture is waiting in queue it will be sent. If the EPS detects that the power is too low it will exit Transmissions mode, and will not allow it to enter it again until it has returned to acceptable levels. Similarly the transmitter can be shut down by a command by the ground station.

Power Critical Mode: In the event that the power level drops to a point where Standby is not sustainable SnapSat will enter this mode. All systems will be turned off with the exception of the EPS for the purpose of charging the batteries. It will exit this mode when the batteries are sufficiently charged.

De-tumble mode: This mode is used to de-tumble the spacecraft after deplomment into orbit as well as to recover it from any spin states (such as after Safe Mode). All Safe Mode components are ON, as well as the ADCS system. Other devices can be turned ON by ground command without a change of state.

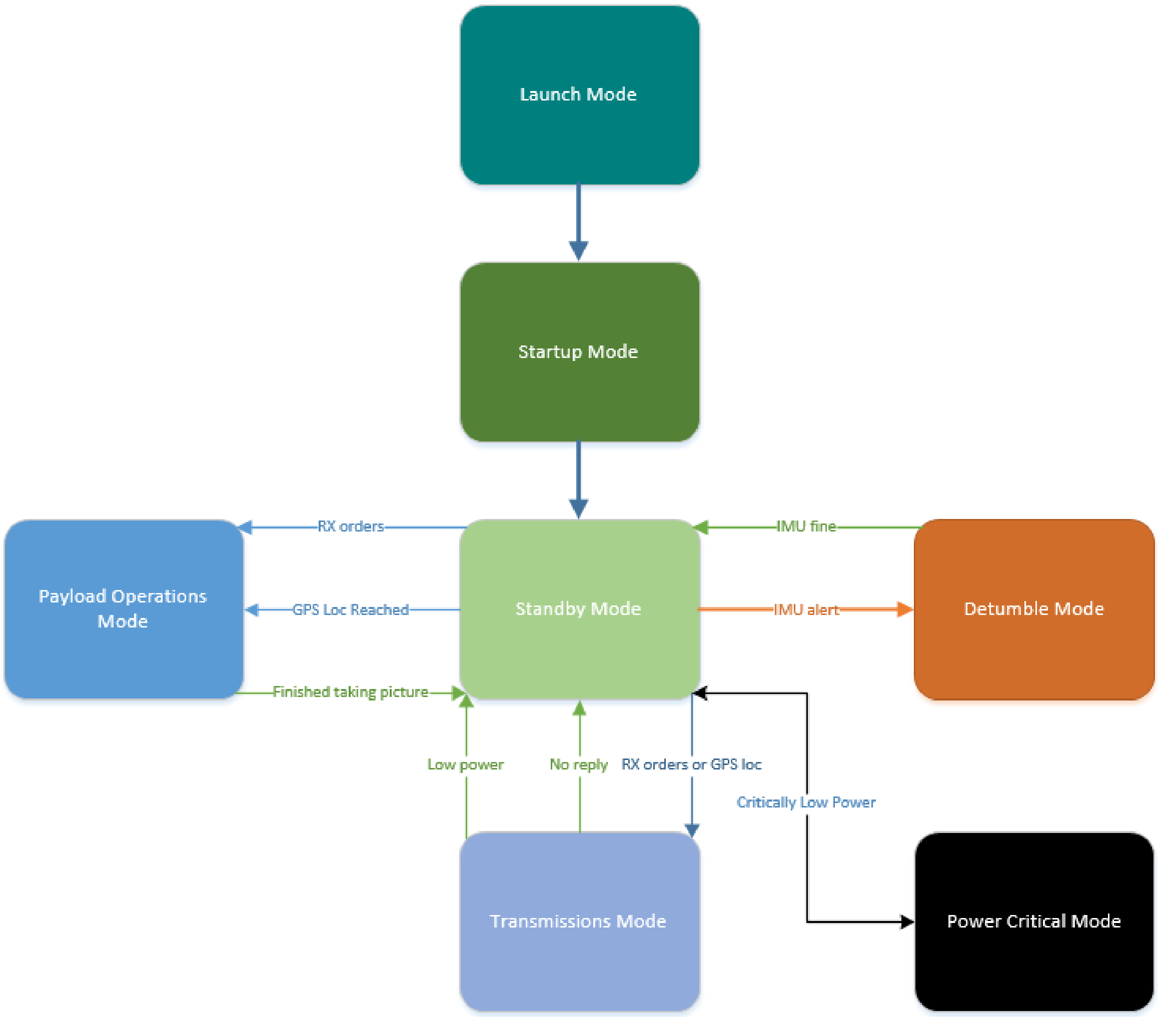


Figure 4.1: Mode Transition Diagram during satellite lifetime

5 System Budgets

5.1 Mass Budget

The mass budget is shown in table 5.1, detailed calculations are included in Appendix A.1. it is ensured that *SnapSat* meets the requirement of a maximum weight of 1kg. The inertial matrix was computed using *Solidworks*. It was found to be:

$$I = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix} = \begin{bmatrix} 297788.17 & 301.92 & 5644.56 \\ 301.92 & 291591.41 & 3021.25 \\ 5644.56 & 3021.25 & 296099.96 \end{bmatrix} \text{g} \cdot \text{mm}^2 \tag{5.1}$$

Table 5.1: SnapSat Mass Budget (target mass of 1000 g)

Subsystem	Mass	Contingency	Mass + Contingency	Fraction of Total Mass
Structural - chassis - solar panels	 100g 9 × 25g	 30g 9 × 5g	415g	47.6%
ADCS - air core coils - sun sensors - IMU	 3 × 50g 6 × 1.8g 2.8g	 3 × 5g 6 × 0.3g 0.4g	180.2g	22%
EPS - batteries - power bus	 103g 10g	 7g 2g	122g	14.9%
