SSPI SATELLITE DESIGN COMPETITION SUBMISSION 2016-2017

The University of Sheffield

Abstract
Report detailing the findings of The University of Sheffield's investigation into a communications satellite constellation for Low Earth Orbit.

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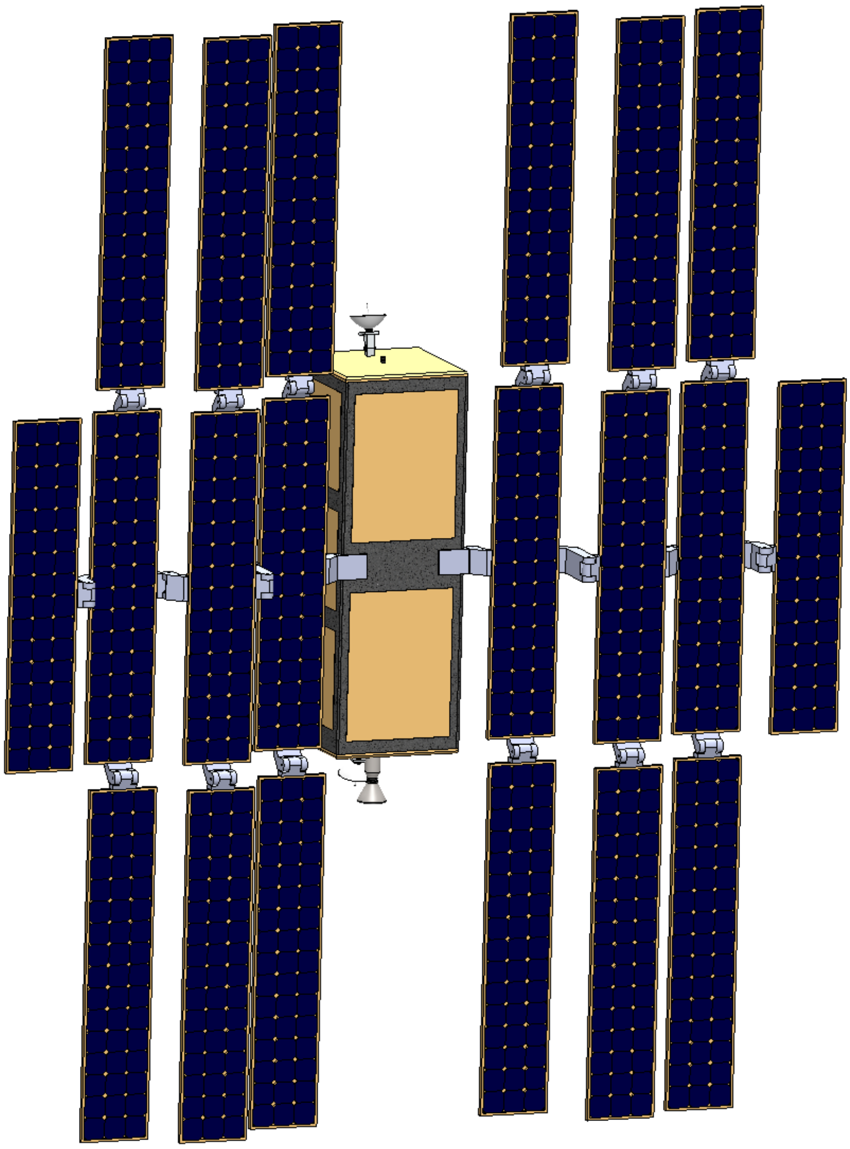
Project Talaris

Delivering affordable connectivity to the world.



1. Introduction

1.1. Description of the Mission

Small satellites are becoming increasingly viable in a commercial sense in recent years with the small satellite market estimated to be worth 7.53 Billion USD by 2022 [1], growing in scope from their value as amateur/educational projects and small scale experiments to a point where they can be considered for use in large-scale constellations for satisfying the world's communications requirements. This report illustrates that viability.

1.2. Team Focus

A satellite can be designed to fill any type of different communications requirements: internet, voice communications, and location-finding services being a few examples. The design considerations and objectives for each type will be markedly different, so there is no one size fits all approach. Therefore, this proposal will consider Internet Coverage.

1.3 Design Methodology

As the optimal design of a constellation of satellites will depend on many factors, the design process will be iterative, and as such a systems-based design process was adopted, a process encapsulated in the Spiral Model:

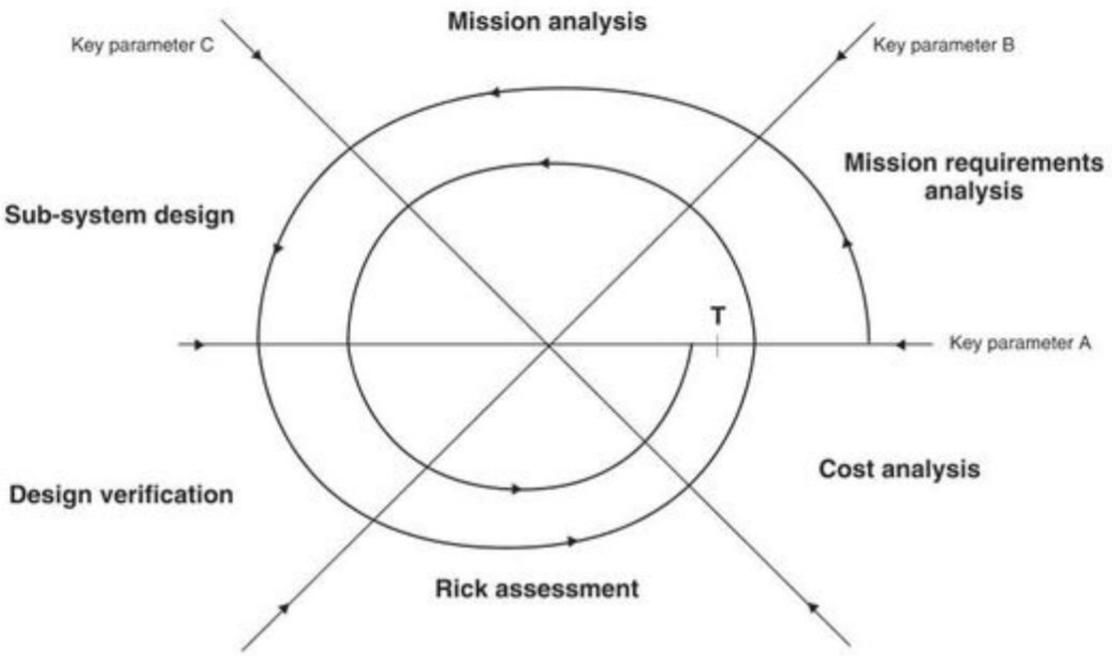


Figure 1: Spiral Model of systems engineering [2].

