The Australian National University
U5162641

# ENGN 2225

Research Portfolio
Version 1.0

## Design of a Platform for Ionosphere Electron Density Measurements and Radio Astronomy Calibration

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**Abstract**

The client has asked for a space based platform designed to allow measurements of the electron density in the ionosphere, for the purpose of calibrating radio astronomical measurements at as low cost as possible. The final design was found to be a 1U sized CubeSat class satellite (10 by 10 by 10 cm cube) with two deployable radio transmitters, orbiting at an altitude of 435 km and an inclination of 45°. The satellite will be able to operate for over a year making at least 4 fly overs of the primary Australian radio telescope per day, with a similar coverage of radio telescope sites in South Africa and North America. The transmitters will alow the electron density and its effect on radio waves to be measured simultaneously at two separate frequencies between 80 and 400 MHz.

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## 1 Introduction

Radio astronomy is the field of astronomy which looks at signals in the radio wave section of the electromagnetic spectrum. A local example of a radio astronomy site is the Murchinson Widefield Array (MWA), which is a low frequency telescope operating between 80 and 300 MHz. The MWA is located in Western Australia and is one of three precursor sights for the Square Kilometer Array [1]. When passing through the ionosphere, a portion of the atmosphere between 75 and 1000 km [2], radio waves undergo a phenomenon known as Faraday Rotation [3–7]. Faraday Rotation is directly related to the free electron density and causes a rotation in the polarization plane of electromagnetic waves [3]. This rotation has a detrimental effect on ground based telescopes picking up radio waves in the same way that atmospheric turbulence impairs optical telescopes [4, 6]. While the ionosphere extends to 1000 km in altitude, studies have shown that the bulk of the electron density is below 400 km [5, 6]. A mission first proposed by Frank Briggs [8] aims to measure the electron density in the ionosphere. The electron density varies both daily and yearly so the mission must have multiple measurements over a single day and continue for at least a year. The seasonal variations in the electron density are more pronounced in the southern hemisphere [9] making this a bigger issue for radio telescope arrays in Australia, like the MWA, than their equivalent in the northern hemisphere. In addition to using these measurements to better understand the structure of the ionosphere, the measurements will be used in 'real time' to calibrate radio astronomical measurements taken at the MWA in Western Australia. Ideally, the flight path would allow the craft to service other radio astronomy arrays such as the Low Wavelength Array in New Mexico, USA and the Square Kilometer Array site in South Africa to open channels for collaboration and extra funding. This satellite is aimed to be constructed by the Advanced Instrumentation and Technology Centre (AITC) here in Canberra.

## 2 Solution

The best platform for measuring the ionosphere electron density and calibrating radio astronomy measurements was determined to be a 1 unit (1U) sized CubeSat, a small satellite $10 \times 10 \times 10$ cm, orbiting at an altitude of 435 km with an orbital inclination of 45°. (Inclination is the angle at which the orbital path crosses the equator.) This satellite will be able to service the primary site, MWA, over 4 times a day as well as other major radio astronomy facilities in South Africa and North America, opening up potential avenues of collaboration.

The satellite will be able to send two radio signals of different frequency between 80 and 400 MHz simultaneously for measurement on the ground. The lifetime of all systems is well over the target of one year and an Australian manufactured end of life system has been selected to remove the satellite from orbit at the end of its life so as not to add to the proliferation of space junk. The CubeSat will be stabilized by utilising a combination of passive gravity gradients and a magnetorquer, and will be powered by standard CubeSat solar panels and an onboard battery. All of the above systems will be controlled by an on board central processing unit (CPU).

## 3 Problem Scoping

The scoping of this problem involved three main steps: identification of design requirements from the science mission objectives listed below; a system boundary analysis to properly identify what affects the system, and which of those factors can be controlled; and a use case chart showing the ideal operation of the spacecraft.

- ◇ Main Objectives
  - Measure the electron density in the ionosphere via Faraday Rotation on a daily basis above the Murchinson Widefield Array (MWA).
  - Mission life of at least a year.
  - Measure using multiple frequencies between 80 and 300 MHz.
  - Cover enough of the ionosphere, altitude wise, to get meaningful results.
  - Low cost mission.
- ◇ Secondary Objectives
  - Have the capability to calibrate and service other radio astronomy sites.

### 3.1 Design Requirements

The science objectives were translated to design requirements, see Table 1, and ranked using a pairwise analysis, see Appendix (section 10). The rankings listed have several double ups. This is because the design requirement is repeated in multiple customer requirements or, in the case of the lifetime requirements, because they all have equal weighting. The lifetime of the satellite is limited to the lifetime of the shortest-lived component; as such, all of the component lifetimes will need to be above the one year benchmark with equal importance. The cost requirement has come out on top, as the only way this mission will fly is if it is relatively cheap (for a space mission). This mission will be used as a stepping stone to enable the Advanced Instrumentation and Technology Centre (AITC) to gain the funding and reputation for building larger, more extensive space missions in the future. Many of the lower ranked requirements such as mission lifetime, time over primary site, and number of radio transmitters, have benchmarks that MUST be met, but improvements after the benchmark is met are of little to no extra value - hence the low ranking in priority. The lowest ranked requirement is the time over alternate sites, which corresponds to the secondary objective. This ranking aligns with the client's views as it was made evident that the ability to service secondary sights was of a low priority. The ability to measure multiple frequencies is directly related to the number of radio transmitters on the satellite as each transmitter can only send one frequency. Hence, a high number of radio transmitters is added as a design requirement. Spacecraft are made up of two components, namely the bus and the payload. The payload is the all of the components or cargo which is aimed at completing the specific mission. The bus unit is all of the other components which alow the spacecraft to survive in orbit and function.