OBS / OBDH - Arduino board - memory storage - PCBs	 25g 1g 4x10g	 5g 0.2g 2g	79.2g	9.1%
TT&C - antennae - transceiver	 3g 75g	 1g 5g	84g	10.3%
Thermal - tapings - MLI	 1g 3g	 0.3g 0.5g	4.8g	0.6%
Payload - camera	 25g	 5g	25g	3.1%
Integration - bolts/rivets - cabling/wires	 5g 2g	 2g 0.5g	9.5g	1.2%
Total			865.7g	
Mass Margin			134.3g	15.5%

5.2 Power Budget

The power to the SnapSat will be supplied by 9 solar panels arranged around the cube as shown in Figure 2.1. This design is contingent on 3D printing the SnapSat as the solar panels will have to be depressed into the framework to allow the secondaries to fold up during transport/inside the pea pod. In the event that it is not possible to use the fold our design, only 5 solar panels will be used and the SnapSat will transmit less and take less pictures accordingly.
We will cut solar panels to 94x94 mm to produce an output of approximately 1.5W per panel. This output will be increase again in space due to the lack of atmosphere.
Since we assume the satellite will spend 65% of it's time in the sun $OAP = 0.65 \cdot 1.5 \cdot 1.44$ per panel assuming it is directly in the sun. By observing the layout we can assume that at any given time in the orbit 2.5 solar panels are exposed at varying angles. This gives an OAP of approximately 2.1 W, however the peak input power from the solar panels is 10.8 W (if incident light occurs from above) which the batteries are capable of charging. If cutting the solar panels does not work as expected we will purchase alternate ones and deal with the diminished power supply.
This power budget is provisional and dependent on the orbital height of the SnapSat as this will greatly increase the power consumption of the S-band Tx unit. Further testing will be required to determine the exact distance the 500mW of power will propagate the signal.