To that end, the following process flow diagram was chosen for the proposal's design methodology:

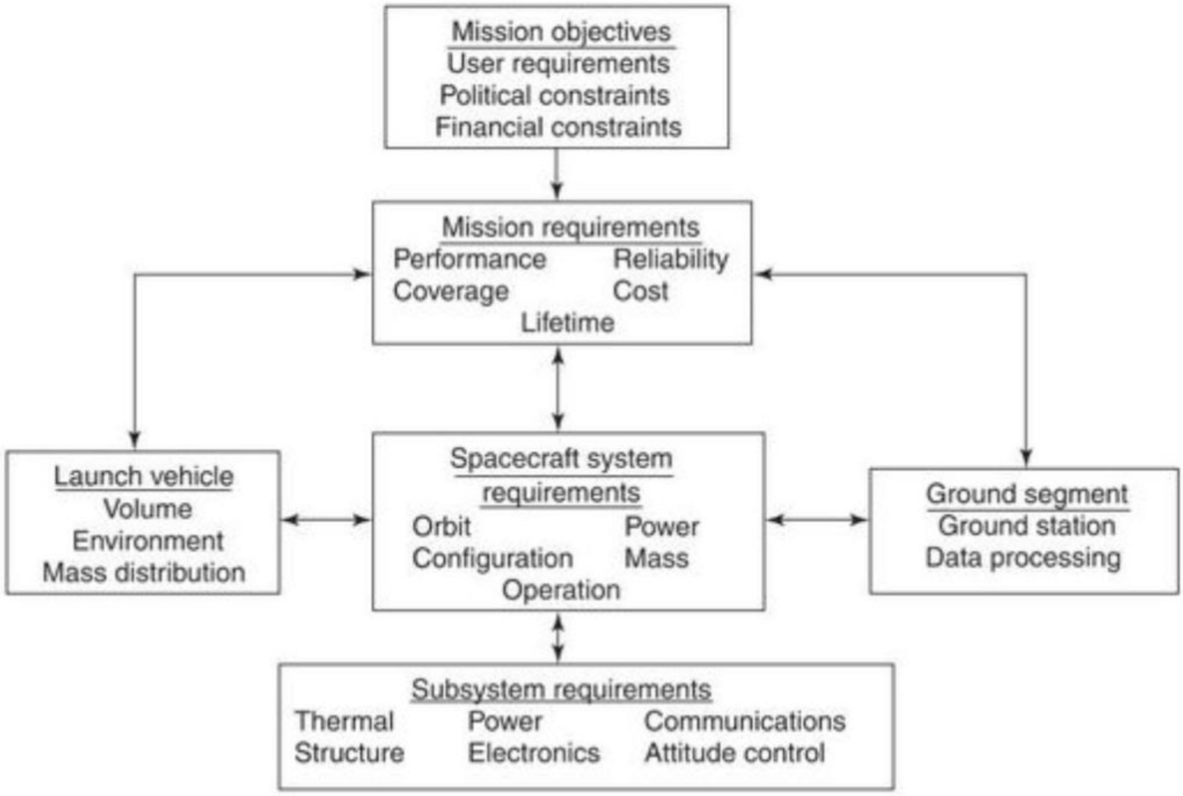


Figure 2:Model structure adopted through design project [2].

Once again, the double-headed nature of the arrows shows that this problem is very complex: one change could have dramatic effects on multiple other components of the system.

1.3.1 A Note on Concurrent Calculations

Due to the above mentioned highly interwoven nature of the spacecraft constellation design, a concurrent approach was identified as the best method for finding an optimal solution, as it goes hand in hand with the systems-based approach of tackling the problem. To achieve these solutions, desirable properties that the constellation and the individual would have been identified, and modelled as functions of their dependent variables, i.e. the required transmission cone size of a satellite's antenna might be dependent on orbital height, input power, and the number of satellites in a plane.

1.4 Overall Mission Objectives

The following are statements of the over-arching goals that the proposal must meet to be of use to the customer:

> "To provide the majority of the global population with internet coverage from LEO with suitable connectivity."

Rationale: *Although there are already constellations oriented towards this service, the number of users and their requirements continue to demand a higher data rate, due to both an increase in the number of users thanks to population growth, and the more strenuous applications the service is used for, with the growing influence of cloud computing.*

1.5 Overall Mission Requirements

To achieve the Mission Objectives, the following requirements are the quantified performance metrics that must be met to deliver on the objectives:

1.5.1 "The constellation must be able to deliver 50Mbps of data connectivity to small antennae on the ground."

Rationale: *One of the performance requirements specified in the brief. This requirement has been taken as being the "required data rate" that represents the need of the customer.*

1.5.2. "Each unit in the constellation must have a weight have no more than 150kg".

Rationale: *One of the performance requirements specified in the brief. The 150kg mandates this must be a small satellite constellation, in order to design to the advantages whist considering the disadvantages of the small satellite approach.*

1.5.3. "Satellite coverage of ± 67° latitude."

Rationale: *After preliminary research, it was found that most major population centres lie in this range. Only research stations and very few other users that would require the service lie outside of it – and the cost of extending the array to those users would be prohibitive when the extra cost is concerned.*

1.5.4. "The entire cost of the constellation and all its required maintenance and other services needed for its continued operation must total no more than $5,000,000.

Rationale: *Research of past constellations such as Iridium Next and OneWeb shows costs of $3 billion* [3]. *Due to the higher coverage requirement, as well as the more advanced technologies being used and the challenges that will be encountered, up to $5 billion has been deemed a suitable price.*

1.5.5. “The constellation as a system should have a total lifetime of 25 years.”

Rationale: *Earlier constellations have shown lifetimes of 5+ years* [3]. *A lifetime that improves on this, whilst not being too long to not take advantage of significant advances in technology has been chosen.*

1.5.6. “The constellation must be built with sufficient redundancy to avoid any significant interruption in its service for the duration of its lifetime.

Rationale: *To account for inevitable unit failures over a long lifetime in LEO, redundancy must be built into the constellation to seamlessly phase out deficient units and preserve good service.*

1.5.7. “The constellation must be able to be disposed of at the end of its lifetime, either by safe de-orbit or disposal in a graveyard orbit.”

Rationale: *Due to the high traffic in LEO, the constellation must not spend any significant amount of time past it’s service life in this orbit and risk collisions with other operational spacecraft, or contribute to Kessler Syndrome. De-orbiting or moving the constellation’s units to a designated graveyard orbit is a suitable solution.*

1.5.8. “The constellation must be able to handle communications data between units.”

Rationale: *To keep continuous service, the constellation must be able to transfer data between its units.*

1.5.9. “The constellation must minimise interference with already existing communications networks that are present in Geostationary Orbit (GEO).”

Rationale: *There are already well-established communications networks that rely on satellites in GEO. These will pass over the LEO constellations units, and so the signals must not interfere to disrupt the service of already established constellations.*

1.5.10. “The constellation must be suitable for providing internet access to more remote and developing areas.”

Rationale: *Developed urban areas such as Europe and Coastal America have well established internet infrastructures made up of high-speed fibre optic cables. Therefore, the market for minimal infrastructure internet would be greatest in regions such as South America and Africa where there are remote communities with no pre-existing infrastructure."*

2. Satellite Constellation

2.1. Constellation Requirements

2.1.1. Performance

- Should deliver internet service at 50Mbps with 16Mbps minimum. Latency should be <300ms (comprising <60ms transit time (based on MEO estimates) + processing time). We need to determine what 'Small Antenna' means, if it means something that can go inside a modern phone that's going to be a lot more difficult than a router sized device.
- Each satellite should deliver 50Mbps of data connectivity to small antennae on the ground.
- Each satellite should hand off communications traffic to other satellites as they pass over the user, to ensure constant connection.
- Must be able to transmit sufficient data between satellites.
- Each satellite should be able to independently maintain its orbital station.
- The system should avoid interference with other satellite communication systems.
- Each satellite should be capable of safely de-orbiting at the end of its useful life.
- The constellation should be comprised of satellites arranged in orbital planes.
- Must be able to maintain connectivity with the ground stations in its coverage range, this means backups must be present to ensure connectivity.

2.1.2. Reliability

- The system should implement redundancies where the financial situation and weight limitations allow it.
- Sufficient spare units should be present in orbit to cope with inevitable failures.
- Each unit must contain and be able to switch to an alternate power system in case of failure.
- All satellite components should be designed to withstand solar radiation, neutral particles and debris for the lifetime of the system.
- The chosen launch vehicle should have an outstanding safety record, to protect our investment from failure and to protect public safety.

2.1.3. Coverage

- Constellation should provide full coverage to the majority of the population of the planet, focusing on remote areas where satellite infrastructure would be costly.