Table 1: Translation of Customer/Science requirements into design requirements.

| Customer Requirement | Design Requirement | Metric (TPM) | Rank |
|---|---|---|---|
| | Frequency of orbits over MWA | + # per day | 4 |
| Measure electron density | High orbital altitude | + km | 5 |
| | Long time over primary site | + minutes per orbit | 8 |
| | High stability | − error in ° | 3 |
| | Long orbit decay time | + years | 9 |
| Mission lifetime over one year | Long power supply lifetime | + years | 9 |
| | Long bus unit lifetime | + years | 9 |
| | Long stability system lifetime | + years | 9 |
| Measure multiple frequencies | High number of transmitters | + # | 7 |
| | Surplus power available | + kW/h | 6 |
| Cover enough ionosphere | High orbital altitude | + km | 5 |
| Low Cost | Low launch cost | − $ AUD | 1 |
| | Low component cost | − $ AUD | 2 |
| Service other sites | High time over alternate sites | + minutes per orbit | 10 |
| | Surplus power available | + kW/h | 6 |

### 3.2 System Boundary

The system boundary chart shown in Table 2 shows different factors of the system and sorts them based on whether they affect the system or not, and can or cannot be controlled. A factor which will affect the system, but was not included in the systems engineering design analysis, is the Government Licensing and Fees involved in getting any craft launched into space. While these fees will change the total cost of the mission, which is the highest ranked design requirement, the fees are the same for all space craft and thus contribute a systematic change to any proposed system. Hence their omission is not biasing the results towards or away from any one design. The other factor which has been excluded from the system analysis is the design of the launch itself. The standard cost of launch for each particular type of satellite was considered but the details of the launch were not analysed. The orbit of

Table 2: System Boundary Chart

| **Endogenous** | **Exogenous** | **Excluded** |
|---|---|---|
| Bus Unit | ISS Orbit | Launch |
| Communications | Government Regulations | Government Licensing and Fees |
| Power System | Space Debris | |
| Control System | | |
| Stability and Propulsion System | | |
| De-Orbiting System | | |
| Orbit | | |

the International Space Station (ISS) has been included as an Exogenous factor because small satellites, specifically CubeSats, are able to purchase space on a transport to the ISS and be launched from there. This option saves a large amount of money on the launch cost, but will fix the altitude of the orbit.

### 3.3 Use Case

| **Table 3** | **Use Case** |
|---|---|
| *Scope:* | System-wide |
| *Level:* | User-goal |
| *Primary Actor:* | End-User |
| *Stakeholders and Interests:* | • Radio Astronomers: End-Users<br>• AITC: System-Builders<br>• Other Space Users: Shared operating environment |
| *Preconditions:* | Successful launch to required orbit |
| *Postconditions:* | None |
| *Main Success Scenario:* | |

1. Craft passes over MWA site 2 - 3 times a day
2. Radio waves transmitted to MWA
3. Radio wave Faraday Rotation measured at MWA for use in calibration
4. Craft continues to operate over one year lifetime
5. Craft de-orbits at end of life time

*Extensions:*
2.a Critical failure:

1. Craft de-orbits without colliding with other space objects

*Frequency of Occurrence:* Once

# 4 Requirements Analysis

## 4.1 Existing Solutions and Customer Requirements

There are currently two main methods for measuring the electron density in the ionosphere: measuring Faraday Rotation of electromagnetic waves and direct measurements with physical probes. Physical probes, specifically Laugmuir probes, have been used to measure the electron density in studies concerned with the plasma characteristics and interactions in the atmosphere [10] and also have been attached to rockets and sent through the entire atmosphere [11]. Measurements of Faraday Rotation have been done on very low frequencies (1 - 7 MHz) by looking at partial reflections of waves propagated from the surface [12] and waves reflected off the moon [3]. Faraday Rotations of high frequency waves (above 1.2 GHz [7]) have been done using GPS satellites [5, 7, 13], and in more recent years using low frequency band (below 1GHz) transmitters specifically designed for investigating the ionosphere on science satellites [6]. Unfortunately, none of these current technologies fit the customer requirements. The only existing technology that can meet the science objectives is equipment on large, expensive science satellites built by national agencies like the National Aeronautics and Space Administration (NASA) and the European Space Agency (ESA). This kind of satellite is for too large, complicated and expensive for what the client wants. As such, the goal of this design project is to take the technology in the large science satellites needed for the ionosphere electron density measurements and assemble it in a smaller, cheaper and shorter-lived satellite.

## 4.2 House of Quality

The customer requirements are related to the design requirements, and the design requirements to each other in the the house of quality (HoQ), Table 4. Care was taken in defining the design requirements into measurable engineering metrics. As such, it was not required to translate these requirements to engineering characteristics in the design analysis. The HoQ highlights that there is a trade off between cost and almost every other other requirement. As cost is the first priority, effort was made to achieve the minimum science requirements as cost effectively as possible. This strategy will limit the potential other uses of the satellite after its primary lifetime; see Life Cycle Analysis section for further discussion. There is a strong negative correlation between the orbital altitude and the launch cost. As such, the orbit selected is at the height of the ISS to reduce the launch cost as much as possible through both low altitude and greater launch opportunities.

