SCHOOL OF AEROSPACE MECHANICAL AND MECHATRONIC ENGINEERING

AERO3760: SPACE ENGINEERING 2

SnapSat Preliminary Design Report

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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 [?]. <to complete!>

2 Spacecraft Deign Overview

Summarised in table 7.1 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
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. (NOTE: this is only an example for now)

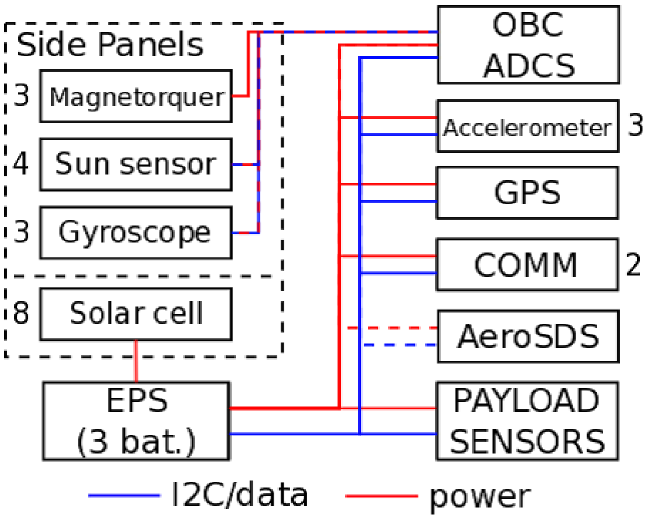


Figure 2.1: Design Schematic

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.

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 table 4.1 below.

Table 4.1: SnapSat Modes of Operation

Spacecraft Mode	**Description**
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.
Safe mode	In this mode, only essential satellite systems are kept ON such as the OBC, power board and VHF receiver. The attitude is not controlled and the transmitter is turned on occasionally for status updates.
Recovery/De-tumble mode	This mode is used to de-tumble the spacecraft after deployment 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.
Establish Contact Mode	In this mode the satellite waits 30 seconds before deploying the antenna and attempting to communicate with the ground station. The ACS system works to orient the satellite correctly.
Payload Operation Mode	This mode is used only when taking a picture. 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. This mode can be triggered by reaching a preset GPS location or manually via communications. This mode can be entered either by reaching a GPS coordinate or through ground control command. It exists this mode straight into Relay Picture Mode.
Relay Picture Mode	This mode is entered after Payload Operation Mode and causes the CubeSat to start sending pictures to the ground station.
Relay Picture Mode	In this mode the camera is almost constantly transmitting images taken through the camera. It exists into Telemetry Mode when the picture has been sent.
Telemetry Mode	In this mode the CubeSat is idle, just displaying basic telemetry. Attitude controlled.
etc. (Other Modes)	

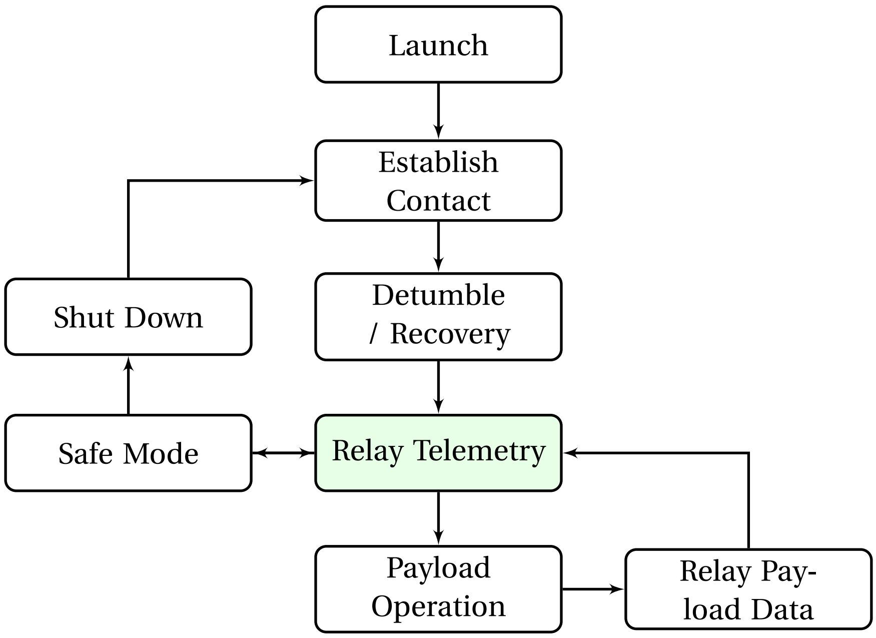


Figure 4.1: Mode Transition Diagram during satellite lifetime

5 Structural Design

The proposed one-unit satellite structure will be 3D printed using ABS plastic. This method was chosen in preference of an aluminium chassis primarily because of the ease of access of production.