Table 5.2: SnapSat Power Budget

			Average Duty Cycle by Mode (%)				
Load	**Power Consumption (mW)**	**Number of Units On**	*Standby Mode*	*Detumble Mode*	*Payload Operations Mode*	*Transmissions Mode*	*Power Critical Mode*
OBC	55	1	25 %	100%	100%	100%	10%
VHF Rx	5	1	100%	100%	100%	100%	0%
S-band Tx	500	1	0%	5%	5%	100%	0%
Magnotorquers	150	1	10 %	100%	0%	0%	0%
Current Sensors	140	1	100 %	100 %	100 %	100 %	100 %
Power Board	140	1	100 %	100 %	100 %	100 %	100 %
Camera	250	1	0%	0%	100 %	0%	0%
IMU	30	1	25%	100 %	25%	25%	0%
Photodiodes	120	1	25%	100 %	25%	25%	0%
GPS	66	1	25%	0%	0%	0%	0%
Sum Loads (mW)			368	665	653	515	286
Efficiency			0.8	0.8	0.8	0.8	0.8
Power Consumed (mW)			450	831	816	644	357
Power Generated (mW)			2100	2100	2100	2100	2100
Power Margin			1640	1269	1284	1456	1743

5.3 Pointing Budget

Since this spacecraft is performing Earth observation, it requires a pointing budget. This refers to the ability to orient the spacecraft towards a target having a specific geographical orientation. Along with the pointing accuracy, the satellite needs to be able to map the location from its own location. Errors in both pointing and mapping accuracies will be discussed here.

The attitude control system for SnapSat will consist of three air core magnetorquers operating on 3 separate planes capable of producing 0.05Am^2 each. Only two of the magnetorquers can work at any one time, which will reduce total power usage for the system. The first component of the determination system is a 9-DOF IMU which will primarily be used in the de-tumble phase due to accumulated error issues with this equipment which are expected to occur later in the mission. The second component is a solar tracker system consisting of six photodiode pins, one on each face, which will be used to accurately determine the attitude of the satellite based on the location of the Sun.

According to the specification data, the IMU will experience a 2% error based on the expected temperature range, although this will increase over the course of the mission due to the accumulated error. Although the exact error will need to be calculated during calibration and testing, based on current literature there are a number of similar solar tracking systems which are able to achieve an accuracy of 0.2% [5]. However given the low budget and subsequently slightly inferior equipment conservative estimate of 0.5% will be used for the solar tracker error. In regards to the magnetorquers expected error based on similar models 1%, although error will be finalised during the calibration and testing phase.

5.3.1 Error Calculation

Due to the fact that the two attitude determination systems will almost always be used separately we have calculated three different total errors. The total errors were calculated using the following formula:

$$\text{System Error} = \sqrt{(\text{IMU error})^2 + (\text{sun sensor error})^2 + (\text{magnetorquers error})^2} \tag{5.2}$$

$$= \sqrt{(2\%)^2 + (0.5\%)^2 + (1\%)^2} \tag{5.3}$$

$$= 2.3\% \tag{5.4}$$

$$= 8.3^\circ \tag{5.5}$$

This is summarised in the table below

Table 5.3: Pointing budget error calculation breakdown

	IMU Error (%)	**Sun Tracker Error (%)**	**Magnetorquer Error (%)**	**Total (%)**	**Total (°)**
Overall System	2.0	0.5	1.0	2.3	8.3
System 1	2.0	-	1.0	2.2	7.9
System 2	-	1.0	1.0	1.1	4.0

The majority of the mission is expected to be spent using system 2 (utilising the sun trackers), which produces an error of 4°. Whilst this is within the range for the widest application of the three mapping scenarios (see Appendix A.2) it is slightly outside of the range of the second more focused scenario. However, it should be noted that these are conservative calculations and the finalised error may be lower than these figures.