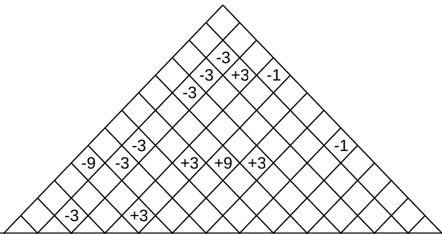
The requirements map shows how the customer requirements are related to the individual subsystems, allowing the location of repercussions from future customer requirement changes to be seen. The requirements map is discussed further in the Subsystem Integration section.

Table 4: House of Quality

| Correlations | |
|---|---|
| Positive | + |
| Negative | − |
| No Correlation | |

| Relationships | |
|---|---|
| Strong | 9 |
| Moderate | 3 |
| Weak | 1 |

| rection of Improvem | |
|---|---|
| Maximize | ▲ |
| Target | ◇ |
| Minimize | ▼ |




| Relative Importance | Column # | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | Direction of Improvement | ▼ | ▼ | ▼ | ▲ | ▲ | ▲ | ▼ | ▲ | ▲ | ▲ | ▲ | ▲ | ▲ |
| | Design Requirements / Customer Requirements | Low Launch Cost | Low Component Cost | High Stability | Frequency of Orbits Over MWA | High Orbital Altitude | Surplus Power Available | High Number of Transmiters | Long Time Over Primary Site | Long Orbital Decay Time | Long Power Supply Life Time | Long BUS Unit Lifetime | Long Stability System Lifetime | Long Time Over Alternate Sites |
| | Rank | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 9 | 9 | 9 | 10 |
| 1 | Low Cost | 9 | 9 | 3 | 1 | 9 | 3 | 3 | 1 | 3 | 1 | 1 | 1 | |
| 2 | Measure Electron Density | | | 9 | 1 | | 3 | 3 | 3 | | | | | |
| 3 | Cover Most of Ionosphere | 3 | | | | 9 | | | | | | | | |
| 4 | Measure Multiple Frequencies | | 3 | | 1 | | 3 | 9 | 1 | | | | | |
| 5 | Mission Lifetime Over One Year | | 3 | | | | | | | 9 | 9 | 9 | 9 | |
| 6 | Servise Other Sites | | | | | | | | | | | | | 9 |
| | Technical Performance Metric | $ AUD | $ AUD | °error | N / orbit | km | kW/h | N | min / orbit | years | years | years | years | min / orbit |

## 5 Logic and Function

### 5.1 Concept Generation

All of the potential solutions brainstormed were listed in a concept generation tree shown in figure 1. The majority of these concepts have already been discussed in the Existing Solutions section. The potential solutions on the concept generation tree that have not been ruled out by the customer requirements are the different CubeSat sizes. As the customer is after a cheap satellite, the smallest size, 1U, was selected. The launch cost of a CubeSat is directly related to the size, mass and orbital altitude [14]. The Appendix gives a table listing estimated launch cost for satellites of varying size, mass and altitude.

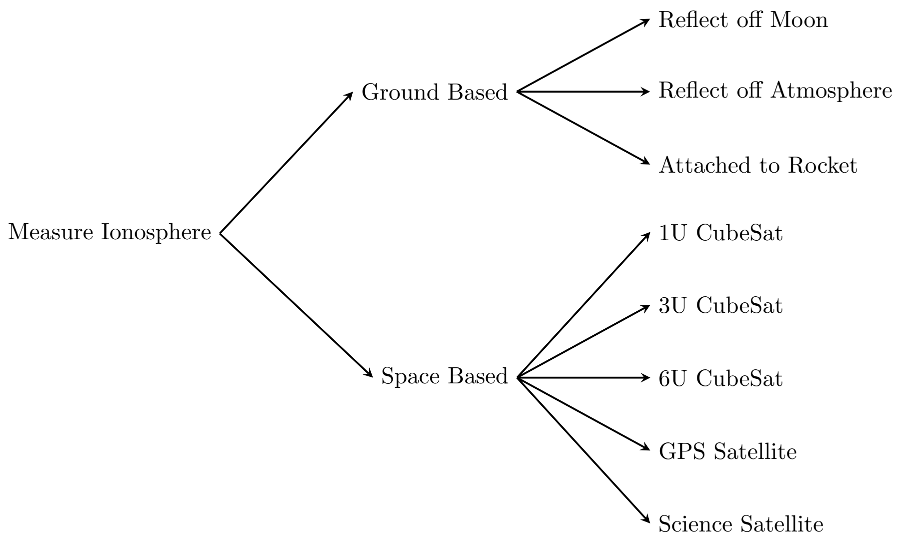



Figure 1: Concept Generation Tree

### 5.2 Logical Flow

The logical flow, figure 2, describes the main steps and decisions that are needed for the successful operation of the satellite. It should be noted that the satellite is required to perform these actions and make these decisions complectly autonomously. The large number of necessary autonomous decisions identified that the final system will need an extensive control system. This requirement is common in most space based systems.