2.1.4. Cost

- Should be <$3 billion, the cost of the OneWeb and Iridium Next Constellations.
- Must take into account launch and long term costs as well as materials and ground stations.
- Comparable to existing constellations.

2.1.5. Lifetime

- Must be able to last a minimum of 25 years. This allows us to give a good lifetime, but not over-invest when new technology could be coming down the line in a couple of years.

2.1.6. Launch

- Must be able to launch from an existing site or sites.
- Must be compatible with existing launch vehicles (Falcon 9, Ariane 5..).
- Must interface with launch vehicle as required.

2.2. Constellation Structure

The constellation design must meet the above requirements, with the main driver being data transmission. In achieving this, the system must pass data from terrestrial telecommunication cables to small antenna on the ground, via the satellite constellation. For an LEO constellation two possible system architectures exist [3]:

1) Bent Pipe

No links between satellites, so data is passed from a ground transmission station up to an orbiting satellite before being transmitted back down to the user on the surface. This architecture results in simpler satellites as inter-satellite communication equipment is not required, however more transmission stations on the ground are required.

2) Inter-Satellite Links

Linking between satellites may be to a varying degree, but as a minimum a satellite would be capable of passing data to and receiving data from its immediate neighbours. Data can then be passed from a ground station to an overhead satellite and forwarded on to another satellite over the user, which can then transmit the data to the user. This reduces the number of ground stations required, but increases the complexity of the satellites.

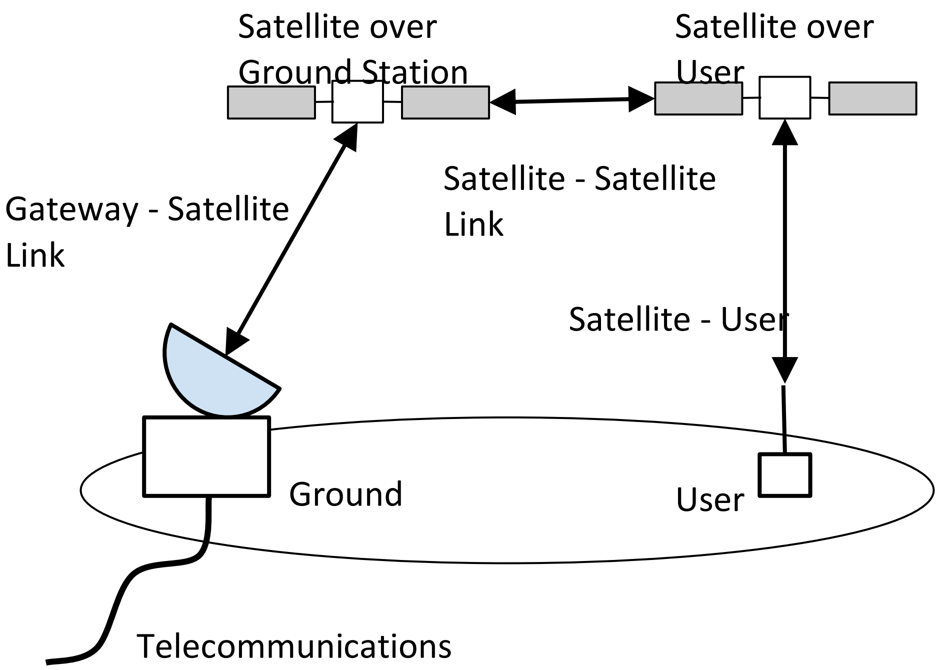


Figure 3: Diagram showing a simplified version of the system, featuring an architecture with Inter-Satellite Links.

An architecture featuring inter-satellite linking was selected because, as the ground stations need not be as close to the user, more remote and less developed areas of the globe can be reached, with less additional infrastructure on the ground. This basic operation of such a system is shown in figure XX, where the Satellite - User Link must provide a minimum 50 Mbps connectivity.

As the constellation is required to only provide coverage between -56 and +67 degrees of latitude, a Walker Delta constellation layout with an orbital inclination of 67 degrees would be most appropriate.

2.3. Top Level Calculations

Once the constellation structure was known, it was necessary to find values for the main constellation parameters, such as the number of satellites, altitude and antenna power to enable further design work to be carried out. Optimisation of these variables was achieved by tracing the relationships between the variable of interest to identify how they were linked.

2.3.1. Inputs

When identifying the relationships between the main constellation variables, it was found that a number of variables would have to be assumed initially to enable solutions for the main parameters to be calculated. These were identified via applying engineering knowledge and then use of an iterative process using MATLAB's Communications modelling.

Table 1: Assumed values for the input variables required to find the constellation parameters.

Input Variable	Value
Atmospheric Attenuation	Negligible
Frequency (GHz)	38
Satellite Length (m)	1.5
Satellite Width (m)	0.5
Ground Antenna Width (m)	0.6
Maximum Latitude (rad)	1.508
LNA/LNB Gain (dB)	65
Receiver Sensitivity (dB)	-120
Effective Receiver End Gain (dB)	185
Subsystem Power Usage (W)	2400
Solar Panel Area (m^2)	9
Solar Panel Efficiency (%)	35
Solar Panel Degradation Rate (%/year)	0.5
Solar Intensity in LEO (W/m^2)	1367
InterSat Antenna Diameter	0.15

2.3.3. Outputs

Based on the dependent inputs and their dependencies between various variables a set of outputs was determined to carry forward in sub-system calculations. Namely, the number of satellites, orbital planes and therefore the required satellites per plane. In addition to; the rate of orbital decay and launch vehicle requirements. With knowledge of these outputs an initial estimate of the characteristics of the constellation could be made and values for sub system level calculations.

Table 2: Values calculated for the constellation parameters.

Output Variable	Value
Number of Satellites Per Plane	29
Number of Planes	48

Total Number of Satellites	1392
Altitude (km)	500
Power Usage per Beam (W)	9.2838
Number of Beams	126
Power Usage per Beam Set (W)	1169.7588
Reflector Diameter (m)	0.0715
InterSat Distance (km)	1377
InterSat Beam Power (W)	23.3967
Total InterSat Beam Power (W)	46.7934

2.4. Constellation Orbital Insertion and Maintenance

2.4.1. Launch Vehicle, Orbital Insertion and Orbit Transfers

The Launch Vehicles were narrowed down based on cost/kg for insertion to orbit, it was found that the SpaceX Falcon series were significantly cheaper in this regard with the Falcon Heavy being approximately half the cost/kg of the Falcon 9, which itself was a fraction of the cost of other launch vehicles such as the Arianespace Ariane 5 or ULA Delta IV [4].Both the Falcon 9 and Falcon Heavy have been analysed as the additional fuel required for additional plane changes is expected to have a significant impact of the payload weight. In terms of future launch vehicles it is expected that the Arianespace Ariane 62 will be of similar cost to the Falcon 9 and 64 to the Falcon Heavy, however as the exact specifications are not yet available it was not possible to analyse. When these are available it would also be necessary to determine the effects of the change in launch site and also the orbit that the Ariane 6 series will insert the payload to.

The Falcon series rockets only insert their payloads to an elliptical orbit of about 200x369km which is not high enough that it would be advantageous to have the Satellites insert themselves into the desired 500km circular orbit from here [5]. Hence it was determined that an additional 3rd Stage would be required in the vehicle to complete the first half of the Hohmann transfer to this higher orbit followed by a fraction of the second half in order to create the correct periodicity between satellite releases. The period of the 500 km circular orbit is 5668.14751 seconds, and with 29 Satellites per plane it is thus necessary to implement a 195.45336 second delay in between the satellites, resulting in a 3rd stage final orbit of 140x500km. As this resulted in an overall greater deltaV due to the periapsis reducing and thus the satellites requiring a greater deltaV to complete the Hohmann transfer, the period difference was halved resulting in a 3rd stage orbit of 319.8x500 km. The third stage would also complete the inclination changes of 3.5924 degrees or a deltaV of 0.4893km/s. The second half of the Hohmann transfer will be completed by the satellites themselves as it would involve a massive amount of fuel otherwise in order to distribute the satellites across the plane by reversing at least some of the second step of the Hohmann transfer for each Satellite.