5.4 Link Budgets

5.4.1 Uplink Budget

The results of the uplink budget (table 5.4) can be seen in the table below, where it satisfies the requirements for a successful link. The ground station utilises a cross yogi antenna for transmission into space while the satellite receives this by utilising a dipole antenna manufactured from measuring tape. The frequency band that is used for uplink is in the UHF band at 315 MHz. A low power transceiver chip was found that could receive at this frequency and also transmit at a higher frequency so the system is being based on this. This also allows the antennas to be cut to a manageable size giving less chance for a mechanical failure. The modulation utilised is Audio Frequency Shift Keying, over the audio tones of 1200Hz and 2200 Hz. With this we will be using a baud rate of 1200. AFSK is not the most efficient but it is simple and allows for less chance of failure. The expected bandwidth will fit in 20kHz at -30 dBc.

5.4.2 Downlink Budget

See table 5.5 for the budget. As stated above the satellite will be using a dipole antenna for transmission whilst the signal will be received by a high gain antenna at the ground station. The frequency band being utilised for downlink will be in the UHF spectrum at 433MHz, due to the data rate available and the compatibility with the chosen transceiver chip. The modulation used is Binary Phase Shift Keying, which works by modulating the phase of the reference signal. With this we will be using a baud rate of 9600 and the expected bandwidth will also fit into a 20kHz range at -30 dBc.

Table 5.4: Uplink Budget

Information on the System	
Transmitter: Ground Station	**Receiver: SnapSat**
Orbit Altitude: 350km	Elevation: 30 degrees
Slant Range: 652.5km	Weather: Clear Sky
Demodulation Method: AFSK	Cable Length: 20m
Antenna Type (TX): Cross Yagi	Antenna Type: Dipole
Transmitter System (Ground Station)	
Ground Station Transmitter Power Output	100 W 20 dBW
Ground Station Total Transmission Line,Losses	3.4 dB
Ground Station Antenna Gain	18.9 dBi
Ground Station ERIP	35.5 dBW
Down Link Path	
Free-Space Path Loss	132 dB
Satellite Antenna Pointing Loss (10 Âř)	10.6 dB
Ground Station Antenna Pointing Loss (10Âř)	2.7 dB
Satellite Transmission Line Losses	0.5 dB
Atmospheric Loss (30Âř)	0.4 dB
Ionspheric Loss	0.4 dB
Rain Loss	0 dB
Total Loss	146.6 dB
Receiver System (on SnapSat)	
Antenna Gain	2.7 dBi
Effective Noise Temperature at Space (350K/Day)	1345K
Figure of Merrit (G/Ta)	-28.6 dB/K
Carrier to Thermal Noise Ratio (C/T)	-136.6 dB
Boltzmann's constant (K)	-228 dBW/K/Hz
Carrier to Noise Density Ratio (C/No)	88.9dBHz
Modulation Process	
System Desired Data Rate	1200 bps
Demodulation Method Selected	AFSK
System Allowed or Specified Bit-Error Rate	1.00E-04
Demodulator Implementation Loss	2 dB
Link Performance	
Required Eb/No	56.1 dB
Threshold Eb/No	23.2 dB
System Link Margin	**32.9 dB**

Table 5.5: Downlink Budget

Information on the System	
Transmitter - SnapSat	**Receiver - Ground Station**
Orbit Altitude âĂŞ 350km	Elevation âĂŞ 30 degrees
Slant Range âĂŞ 652.5km	Weather âĂŞ Clear Sky
Demodulation Method âĂŞ BPSK	Cable Length âĂŞ 20m
Antenna Type (TX) âĂŞ Dipole	Antenna Type âĂŞ Cross Yagi
Transmitter System (SnapSat)	
Satellite Transmitter Power Output	0.5 W
	-3.01 dBW
Satellite Total Transmission Line Losses	0.5 dB
Satellite Antenna Gain	2.7dBi
Satellite ERIP	-0.81 dBW
Down Link Path	
Free-Space Path Loss	132 dB
Satellite Antenna Pointing Loss (10 Âř)	10.6 dB
Ground Station Antenna Pointing Loss (10Âř)	2.7 dB
Ground Station Transmission Line Losses	1.8 dB
Atmospheric Loss (30Âř)	0.4 dB
Ionspheric Loss	0.8 dB
Rain Loss	0 dB
Total Loss	146.4 dB
Receiver System (Ground Station)	
Antenna Gain	14.4 dBi
Effective Noise Temperature at Sydney (350K/Day)	610.1 K
Figure of Merrit (G/Ta)	13.5 dB/K
Carrier to Thermal Noise Ratio (C/T)	-160.71 dB
Boltzmann's constant (K)	-228.6 dBW/K/Hz
Carrier to Noise Density Ratio (C/No)	67.89 dBHz
Modulation Process	
System Desired Data Rate	9600 bps
Demodulation Method Selected	BPSK
System Allowed or Specified Bit-Error Rate	1.00E-04
Demodulator Implementation Loss	2 dB
Link Performance	
Required Eb/No	26.07 dB
Threshold Eb/No	10.5 dB
System Link Margin	**15.56 dB**