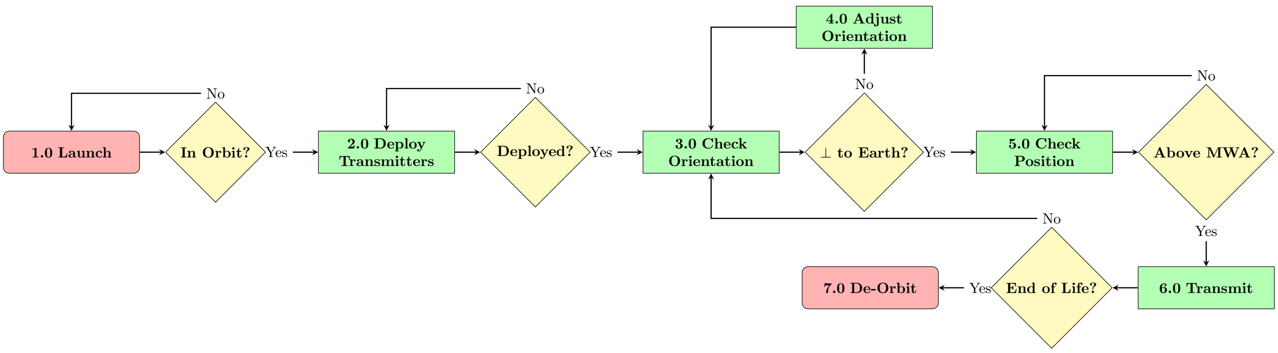



Figure 2: Logical Flow

### 5.3 Functional Flow Block Diagram

The tasks identified in the logical flow, figure 2, are further developed into the functional flow block diagram (FFBD) shown in figure 3. As the details of the launch itself are out of the scope of this design analysis, the FFBD does not include a functional breakdown of the launch task. Via analysis of several FFBD's, it was decided that the best method for transmitting the signal at the correct time was to have a signal from the MWA, which when received triggers the satellite to transmit. The alternate options explored were having the satellite with sensors capable of detecting where it was (GPS) and automatically starting the transmission when in the correct place, or having the satellite continuously broadcasting its radio signal. Of these three potential methods continuous broadcasting was the simplest with the least amount of steps involved in its FFBD (not shown). However, continuous transmission will use up the most power and could interfere with other space or ground based operations along the satellite's flight path. Having the satellite with the ability to know where it is and start transmitting on its own was the most complex system identified, and would use a considerable amount of power. The advantage of having the transmission started by a signal from the ground station, aside from being less complex and more power efficient, is that the satellite functionality could then easily be expanded to service ground sites other than the MWA along its flight path. All that would be required is for the secondary ground site to transmit the 'on' signal and the satellite would broadcast when within range of the signal.