6 System Budgets

This section detail the power and mass budgets of SnapSat. (overview/description)

6.1 Mass Budget

The mass budget is shown in table 6.1, it is ensured that *SnapSat* meets the requirement of a maximum weight of 1kg. 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 (magneto-torquers) 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

$$I_{xx} = \frac{1}{\Sigma m_i} \cdot \left(\Sigma \sqrt{(y_i - y_{cg})^2 + (z_i - z_{cg})^2} \cdot m_i \right) \tag{6.1}$$

$$I_{yy} = \frac{1}{\Sigma m_i} \cdot \left(\Sigma \sqrt{(x_i - x_{cg})^2 + (z_i - z_{cg})^2} \cdot m_i \right) \tag{6.2}$$

$$I_{zz} = \frac{1}{\Sigma m_i} \cdot \left(\Sigma \sqrt{(x_i - x_{cg})^2 + (y_i - y_{cg})^2} \cdot m_i \right) \tag{6.3}$$

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. The inertial matrix was computed using *Solidworks*. It was found to be:

$$I = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix} = \tag{6.4}$$

Table 6.1: SnapSat Mass Budget (target mass of 1000 g)

Subsystem	**Mass**	**Contingency**	**Mass + Contingency**	**Fraction of Total Mass**
Structural - chassis - solar panels	 140g 9 × 25g	 25g 9 × 5g	435g	
ADCS - air core coils - sun sensors - IMU	 3 × 50g 6 × 1.8g 2.8g	 3 × 5g 6 × 0.3g 0.4g	180.2g	
EPS - batteries - power bus				
OBS / OBDH - Arduino board - memory storage	 25g 1g	 5g 0.2g	31.2g	
TT&C - antennae				
Thermal - tapings - MLI	 1g 3g	 0.3g 0.5g	4.8g	
Payload - camera	 25g	 5g	25g	
Integration - bolts/rivets - cabling/wires	 5g 2g	 2g 0.5g	9.5g	
Total				
Mass Margin				

6.2 Power Budget

Table 6.2: SnapSat Power Budget

			Average Duty Cycle by Mode (%)				
Load	**Power Consumption (W)**	**Number of Units On**	*Safe Mode*	*Recovery Mode*	*Payload Mode*	*Other Mode*	
OBC							
VHF Rx							
S-band Tx							
Magnotorquers	150mW	1	0 %	on			
Power Board							
Camera	1950mW	1	0	0	100 %		
IMU	30mW	1	off	on	on	on	
GPS	100mW	1	off				
Solar Panels	935mW	5	on	on	on		
Sum Loads (W)							
Efficiency							
Power Consumed (W)							
Power Generated (W)							
Power Margin							

6.3 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 lost cost budget a 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.

6.3.1 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:

$$\begin{aligned} \text{System Error} &= \sqrt{(\text{IMU error})^2 + (\text{sun sensor error})^2 + (\text{magnetorquers error})^2} \quad (6.5) \\ &= \sqrt{(2\%)^2 + (0.5\%)^2 + (1\%)^2} \quad (6.6) \\ &= 2.3\% \quad (6.7) \\ &= 8.3^\circ \quad (6.8) \end{aligned}$$

This is summarised in the table below

Table 6.3: 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.1) 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.

6.4 Link Budgets

Calculations for both link budgets (list assumptions here).

6.4.1 Uplink Budget

The uplink budget allows for XXX. The specifications are

- Antenna type at satellite: (omni, directional+gain)
- Frequency Band: (VHF (145.800MHz) , UHF (435.xxx MHz), SHF etc.)
- Objective C/N:
- Bit rate and modulation type:
- Expected occupied bandwidth:

6.4.2 Downlink Budget

The downlink budget allows for XXX. The specifications are

- Antenna type at satellite: (omni, directional+gain)
- Frequency Band: (VHF (145.800MHz) , UHF (435.xxx MHz), SHF etc.)
- Objective C/N:
- Bit rate and modulation type:
- Expected occupied bandwidth:

6.5 Data Budget

Data budget CALCULATIONS.