5.5 Data Budget

Data budget CALCULATIONS.

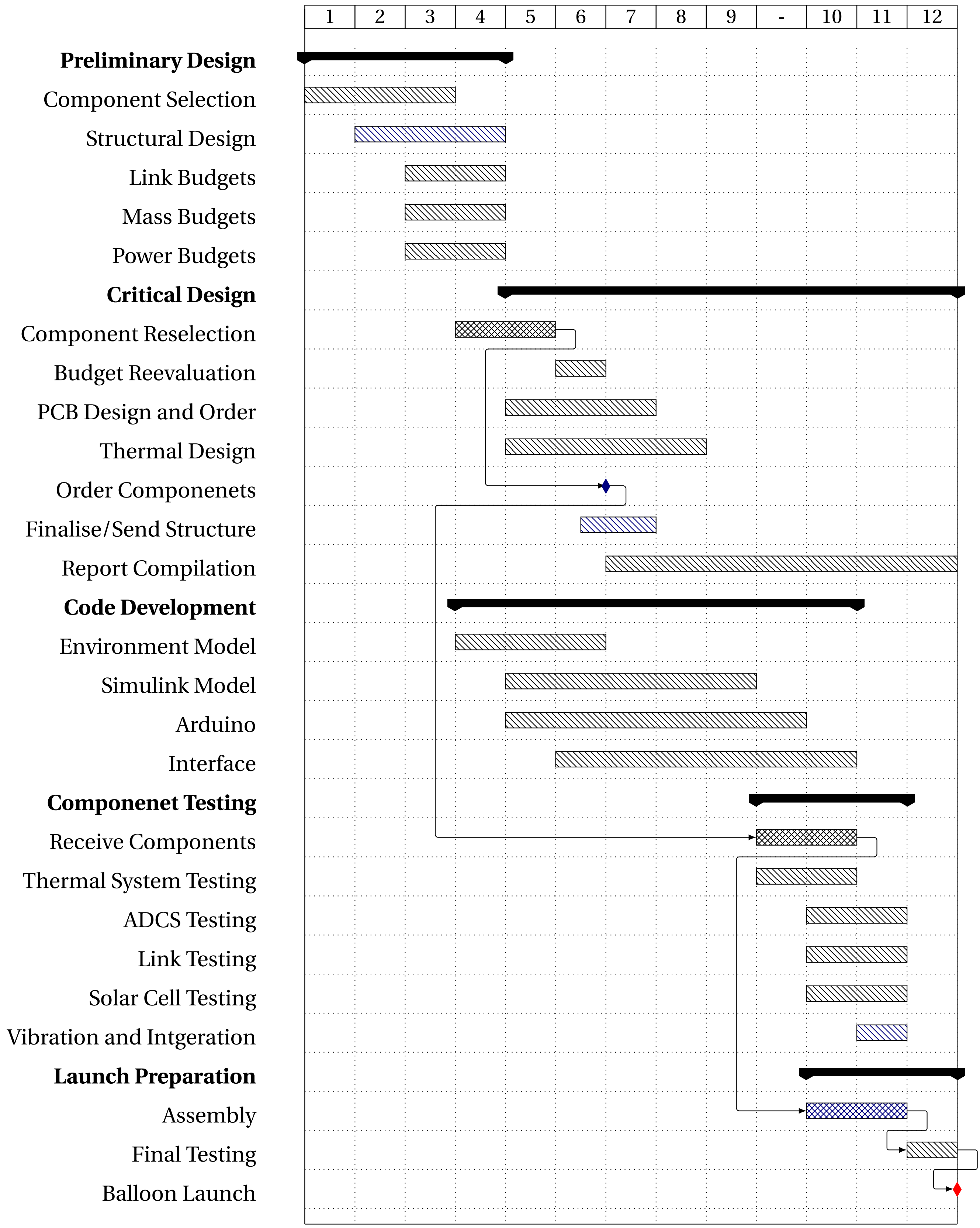
6 Project Plans and Schedule

A general schedule for the SnapSat project is outlined below. A Gantt chart is provided on the following page.

Table 6.1: SnapSat Project Schedule

Major Task	Responsibility	Start Date	End Date
Structural Development	Oscar McNulty	03/08/2015	21/08/2015
Preliminary Design Report	Nikita Sardesai	03/08/2015	23/08/2015
Component Selection	All members	03/10/2015	17/10/2015
Link Budgets	Thomas Forbutt	10/08/2015	23/08/2015
Mass Budgets	Nikita Sardesai	10/08/2015	23/08/2015
Power Budgets	Penelope Player	10/08/2015	23/08/2015
Pointing Budget	James Allworth	10/08/2015	23/08/2015
Component Reselection	All members	21/08/2015	28/08/2015
PCB Design	Penelope Player	23/08/2015	11/09/2015
Thermal Design	Nikita Sardesai	23/08/2015	18/09/2015
Code Development	All members	16/08/2015	09/09/2015
Environment Model	Nikita Sardesai	17/08/2015	04/09/2015
Simulink Model	James Allworth	24/08/2015	25/09/2015
Arduino Code	Thomas Forbutt	24/08/2015	02/10/2015
Interface	Penelope Player	07/09/2015	09/09/2015
Report Compilation	All members	07/09/2015	23/10/2015
Component Testing	All members	25/09/2015	16/10/2015
Thermal Systems Testing	Nikita Sardesai	28/09/2015	09/10/2015
ADCS Systems Testing	James Allworth	05/09/2015	16/10/2015
Link Systems Testing	Thomas Forbutt	05/09/2015	16/10/2015
Solar Cell Testing	Oscar McNulty	05/09/2015	16/10/2015
Final Assembly	Oscar McNulty	05/10/2015	16/10/2015
Vibration & Integration Testing	Penelope Player	12/10/2015	16/10/2015