By using iterative calculations with launch vehicle payload as the input, and analysing the additional weight of satellites and fuel required for each additional plane insertion, the optimum number of planes per launch for each launch vehicle was determined and as a result the average cost to insert a Satellite into orbit for the initial global coverage. It was assumed that the structural weight of the 3rd stage would be >6% as similar vehicles had a structural weight of 4.5-5.5% but our vehicle would need additional equipment to hold and release the large number of satellites onboard.

This resulted in the Falcon Heavy being chosen with a cost of $367,350 per satellite with 7 planes or 203 satellites being introduced each launch compared to $590,480 per satellite with 3 planes or 87 satellites introduced for the Falcon 9. A total of 4081.9kg, or up to 6.4% of payload weight, was left available for the structure of the 3rd stage vehicle.

2.4.3. Satellite Orbital Transfer Phase

As mentioned in the previous section the Satellites are released at the apoapsis of an elliptical orbit of 319.8x500km, meaning they must circularise to 500km. The deltaV required for this transfer is 50.8m/s, assuming hypergolic fuels are used this corresponds to a mass fraction of 1.76% which is an accommodatable amount.

2.4.4. Orbital Maintenance

The orbital maintenance for a constellation at an altitude of 500 km (Section 2.3) was estimated to further indicate design requirements. The following calculation was used to determine this.

The total specific energy of an object in orbit is assumed to be the sum of its specific gravitational potential, and kinetic energy.

$$e_{total} = e_{GP} + e_{KE} \tag{A}$$

Recalling formulae:

1. Specific GPE in a gravitational field pertaining to mass M:
$$e_{GP} = -GM/r \tag{B}$$
2. Specific KE:
$$e_{KE} = v^2/2 \tag{C}$$
3. Circular orbit centripetal force relation:
$$v = \sqrt{GM/r} \tag{D}$$

Substituting D into C:

$$e_{KE} = GM/2r \tag{E}$$

Substituting E and B into A:

$$e_{total} = e_{GP} + e_{KE} = -GM/r + GM/2r = -GM/2r \qquad \text{(F)}$$

F is an expression that relates total energy of the satellite, e_{total}, to altitude by means of radius, r. Note that here, the maximum energy is zero; therefore r is not inversely proportional to e_{total}. Hence, by calculating work done against drag in one orbit and equating this to a total energy loss (due to conservation of energy), the change in radius after one orbit can be determined.

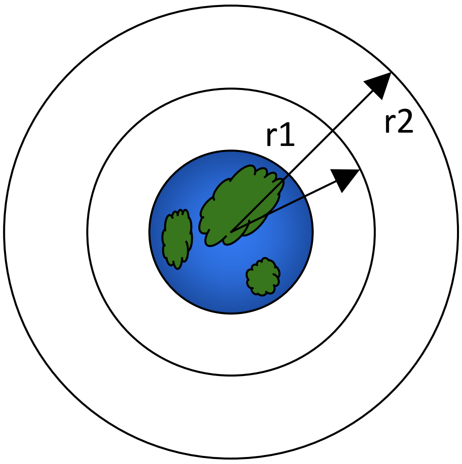


Figure 4: Depicts the energy states of the satellite.

Consider two states of a satellite (Figure 4): 1 and 2 at radii r1 and r2 respectively, where the orbital height change r1-r2, occurring over a single orbit, is caused by drag.

Total Energy at the two states:

$$E_1 = -GMm/2r_2 \,, E_2 = -GMm/2r_1$$

Therefore, the change in energy is given by:

$$E_1 - E_2 = \frac{GMm}{2}\left(\frac{1}{r_2} - \frac{1}{r_1}\right)$$

$$\frac{1}{r_2} = \frac{1}{r_1} + \frac{2(E_1 - E_2)}{GMm} \qquad \text{(G)}$$

Assuming that the change in energy is due to a work transfer against drag

$E_1 - E_2 = W_{drag}$.

Here, the work done against drag is modelled, for a single orbit, as the product of the drag force and a single circular orbit. The assumption of the constant radius per orbit is made here, however it cancels in the following step.

$$W_{drag} = F_{drag}\, s_{orbit} = \frac{1}{2}\rho A c_d v^2 \times 2\pi r_1$$

Velocity in a circular orbit:

$v^2 = GM/r_1$

Therefore

$$W_{drag} = \pi \rho A c_d GM. \qquad \text{(H)}$$

Substituting (H) into (G):

$$\frac{1}{r_2} = \frac{1}{r_1} + \frac{\pi \rho A c_d}{m},$$

$$r_2 = (\frac{1}{r_1} + \frac{\pi \rho A c_d}{m})^{-1}.$$

Orbital decay per orbit due to drag:

$$r_1 - r_2 = r_1 - (\frac{1}{r_1} + \frac{\pi \rho A c_d}{m})^{-1} \qquad \text{(I)}$$

A simple atmospheric model was created for the density variation with altitude through a power fit interpolation of data from [6]. A script was written to implement equation (I) iteratively until a specified decay tolerance was exceeded.

Table 3: Calculated parameters for satellite undergoing circular orbit at 500km altitude with a lowest tolerated altitude of 490km.

Number of Orbits before tolerance is exceeded	Elapsed Time before tolerance is exceeded	Energy required to return satellite to 500km	Delta V required to return satellite to 500km
1827	2874 hours	6.34 MJ	5.5486 m/s

For a tolerance of 10 km the results were as seen in Table XX. The result is that for this tolerance, the satellites are estimated to need boosting back to 500 km altitude approximately three times a year; giving confidence to the chosen altitude. Over a lifetime of 20 years this will result in a total deltaV requirement of 0.3383km/s or a mass fraction of 11.16% which is a reasonable amount.

2.5. Final Constellation Design

The constellation will operate by passing data around each orbital plane, from the satellite nearest to the ground station to the one above a given user as illustrated by figure xx. As the position of the satellites above the surface constantly changes in LEO, the satellite above the ground station will change and so will the satellites above the ground users.