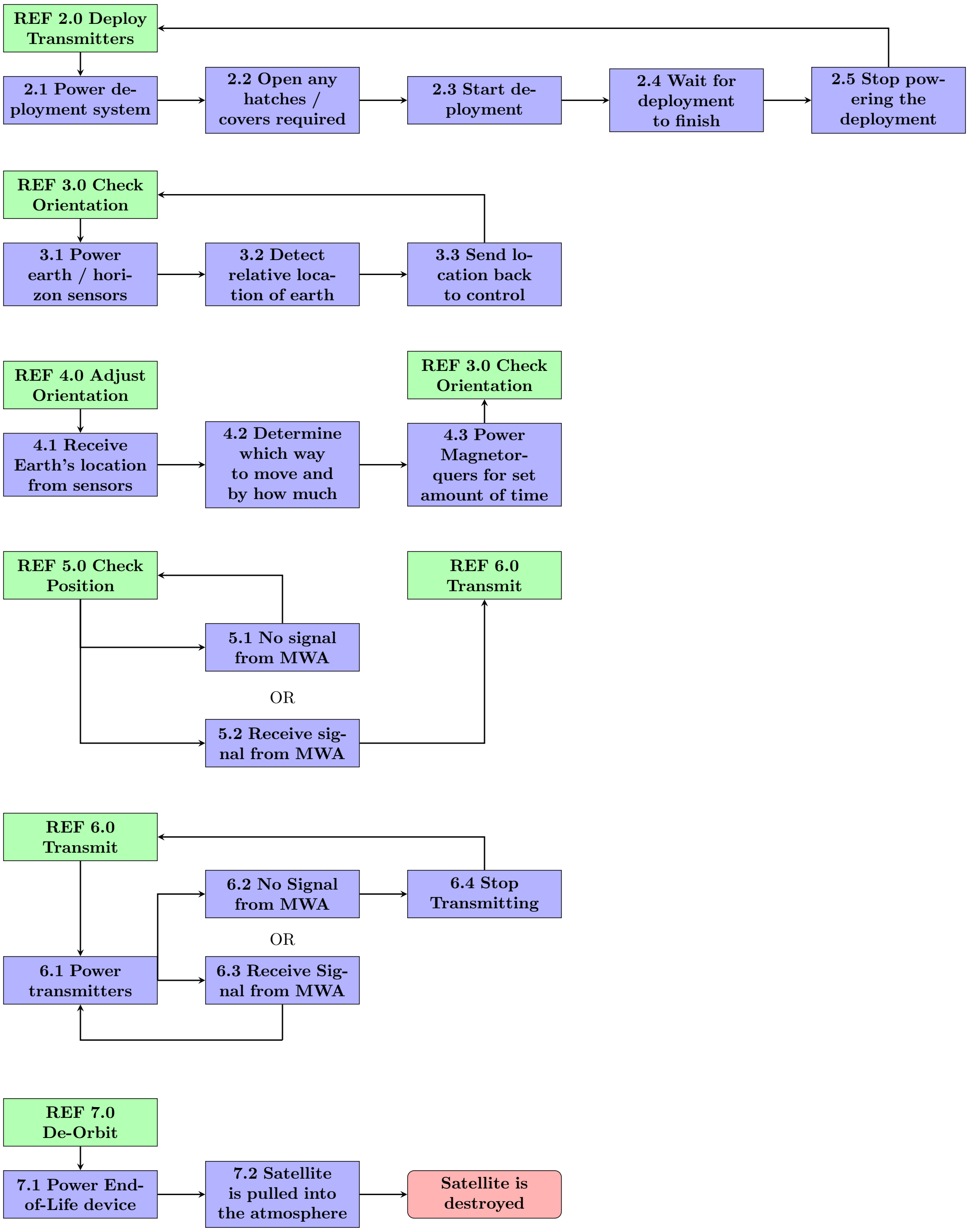



Figure 3: Functional flow block diagram

# 6 Subsystem Integration

## 6.1 System Interface

The subsystems included in the CubeSat are all standard for a space satellite mission. There is no individual subsystem dedicated to the science measurements as this will be performed by the communications subsystem by transmitting radio waves down to Earth where the Faraday Rotation will be measured. It can be seen clearly from the subsystem interface, figure 4, that the control and power subsystems are critical to all other subsystems. As such, care must be taken to make the components of these subsystems reliable. The operation of most of the subsystems relies on the system interacting with the surrounding environment in orbit, whether it be via radiation or magnetic fields. The possibility of the environment and subsystems interacting in an undesired way via radiation and / or magnetic fields was considered. Fortunately, this is a problem common for most space system components and these components almost always have shielding built into and around the critical components.

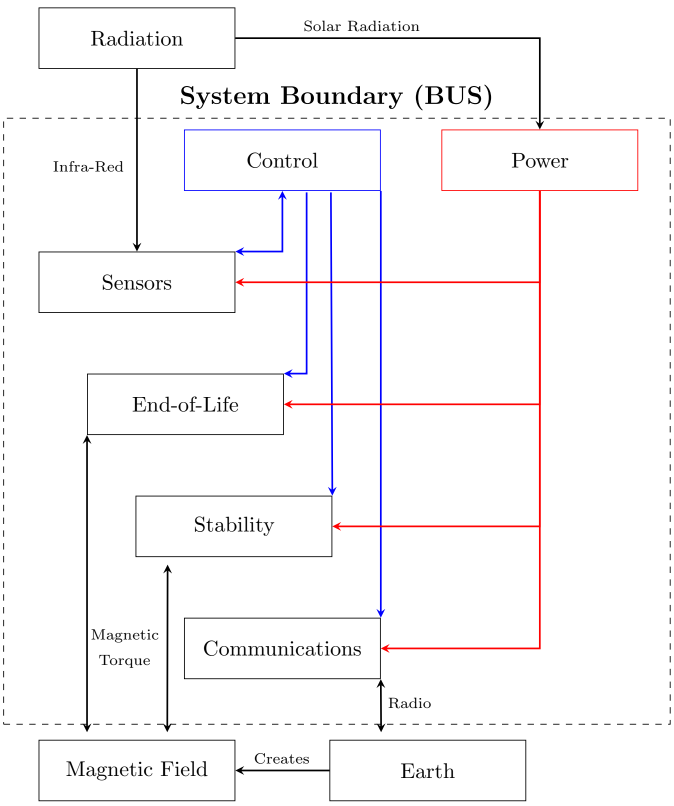



Figure 4: Subsystem Interface. Blue lines represent control signals and red lines represent the transfer of electrical power.

End of life is a subsystem in itself as it has distinct and separate functions and components from all other subsystems. The component for this subsystem was selected to be the DradEN produced by Australian company Saber Astronautics. DragEN is a tether that when deployed falls down towards Earth via gravitational gradients and interacts with the Earth's magnetic field to slow the satellite until it drops into the atmosphere and burns up on re-entry.

### 6.2 Functional Allocation and Requirements Mapping

The functions, as numbered in figure 3, were mapped to the subsystems and the required components to complete the functions listed as shown in figure 5. Figure 5 further illustrates that the system is control dominated with few functions being outside of the control (and power) subsystems. The components listed in figure 5 are for the final design. The evaluation to determine what these components should be is discussed in the Evaluation section below. The subsystems were also mapped to the customer requirements, which are shown in Table 4 above. By mapping each customer requirement to the subsystems, the ramifications of any changes to customer requirements in the later stages of development can immediately be seen. What is of most importance is that the low cost requirement is linked to every single subsystem. This connection poses a risk as a change in cost requirement will require a large redesign of the entire system. such consequences are not unexpected and somewhat unavoidable as everything costs money. What is advantageous in the present design is that it is based on the lowest possible cost to achieve the customer requirements, the only direction that cost requirement can go is up and more money becoming available is never a problem!