Final Testing	All members	16/10/2015	23/10/2015
Balloon Launch	Available members	24/10/2015	25/10/2015

6.1 Gantt Chart



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A Appendix: Supplementary Calculations

A.1 Mass Budget Calculations

The component masses were used to determine the center of gravity for the satellite, it is desirable to keep this located close to the geometric centre as the attitude control system (magnetotorquers) are placed on the outermost surfaces.Centroid averaging across all three axes was used to calculate this as follows

$$x_{cg} = \frac{\Sigma x_i \cdot m_i}{\Sigma m_i} \qquad y_{cg} = \frac{\Sigma y_i \cdot m_i}{\Sigma m_i} \qquad z_{cg} = \frac{\Sigma z_i \cdot m_i}{\Sigma m_i}$$

The moment of inertia about all three axes is given by

$$I = \int r^2 \, dm$$

The inertial of each individual component was ignored, assuming these were roughly symmetrical. Only the relative locations of the component contributed to the total inertia. Thus, the inertias were given

$$I_{xx} = \frac{1}{\Sigma m_i} \cdot \left(\Sigma \sqrt{(y_i - y_{cg})^2 + (z_i - z_{cg})^2} \cdot m_i \right) \tag{A.1}$$

$$I_{yy} = \frac{1}{\Sigma m_i} \cdot \left(\Sigma \sqrt{(x_i - x_{cg})^2 + (z_i - z_{cg})^2} \cdot m_i \right) \tag{A.2}$$

$$I_{zz} = \frac{1}{\Sigma m_i} \cdot \left(\Sigma \sqrt{(x_i - x_{cg})^2 + (y_i - y_{cg})^2} \cdot m_i \right) \tag{A.3}$$

and so on; where x_i, y_i and z_i are the positions of each component and m_i is the mass of each component.