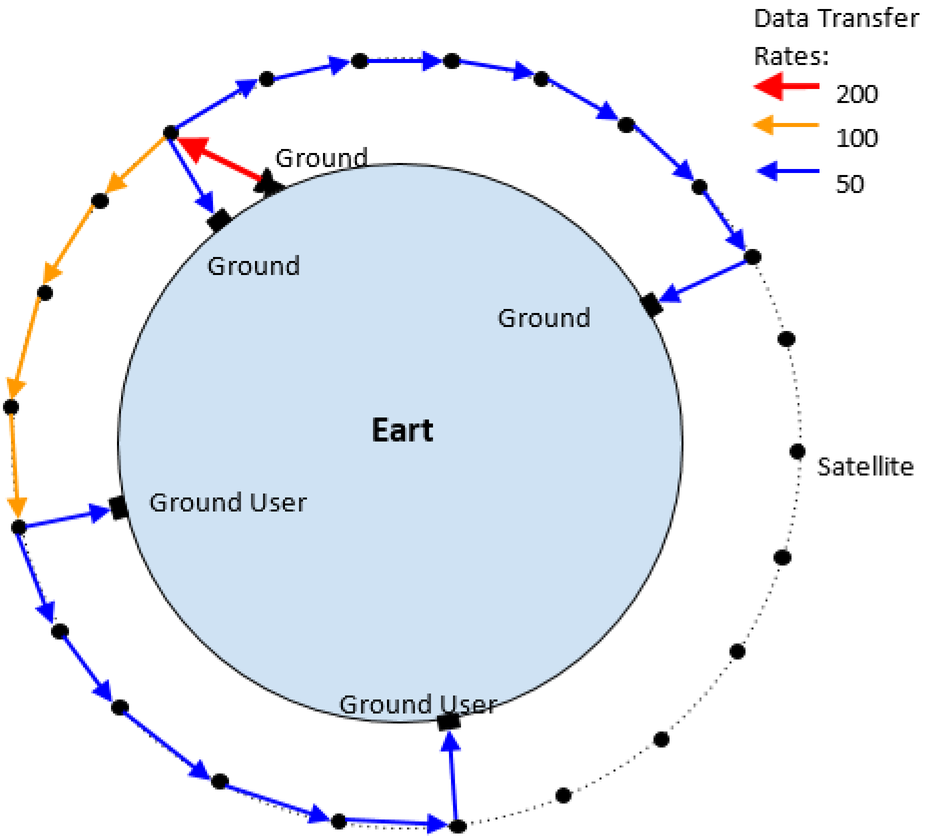


Figure 5: Outward data flow from the Ground Station (connected to terrestrial internet infrastructure) to Ground Users A-D via a single orbital plane of the constellation.

This has two implications. Firstly, the flow of data must be handed from one satellite to another seamlessly as they pass over the surface. Secondly, each satellite must be capable of communicating with the ground station, neighbouring satellites and ground users, meaning each satellite must be identical and also able to process the entire data requirements of one orbital plane. This is shown in figure xx, where the satellite above the ground station must handle all of the data for the four ground users shown.

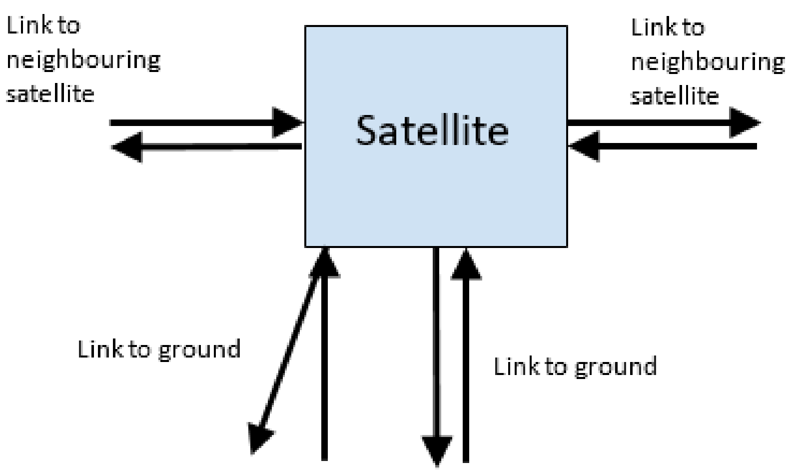


Figure 6: Communications requirements of a single satellite.

Table 4: Summary of Satellite Characteristics.

Variable	Value
Orbit (Km)	500
Number of Satellites per plane	29
Number of planes	48
Total number of Satellites	1392
Sat-Ground Beam Power (W)	9.2838
Number of Sat-Ground Beams per Satellite	126
Sat-Ground Power Usage (W)	1169.76
Sat-Ground Reflector Diameter (m)	0.0715
Sat-Sat Beam Power (W)	23.40
Total Sat-Sat Power Usage (W)	46.79
Sat-Sat Reflector Diameter (m)	0.15
Coverage	100% for all longitudes, latitudes up to ±86.172 degrees
Satellite Weight (Kg)	150
Propellant mass fraction	12.92%
Launch Vehicle	SpaceX Falcon Heavy
Number of Launches Required	7
Average launch cost per Satellite ($)	367,350
Total Launch Costs ($)	630,000,000

3. Satellite Hardware

3.1. Communications Equipment

3.1.1. Antenna

Due to the high frequency used it is possible to use parabolic antennas due to the small diameter required for high gains. As in the top level numbers this resulted in 126x 7.15cm diameter parabolic dishes running at 9.28W transmission power each for Sat-Ground Communications and 2x 15cm diameter dishes running at 23.4W transmission power each for InterSat communications.

3.1.2. Transceiver

The Transceiver transmits and receives signals between the satellite and the ground. The most important part is the modulator as this will determine possible data rates. Inspiration for this design has been taken from the DVB-S2X standard and as such 64APSK shall be used for the modulation scheme between the Satellite and User Ground Stations as it offers a high data rate even in poor weather conditions [7].

For communication from the provider ground stations and between satellites, 256QAM will be used to offer higher bit rates, this is possible due to the high power input available at provider ground stations and the low noise between satellites, both resulting in a higher signal to noise ratio.

The transceiver will be software defined as this reduces the cost, weight, and size significantly over hardware designed solutions while also allowing for optimisations to the software in the future to improve the performance of the satellite.

Ideally this would be custom designed as there are no off-the-shelf solutions offering Ka band radio with 64APSK and 256QAM capabilitiies for 128 channels, as time did not allow for designing a solution to this problem our solution is based upon an off-the-shelf part which offers similar capabilities for just 1 channel. The HackRF One is an open source Software Defined Radio offering frequencies up to 6GHz, an assumption is made that in several years time it would be possible to build a similar device which offers up to 38GHz. Given the presence of other SDR Ka radios such as the Harris-NASA SDR and the increasing use of Ka band, we believe this is a valid assumption. The total of 128 HackRF Ones result in a max power usage of 320W with a total weight of 12.8kg. A custom solution would have much better performance in these due to centralisation of large, power hungry components such as the modulator.

3.1.3. Data Rate

Using the DVB-S2x charts for Signal-To-Noise Ratio (SNR) against Spectral Efficiency (bps/Hz), it was possible to determine the data rate given estimates of the SNR and a bandwidth calculated using the Communications Toolbox in MATLAB.

A conservative estimate for the SNR was 7 which corresponds to 2bps/Hz, while a more liberal estimate is an SNR of 20 which gives 5.5bps/Hz. The bandwidths were then estimated by creating a signal at 3.8GHz, which is sufficiently low enough that it could be adequately sampled by a 38GHz frequency, then adding gaussian white noise at the max and minimum SNR. This provided bandwidths of 1.2288GHz for SNR=7 and 1.1468GHz for SNR=20. For InterSat transmissions an SNR of 30 and 256QAM modulation will be used and for Provider-Sat transmissions SNR of 40 and 256QAM will be used.

By using Shannon-Hartley Theorem it is then possible to determine minimum and maximum data rates provided by each beam. This provides the following data rates per beam:

Table 5: Data rates per beams.

Signal	Data Rate (Gbps)	Users
Sat-Ground Minimum	7.3731	147
Sat-Ground Maximum	28.208	564
InterSat	44.954	899
Provider-Sat	48.459	969

In order to saturate a satellite under a low SNR scenario it would be necessary to have 17 uplinks from the Provider station to fulfil the requirements of the satellite alone. As each satellite covers a ground area of 1420x400km or 568,000km^2 it is reasonable to assume that the amount of InterSat Communication will be limited as on a single plane there will only be one or two satellites in each direction before a new Provider station is available. As a result an additional 2 uplink beams will be needed to enable this, leaving 107 beams for Sat-User Communication. In certain situations it would be possible to boost the data rate by linking the Provider station to an InterSat antenna, resulting in a doubling of capacity for medium priority satellites. This would be useful for servicing areas like Kashmir, Central Africa, and the Gulf states where it would be difficult to install Provider stations in the surrounding area due to political issues but a high number of users can be expected.