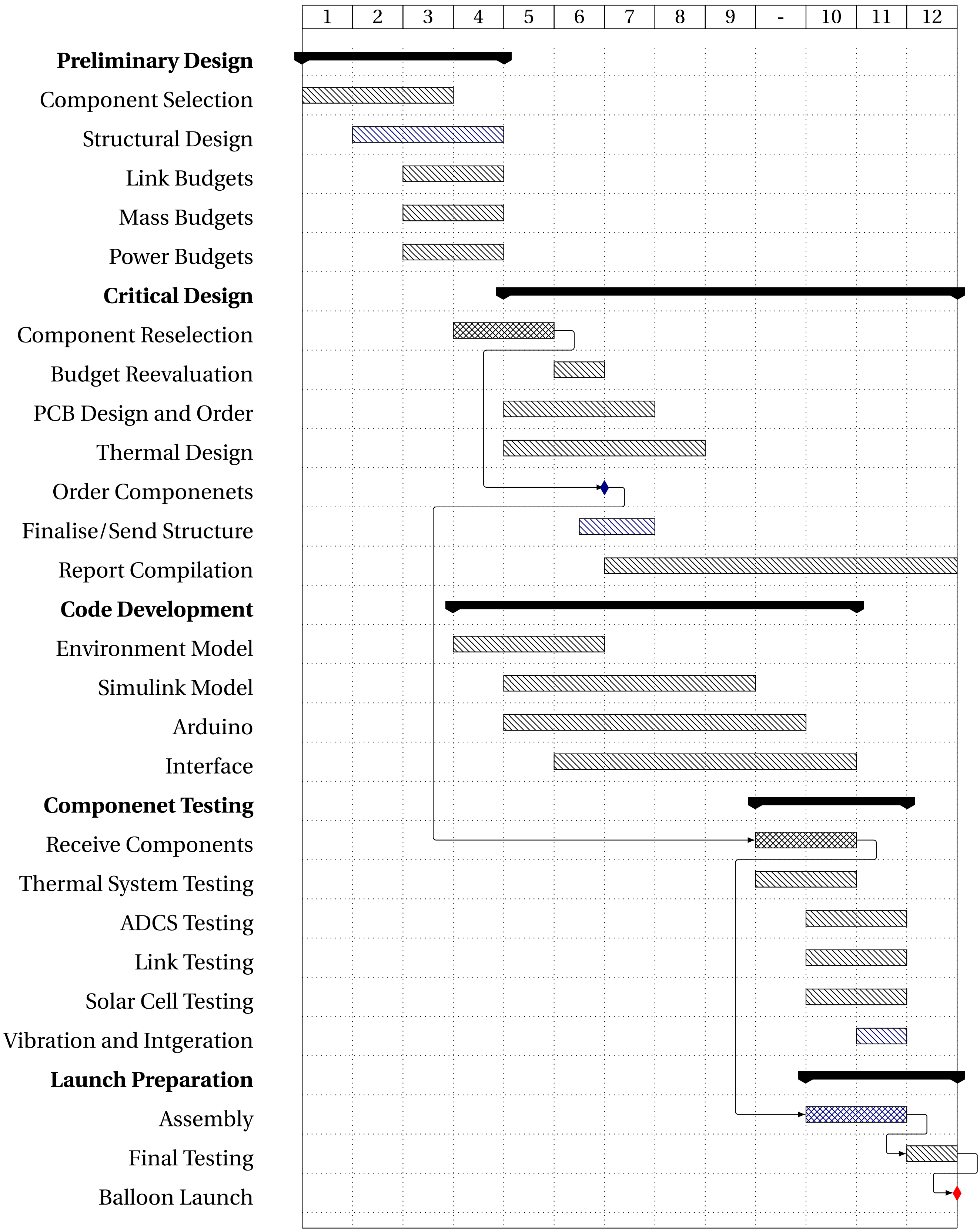
7 Project Plans and Schedule

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

Table 7.1: SnapSat Project Schedule

Major Task	Responsibility	Start Date	End Date
Structural Development	Oscar McNulty		

7.1 Gantt Chart



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A Appendix: Supplementary Calculations

A.1 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 km2. 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.2 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:

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

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

$$\begin{aligned} \alpha &= \frac{1.34\times10^{-6}}{1\times0.05^2} && \text{(A.1)}\\ &= 5.36\times10^{-4}\ \mathrm{rad/s^2} && \text{(A.2)}\\ &= 0.0307\ \mathrm{deg/s^2} && \text{(A.3)} \end{aligned}$$

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.

$$\begin{aligned} t &= \frac{\omega}{\alpha} && \text{(A.4)}\\ &= \frac{10}{0.0307} && \text{(A.5)}\\ &= 326\ \text{seconds} && \text{(A.6)}\\ &= 5.5\ \text{minutes} && \text{(A.7)} \end{aligned}$$

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.