A.2 Mapping Calculation Considerations

Calculations are based on an orbit of 300km altitude. In general if we are targeting cities, metropolitan areas for a number of major cities are approximately 100 km^2. Thus assume a target size of 10km by 10km. For an orbit path that goes directly over the city:

$$\alpha = 2 \times \tan^{-1}\left(\frac{5}{300}\right) = 1.9^\circ$$

This is a very narrow window however if we instead photograph an area of 100km by 100km, or 10,000km^2 (roughly the area of Sydney), this equation changes to:

$$\alpha = 2 \times \tan^{-1}\left(\frac{50}{300}\right) = 18.9^\circ$$

Thus if the satellite is misaligned by as much as 8.5°in any direction it will still capture the original 10km by 10km area that was intended. However this is a much larger area and as such there will be less focus on the intended target. Thus the third option is a 50km by 50km picture:

$$\alpha = 2 \times \tan^{-1}\left(\frac{25}{300}\right) = 9.5^{\circ}$$

This image would provide greater focus but would require the attitude of the satellite to be within 3.8°accuracy of the measured attitude. It must be noted that this calculation is for an orbit where the satellite will pass directly over the target area. If this does not occur it will require the satellite to be more accurately aligned due to the fact that it is aiming at a comparatively smaller target. However at small angles this effect is not that significant and since the idea of the mission is to capture cities only when the satellite passes over them it should not be a major issue or consideration.

A.3 Magnetorquer Calculations

Although there is significant data to support the fact that a 0.05Am2 magnetorquer will be powerful enough to control a satellite in space I did some simplistic calculations to check it on an order of magnitude basis. Calculation of the magnetic dipole of the magnetorquer:

$$M = NiA = 312 \times 0.025 \times 0.064 \;=\; 0.05\ \mathrm{Am}^2$$

Calculation of the minimum earthâĂŹs magnetic field at 300km:

$$B = \frac{\mu_0 m_e}{4\pi R^3} \;=\; 2.68 \times 10^{-6}$$

Calculation of torque:

$$T = M \times B \;=\; 1.34 \times 10^{-6}$$

Given the satellite will weigh 1kg and its centre of mass is at the structural centre of the satellite. Assuming that the magnetorquers are located 1cm away from the edge of the satellite. Thus we use the equation:

$$T = mr^2\alpha$$

so then

$$\begin{aligned} \alpha &= \frac{1.34 \times 10^{-6}}{1 \times 0.05^2} && \text{(A.4)} \\ &= 5.36 \times 10^{-4}\ \mathrm{rad/s^2} && \text{(A.5)} \\ &= 0.0307\ \mathrm{deg/s^2} && \text{(A.6)} \end{aligned}$$

Design requirements for the system are that it can recover from a 10 deg/s spin within two days. Assuming average acceleration and that the correct axis is perpendicular to the earths magnetic field.

$$
\begin{aligned}
t &= \frac{\omega}{\alpha} && \text{(A.7)} \\
&= \frac{10}{0.0307} && \text{(A.8)} \\
&= 326 \text{ seconds} && \text{(A.9)} \\
&= 5.5 \text{ minutes} && \text{(A.10)}
\end{aligned}
$$

This is obviously an oversimplification and there are a number of other factors involved which will cause this number to increase. However it is clear from these calculations that the system will have the power to recover form a 10 deg/s spin within the two-day limit.