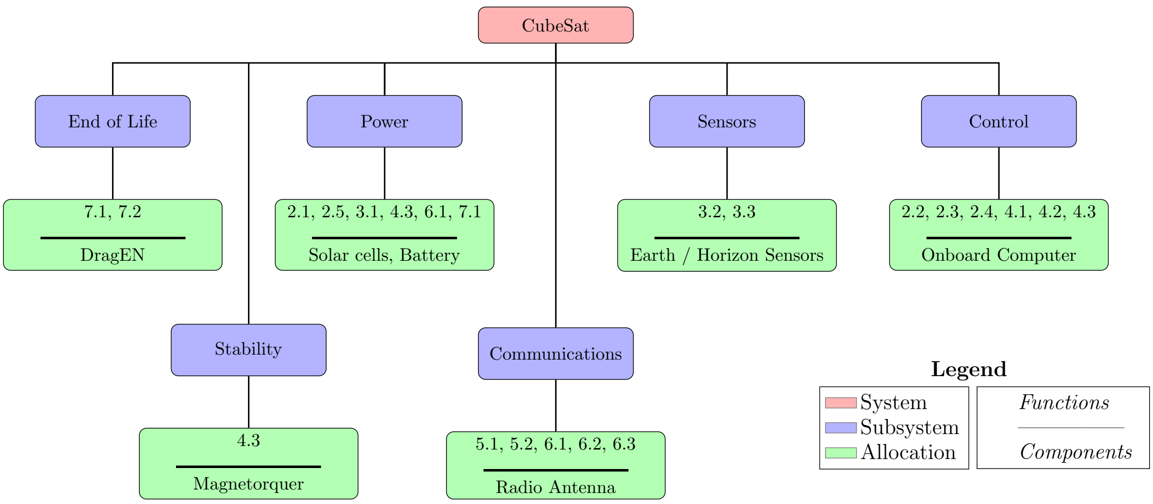



Figure 5: Subsystem functional and component allocation.

## 7 Life-cycle Analysis

### 7.1 Design and Production

The assembly of the CubeSat is modular in nature, which has advantages and drawbacks. The main advantage of a modular system is that it allows last minute design changes to occur and have minimal impact on the overall design of the system; only one of the modular components needs to change. The drawback from a life-cycle perspective is that the design of the CubeSat is made up from off-the-shelf components. As such, a large part of the production occurs by companies out of the scope of the project so the recyclability cannot be analysed.

### 7.2 Operation

During the operational life time of the CubeSat hardware changes are impossible and firmware updates can be difficult. This is not ideal as an ideal operational life-cycle should be able to update to continue its life for longer without becoming redundant. However this situation is a restriction of the environment in which the system has to operate (in orbit) rather than a design flaw and there is no financially feasible or practical way to allow the system to be updated or repaired once launched.

### 7.3 End-of-Life

The end of life for most systems is concerned with recycling in an environmentally friendly way. For satellites this presents itself differently to most systems. Recycling of satellites is mainly in the form of transferring the control of the satellite from its original use to another organisation or purpose which it is capable of. This modulation approach is the plan in place for both the Hubble Space Telescope (HSC) and the International Space Station (ISS). For a small 1U CubeSat there is limited to no alternate uses as the design is streamlined for its primary purpose. The design of the CubeSat will also leave little operational lifetime of the components after the end of its mission. As such, there is no potential to adopt cradle-to-cradle thinking. Disposing of the satellite in an environmentally friendly way involves de-orbiting the satellite in a way that drops its orbit into the atmosphere where it will burn up on re-entry. This approach will prevent the unused satellite from adding to the current problem of space debris. Further, more the DragEN chosen as the end of life tether is rated to deorbit the satellite within 15 days. As the CubeSat for this mission is in lower orbit than is standard, the deorbit time could be even shorter. This rapid de-orbit time is ideal as a safeguard in case of critical failure of the system once in orbit, as it will be able to de-orbit the satellite swiftly with minimal risk to other space operations.

## 8 Testing

Analytical tests on the CubeSat took the form of an orbital analysis using the software Systems Tools Kit. This simulation determined the coverage of the satellite over the primary site and other major radio telescope sites as well as the amount of time in the direct sunlight for power generation. The coverage was used to determine the best altitude and inclination of the orbit. The final orbit selected is at an altitude of 435 km (ISS height) with an inclination of 45°. The coverage of the satellite for a single day is shown in figure 6. The satellite will pass over the

MWA over 4 times a day as well as the other sites with roughly the same frequency. A higher altitude orbit was also examined as more of the ionosphere could be covered (vertically), but the relatively small increase in coverage and solar exposure for power generation was far out weighted by the greater cost of launching the satellite to the higher orbital altitude. The analytical simulations identified that the satellite will be in direct sunlight for just over half of the time. Using the common solar panel efficiency of 30% quoted for various typical solar panels [15, 16], an average power generation whilst in the sunlight was calculated as 2.3 - 5 W. The minimum of this range is not quite enough to power all of the onboard systems and this power will not be generated when the satellite is eclipsed by the Earth. Two options to generate more power were considered: a larger deployable solar panel array, and an onboard battery. The deployable solar panel array complicates the satellite with more moving parts and will still not function when out of direct sunlight, so a standard CubeSat battery was selected to supplement the standard exterior solar panels attached to the sides of the satellite.