Where the number of users connecting to a satellite becomes concentrated into a small handful of beams, it is possible to reroute power to these antennas from unused antennas, allowing for higher spectral efficiency in order to maximise the data rate. By doing so a single beam can accommodate 1,145 users.

Further development should focus on this power rerouting as populations tend to be concentrated so many fewer beams would be required in practice than the satellite is designed for. This would have knock on effects leading to reduced power usage. It would also be necessary to increase the total InterSat data rate by increasing the number of beams available for this communication.

3.1.4. Provider Ground Station

Provider stations are necessary to provide the data uplink to the satellites. The number of provider stations per plane will vary depending on demand within those planes. It is expected that an average of 4 stations will needed per plane, leading to a total of ~120 stations worldwide. Due to the requirement for these being in politically stable countries, there may be problems servicing Southern Africa and Central Asia, the latter depending on whether stations in Saudi Arabia and Mongolia are viable.

3.1.5. User Ground Station

Users will require a kit comprising a Software Defined Radio or Hardware Defined Radio, a high sensitivity receiver, an LNB with good gain, and a 0.5m Satellite Dish. If the user chooses a software defined radio, that will negate the need to upgrade their system as support for higher frequencies and new modulation can be added. A hardware defined radio would require upgrading whenever these are changed. A quick estimate would price the SDR option at ~$500 while the Hardware option would be $300.

3.2. Power

As shown in the Top Level Design, Communications Satellites require a large amount of power in order to run the communications system and also the climate control system. To define the power specifications of the satellite, it was decided to model the satellite's orbit as two distinct phases one for night and one for day. Here, by defining consumption for these two phases, the estimated storage requirement and requisite daytime generation could be calculated.

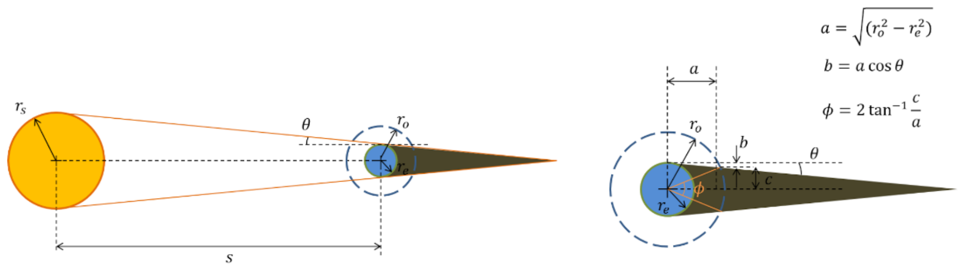


Figure 7: Schematic for obtaining Power model

The above schematics show how the model was obtained. The results were: 4.61E+03 seconds day time and 1.06E+03 seconds night time per orbit.

3.2.1. Generation

Power for the satellite will be generated using solar cells as this offers the best compromise of lifetime, cost, and weight. Approximately 3450w of power is required in order to power all the spacecraft subsystems and maintain a reserve for night-time operations. Taking into a degradation of 0.45% per year, based upon degradation rates of panels on the ISS, this gives a panel capacity requirement of 3780W [8].

The solar panel choice was narrowed down to 2 options:

Manufacturer	Area (m^2)	Weight (kg)	Cost ($)
SolAero	9	7.56	583,515
HighFlex	14	16.2	1,512

The cost of the additional efficiency, radiation resistance, and lower weight and area of the SolAero cells was too high to warrant its use and so the HighFlex cells have been chosen.

These solar panels will comprise wings which fold out of the satellite and will pitch and yaw in order to maintain optimum direction towards the Sun.

3.2.2. Storage

As solar panels are used for the power generation the time in eclipse must be accounted for. It was determined that the satellite would be in eclipse 18.9% of the time, which means 23.3% of energy generated must be stored to cover the deficit during eclipse. The total amount of energy required during this period is 2.89MJ, or 802Wh. Taking into account the overall efficiency of the battery, 80%, the energy that must be stored is 1003Wh. EXA BA0x Lithium Polymer batteries were chosen as these have a proven record in Satellite environments. As these were 53.2Wh batteries, a total of 19 were needed.

The above estimate did not take into account the degradation of the batteries however, due to the low orbit and thus high night time usage and cycle rate of the batteries, in order to maintain a sufficient battery capacity for even 5 years the actual battery capacity would need to be much higher to the point where the weight of such an array would not be feasible within the 150kg limit. A better option to ensure long term functionality would be to use a regenerative fuel cell loop comprising of a dual Proton Exchange Membrane system, creating water which would be recycled through the system. While this would have a much lower initial efficiency of 49% using practical estimates, the system would suffer from minimal degradation, with a 0.1% loss in efficiency after 20 years. It would however result in a much greater weight for the power storage contribution than the initial estimate without degradation, although this would not put the weight estimate outside of the 150kg limit.

3.3. Thermal Control

Selection of a suitable thermal control subsystem is necessary to ensure the satellite is able to operate optimally due to temperature constraints of various pieces of hardware. Further to this, is the fact that the satellite is being heated via solar and albedo radiation and also being cooled whilst in the Earth's eclipse to extremely low temperatures the external elements of the satellite is exposed to.

Research was undertaken and it was found that for the operating altitude the radiation experienced is as seen in table 6.

Table 6: Thermal Radiation data.

Solar Radiation (Wm^{-2})	Albedo Radiation	Terrestrial Radiation (Wm^{-2})
1376	35% of Solar Radiation	239

Based on further research and assumptions the following input data was found to form the basis of design calculations for the required thermal control system in table 7. For the basis of first order calculations the following assumptions were made:

- System is shadowed by the Sun and is unaffected by albedo radiation and IR emission.
- Satellite is assumed to be a spherical shape.

From first order calculations altitude of 500km was selected and coupled with the Earth's radius the operational height could be found. The emissivity and absorptivity of solar cells was taken from data of a commercially available triple junction solar cells [9].

Table 7: Inputs for first order calculation.

Electronics Temperature margin (°C)	0 - 40
Temperature Margin (°C)	5
Surface Area (m^2)	3.5
Maximum Electrical Power dissipation (W)	170
Minimum Electrical Power dissipation (W)	80
Altitude (km)	500
Maximum Earth IR emission (Wm^{-2})	258
Minimum Earth IR emission (Wm^{-2})	216
IR emissivity	0.85
Solar absorptivity	0.92
Earth's Radius (km)	6371

From this information a set of design calculations were derived via thermal network analysis and then performed yielding the following outputs:

Table 8: First Order Outputs.

Radiation view factor	0.3128
Albedo radiation correction	0.9892
Maximum satellite temperature (K)	329.64
Minimum satellite temperature (K)	202
Area of Radiator (m^2)	0.392
Radiator Temperature (K)	255.1

The main quantities useful from these calculations are that the area of radiator required to assist in the structure design and operating temperatures.

The value obtained for power needed was to act as a guide in designing the specifics of the thermal control system; it is the total power requirement that the system components must provide collectively. As the satellite system is intended for a lifetime of 20 years, efforts will be made to reduce the likelihood of failures through maximising the use of passive components.

3.3.1 Passive system

The basis of the passive thermal control system was to be from a Multi-layer insulation (MLI). This is layers of of thermal insulation to prevent losses by thermal radiation. To reduce complexity and cost of the design it would be preferable that a significant portion of the overall thermal control was achieved by the passive system.

MLI typically consists of 3 layers- an outer cover which is opaque to sunlight and can survive the extreme temperatures of space, interior with low emittance properties and an inner cover which covers the hardware of the spacecraft.

Research was undertaken and suitable materials were found; outer cover - PTFE Teflon, interior - Mylar, inner cover - Nomex. With knowledge of this a thermal resistance network analysis was performed to determine the rate of heat transfer through the satellite walls assuming cuboid construction.