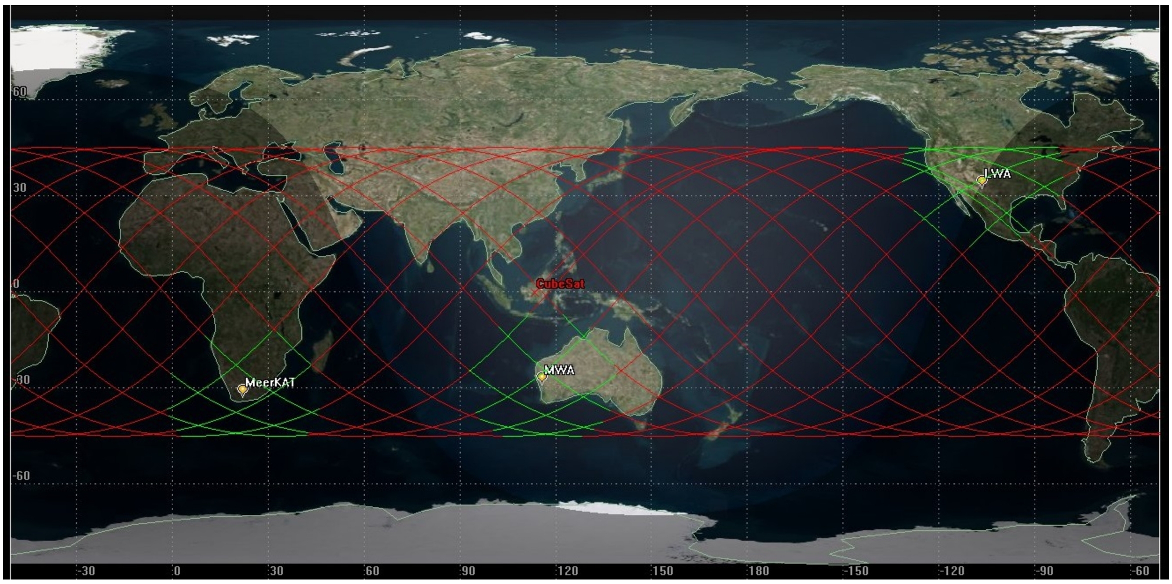



Figure 6: Orbital path of the CubeSat over one day. Green denotes when the satellite has access to a ground radio astronomy station. Red denotes when the satellite has no access and would not be transmitting.

The proof-of-concept tests for the function of each component were conducted by the manufacturer of each component. Other than verifying the specifications given by the manufacturer, there is little more to be done on proof-of-concept testing. The main focus of the testing will need to be on the interface between subsystems and the control subsystem. In this regard there are strict regulations which must be adhered to if the satellite is to gain a license to fly. The regulations are described extensively by Australian Communications and

Media Authority [17]. Further stages of testing are beyond the scope of this design analysis.

## 9 Evaluation

Shown in Table 5 and Table 6 are the data and evaluation matrices for three potential components in the stability system. Only the stability system was analysed in the design analysis presented here as it was deemed the most applicable. The requirements listed in the tables differ slightly from design requirements, but are directly related. With the aim of a 1U CubeSat sized BUS unit the launch cost is only dependent on weight. However, the components must fit into the small space provided: a 10 cm × 10 cm × 10 cm cube. To account for these new factors the requirement of low launch cost (ranked as second) was split into two: size and mass. The only other change to the design requirements defined above is in the surplus power requirement. The surplus power available requirement has been redefined for the purpose of analysing the power usage of individual components for nominal operation.

The three components analysed were passive gravity gradient, magnetorquers, and momentum wheels. A passive gravity gradient system does not actually add any physical components to the system but instead is the process of weighting the satellite such that there is an imbalance in the moments of inertia around each axis such that the craft is biased to keeping one face pointing towards the Earth at all times - hence the zero size, mass, cost and power usage for this aspect in Table 5. A large scale example of this gravitational bias in practice is the moon. The moon is not a perfect sphere and has an imbalance in its moments of inertia giving a bias to one face, which is why only one side of the moon is always pointing towards Earth. The second component considered, the magnetorquer, is a rod with a tightly coiled wire around it. By passing current through this wire a magnetic field is produced which will interact with the Earth's magnetic field and create a torque which can be used to stabilize the satellite. Momentum wheels are a system where wheels inside the CubeSat can be spun up or down via a motor, which will rotate the satellite in orbit via the conservation of angular momentum.

Table 5: Data for potential stabilization components.

| Requirement | Rank | Metric (TPM) | Gravity Gradient | Mangetorquer | Momentum Wheel |
|---|---|---|---|---|---|
| Cost | 6 | − $ AUD | ≈ 0 | $ 846 | $ 43846 |
| Size | 5 | − $cm^3$ | ≈ 0 | 18.85 | 800 |
| Mass | 4 | − g | ≈ 0 | 22 | 1500 |
| Stability | 3 | −° error | Large | ± 3 | ± 0.0005 |
| Power Usage | 2 | − W | ≈ 0 | 0.209 | 28 |
| Lifetime | 1 | + years | Indefinite | 7 | 5 - 7 |

Table 6: Evaluation matrix for potential stabilization components.