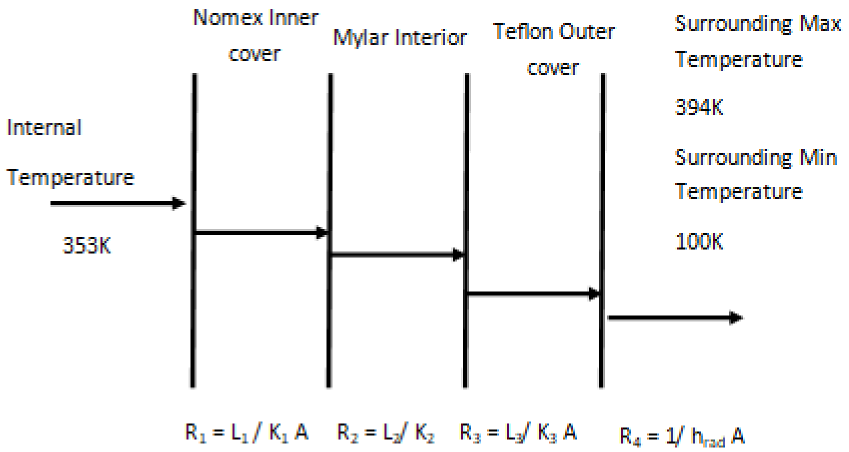


Figure 8: Schematic used to determine insulation requirements.

$$hrad = \varepsilon\sigma(Ts^2 + Tsur^2)(Ts + Tsur)$$

Table 9: Inputs and Outputs for Insulation.

A1 (m^2)	0.75
A2 (m^2)	0.25

K1 (W/mK) [10]	0.1
L1 (mm)	3
K2 (W/mK) [11]	0.15
L2 (mm)	1
K3 (W/mK) [12]	0.238
L3 (mm)	7
h_{rad}	0.93445
Max temperature scenario:	
Total cooling required (W)	866
Min temperature scenario:	
Total heating required (W)	1070

The total thermal resistance was determined via summation and then the heat transfer was found via:

$$Q = \frac{T1 - T\,sur}{Rtot}$$

It was found that the required cooling for the satellite would be 886W sourced by a heat pipe and a heating requirement of 1070W from a thin film heater.

3.4. Electronics

3.4.1. Block Diagram

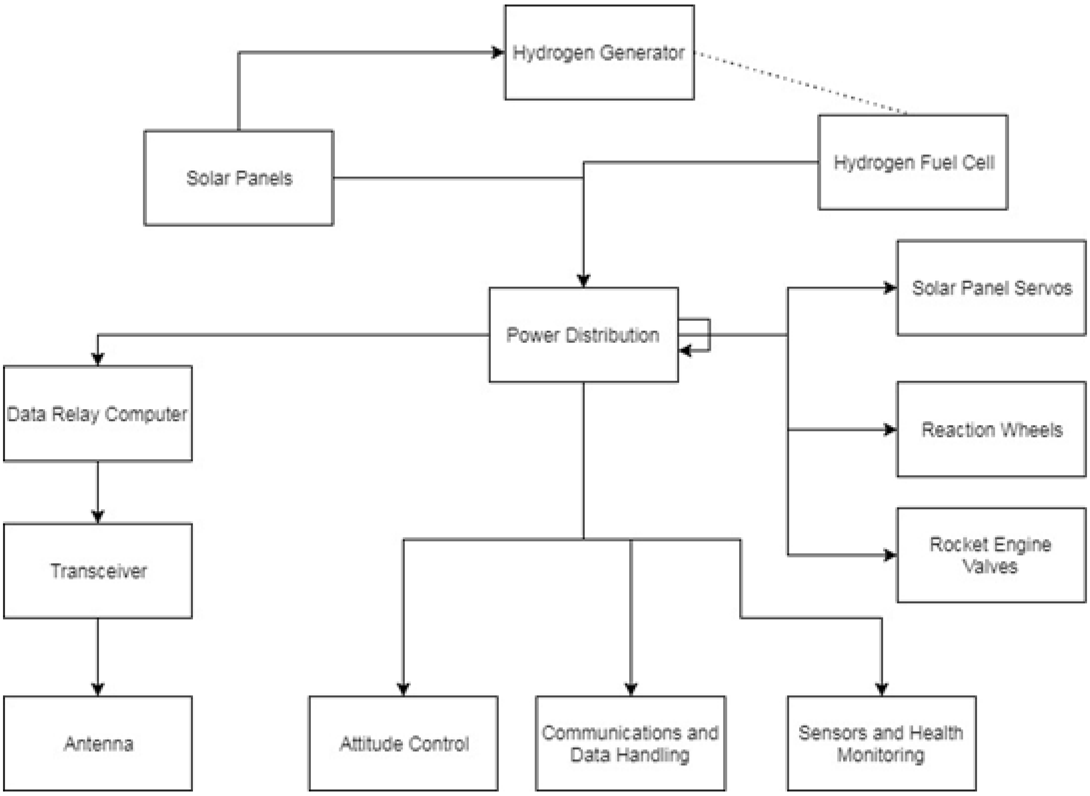


Figure 9: Displays satellites block diagram.

3.5. Propulsion

The satellite requires 2 separate propulsion systems, one on the Satellite itself and one on the 3rd Stage Satellite Carrier. As both of these require stopping and starting frequently it was decided to use hypergolic propellants, as efficiency is key the chosen pair of propellants is N_2O_4 and N_2H_4. While there are more efficient hypergolic propellant pairings, this one is the highest which does not have environmental concerns such as containing Fluorine, Aluminium or Beryllium. Given the risks associated with a rocket launch this makes it the best fit.

3.5.1. Satellite Carrier

The engine requires large, fast impulses in order to accurately perform the manoeuvres necessary for this vehicle. The acceptable manoeuvre period was decided to be 1% of the total orbital period, so 55.71 seconds or less to perform a manoeuvre with a deltaV of 498.3m/s. It will also be necessary to optimise this for a throttling system given the large difference in the spacecraft's weight between the first and final manoeuvres. This resulted in the specifications in table 10 below:

Table 10: Satellite Carrier data.

Variable	Value
Exit Velocity	3370m/s

Mass flow rate	168.34kg/s
Throat Diameter	15.80cm
Nozzle Diameter	99.92cm
Combustion Chamber Volume	$0.0996m^3$
Weight	323kg

3.5.2. Orbital Insertion and Station Keeping

As this engine needs to provide accurate impulses it should provide a low acceleration given the spooling up and down times of conventional rocket engines. It was decided that an acceleration of 1.6m/s would be sufficiently low. A high characteristic chamber length of 89cm was chosen in order to maximise mixing of the propellants and ensure the components were not prohibitively small that they would melt or explode under the chamber pressure [13]. This resulted in a rocket with the specifications outlined in table 11 below.

Table 11: Orbital Insertion Data.

Variable	Value
Exit Velocity	3370m/s
Mass flow rate	0.0712kg/s
Throat Diameter	3.2mm
Nozzle Diameter	20.5mm
Combustion chamber Volume	$129.352cm^3$
Weight	2kg

3.5.3. Attitude Control

The attitude will be controlled using a combination of Gravity Gradient Stabilisation for the main rotation of the satellite and small Reaction Wheels for any corrective procedures. The Gravity Gradient Stabilisation requires an elongated design, i.e. length is significantly longer than width. As the reaction wheels are for corrective manoeuvres they do not need a high torque and in fact a low speed may be an advantage due to the precision required. As a result the Blue Canyon Technologies RWP500 was chosen, offering a Torque of 0.025Nm which results in an angular acceleration of 0.0170 degrees s^{-2} for our Satellite based on 1.5m long Satellite 4 Reaction Wheels will be used, with 3 distributed across the Roll, Yaw, and Pitch axes and a 4th which sits between the Pitch and Roll axes as a redundancy for either of these. During the failure mode it will work in combination with the other functioning wheel cancel out some of the rotation the inter-axis wheel creates to produce rotation only around the desired axis. Performance of the satellite once inserted into orbit is independent of Yaw, and while a failure of the Yaw wheel is possible between release from the launch vehicle and orbital insertion, this can be mitigated by designing the release mechanism to release the satellite in the desired direction. As a result, a 5th backup wheel is unnecessary. The RWP500 weighs 0.75kg resulting in a total mass contribution of 3kg.