| Requirement | Weight | Metric (TPM) | Gravity Gradient Rank | Score | Mangetorquer Rank | Score | Momentum Wheel Rank | Score |
|---|---|---|---|---|---|---|---|---|
| Cost | 6 | − $ AUD | 3 | 18 | 2 | 12 | 1 | 6 |
| Size | 5 | − $cm^3$ | 3 | 15 | 2 | 10 | 1 | 5 |
| Mass | 4 | − kg | 3 | 12 | 2 | 8 | 1 | 4 |
| Stability | 3 | $-^\circ$ error | 1 | 3 | 2 | 6 | 3 | 9 |
| Power Usage | 2 | − W | 3 | 6 | 2 | 4 | 1 | 2 |
| Lifetime | 1 | + years | 3 | 3 | 2 | 2 | 1 | 1 |
| | | Total | | **57** | | 42 | | 27 |

The evaluation matrix for the stability subsystem components, Table 6, reveals that the best scoring solution is to use gravity gradients to point the satellite towards Earth. Unfortunately, the large and qualitative stability error of this system is too high. Thus the final system incorporates a gravity gradient system to keep the craft mostly stable and a magnetorquer for fine adjustments. By combining these systems the performance required will be achieved while saving power as the magnetorquer does not need to operate at all times. The momentum wheels proved to be poorly suited for this low cost 1U mission as the very accurate stability they provide is not required for the mission purpose.

## 10 Conclusion

A systems engineering approach has been taken to design the best low cost satellite-based platform for measuring the ionosphere electron density and calibrating radio astronomy measurements. The design resulted in a 1 U sized CubeSat orbiting at an altitude of 435 km with an orbital inclination of 45°. This satellite will be able to service the primary site, MWA, over 4 times a day as well as other major radio astronomy facilities in South Africa and North America. The satellite will be able to send two radio signals of different frequency between 80 and 400 MHz simultaneously for measurement on the ground. The lifetime of all systems is well over the target of one year and an Australian manufactured end of life system has been selected to remove the satellite from orbit at the end of its life so as not to add to the proliferation of space junk. The CubeSat will be stabilized by utilising a combination of passive gravity gradients and a magnetorquer. It will be powered by standard CubeSat solar panels and an onboard battery. All of the above systems will be controlled by an on board central processing unit (CPU).

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# Appendices

| Design Requirement | | A | B | C | D | E | F | G | H | I | J | K | L | M | Total | Rank |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Frequency of orbits over primary site | A | | 1 | 1 | 0 | 1 | 1 | 1 | 1 | 1 | 1 | 0 | 0 | 1 | 9 | 4 |
| Altitude | B | 0 | | 1 | 0 | 1 | 1 | 1 | 1 | 1 | 1 | 0 | 0 | 1 | 8 | 5 |
| Time over primary site | C | 0 | 0 | | 0 | 1 | 1 | 1 | 1 | 0 | 0 | 0 | 0 | 1 | 5 | 8 |
| Stability | D | 1 | 1 | 1 | | 1 | 1 | 1 | 1 | 1 | 1 | 0 | 0 | 1 | 10 | 3 |
| Orbit decay time | E | 0 | 0 | 0 | 0 | | 0.5 | 0.5 | 0.5 | 0 | 0 | 0 | 0 | 1 | 2.5 | 9 |
| Power supply lifetime | F | 0 | 0 | 0 | 0 | 0.5 | | 0.5 | 0.5 | 0 | 0 | 0 | 0 | 1 | 2.5 | 9 |
| BUS unit lifetime | G | 0 | 0 | 0 | 0 | 0.5 | 0.5 | | 0.5 | 0 | 0 | 0 | 0 | 1 | 2.5 | 9 |
| Stability system lifetime | H | 0 | 0 | 0 | 0 | 0.5 | 0.5 | 0.5 | | 0 | 0 | 0 | 0 | 1 | 2.5 | 9 |
| Radio transmitters | I | 0 | 0 | 1 | 0 | 1 | 1 | 1 | 1 | | 0 | 0 | 0 | 1 | 6 | 7 |
| Power available | J | 0 | 0 | 1 | 0 | 1 | 1 | 1 | 1 | 1 | | 0 | 0 | 1 | 7 | 6 |
| Launch cost | K | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | | 1 | 1 | 12 | 1 |
| Component cost | L | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 0 | | 1 | 11 | 2 |
| Time over alternate sites | M | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | | 0 | 10 |

Figure 7: Pairwise analysis of design requirements.

| | Containerized | | | Satellite Class | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| Payload Class | 3U (5kg) | 6U (10kg) | 12U (20kg) | 50 kg | 100 kg | 150 kg | 200 kg | 300 kg | 450 kg* | 750 kg* | 1000 kg* |
| Length (cm) | 34.05 | 34.05 | 34.05 | 80 | 100 | 100 | 100 | 125 | 200 | 300 | 350 |
| Height/Diameter (cm) | 10.0 | 10.0 | 22.63 | 40 | 50 | 60 | 80 | 100 | 150 | 200 | 200 |
| Width (cm) | 10.0 | 22.63 | 22.63 | 40 | 50 | 60 | 80 | 100 | | | |
| LEO | $295 | $545 | $995 | $1,750 | $3,950 | $4,950 | $5,950 | $7,950 | $17,500 | $22,000 | $28,000 |
| GTO | $650 | $995 | $1,950 | $3,250 | $5,950 | $6,950 | $7,950 | $9,950 | | | |
| GSO/LLO | $995 | $1,990 | $3,250 | $6,500 | $9,950 | $12,950 | $15,950 | $19,900 | | | |

Table 7: Table of estimated launch costs for satellites of various of size, mass and orbital altitude. Table taken from [14].