3.5.4. De-orbiting

Three methods are possible for de-orbiting the Satellite: Capture and Return, Capture and Replace, Dock and Upgrade, or a simple reduction of periapsis. This will be based upon a lifetime of 20 years.

Capture and Return would require a craft similar to the original 3rd stage vehicle except with addition equipment similar to Canadarm in order to catch the satellite. The satellites would also need additional fuel capacity in order to match the satellite and Capture Vehicle orbit as it would be more efficient to put both in the periodic insertion orbit of 319.8x500km as described in 2.4.1 than requiring the Capture Vehicle to move inbetween the periodic orbit and the circular orbit of the satellites. They would also need an additional margin of error in the fuel capacity for manoeuvring due to inaccuracies in the trajectory. This would add about 0.15km/s to each satellite's DeltaV in total. Using this method and the same Falcon Heavy vehicle, 6 planes could be captured with each launch, resulting in a Capture and Replace rate of 3.25 planes per launch.

Capture and Replace would use a similar craft to Capture and Return except it would also insert new satellites into orbit when capturing the old satellite. In order to minimise the weight of the capture vehicle, the satellites would be released during the periodic insertion orbit like in the original launch vehicle. It would also be advantageous to use one vehicle for each plane due to the large additional fuel mass required to provide the same DeltaV for the larger mass satellite. This would cause considerable complexities in the structure however, adding additional mass for the structure required. The Capture and Return method combined with launching the new satellites separately as the Falcon Heavy would only be able to Capture and Replace two planes per launch, resulting in a much higher cost than Capture and Return.

The most simple way would be by reducing the periapsis of the Satellites' orbits to 40km and allowing atmospheric drag to de-orbit the Satellites. This comes with the disadvantage of requiring a further 0.1331km/s of DeltaV from each Satellite which will impact the mass distribution within the Satellite. It may also cause significant public relations issues as many members of the public will be distressed at the idea of over 1000x 150kg Satellites raining down on the Earth in a relatively short span of time and so this cannot be considered to be a viable option.

Overall the Capture and Return method provides the most suitable method using near future technology. This increases the Satellite's DeltaV requirement to 0.5391km/s or a mass fraction of 17.2%. It also adds a decommissioning cost of $395,610 per satellite and the cost to launch a new satellite will also increase to the same amount. This is likely to change significantly however due to the timeframe involved.

3.6. Structure

The satellite structure was designed to position and protect the components whilst ensuring ease of manufacture. It is acknowledged that the structural design of the satellite remains highly conceptual.

Here, a design is presented and discussed for a layout of major components only; an idea of dimensions and assembly.

3.6.1. Main Frame

The main frame is the central structure to which all components are secured. Briefly, the design is a cuboid frame with rails to mount the components in layers; inspired by the cubesat design.

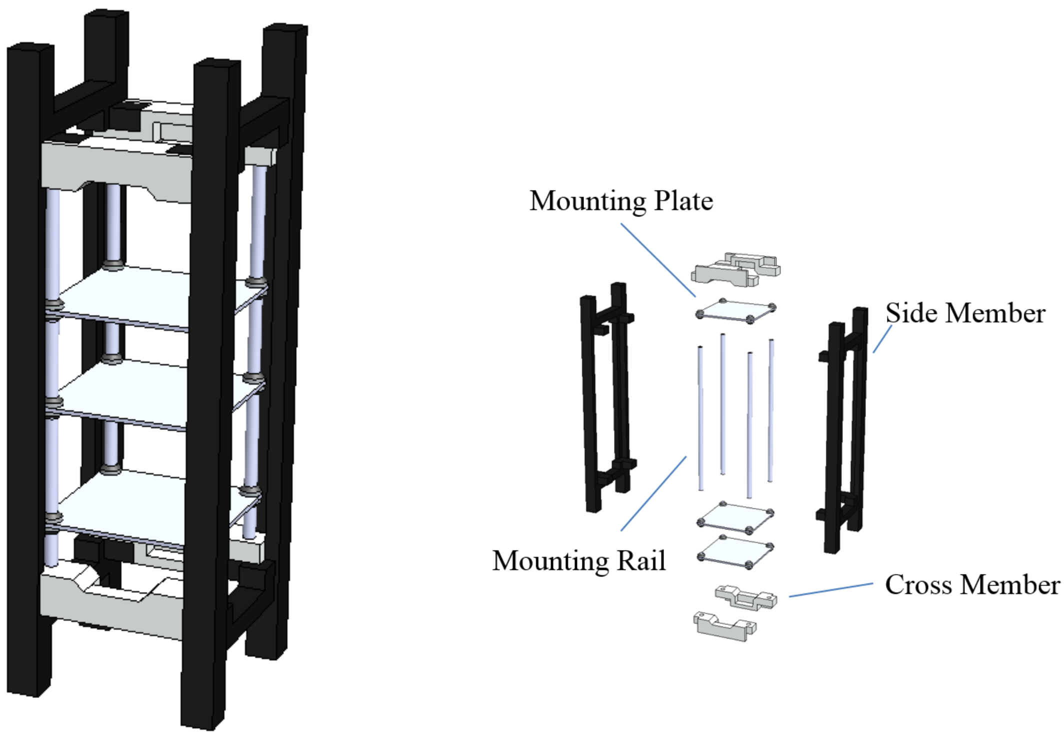


Figure 10: Schematics of main structural components.

As figure 10 Shows there are only 4 key components to this design. Note that each component here has at least one plane of symmetry thus reducing the number of unique parts to be manufactured. The members are to provide the structural integrity of the satellite; bearing the loading during launch. The members also offer fastenings on their exterior surfaces to secure shielding, insulation and other systems such as the solar array and onboard engine (not shown in figure 10). The internal components (CPU), Attitude control system.

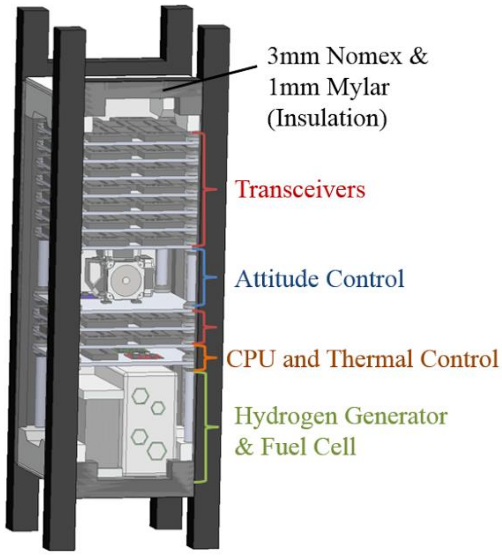


Figure 11: Colour coded CAD of internal components arranged in main frame (inner insulation is shown translucent)

The modularity of the system layout is apparent in Figure 11. Here, the various internal systems are arranged separate mounting plates. The thought behind this feature is that each mounting plates can be manufactured separately before slotting into the frame. The insulation surrounds these major components and the volume shown is maintained at an optimum temperature as discussed in section 3.5.

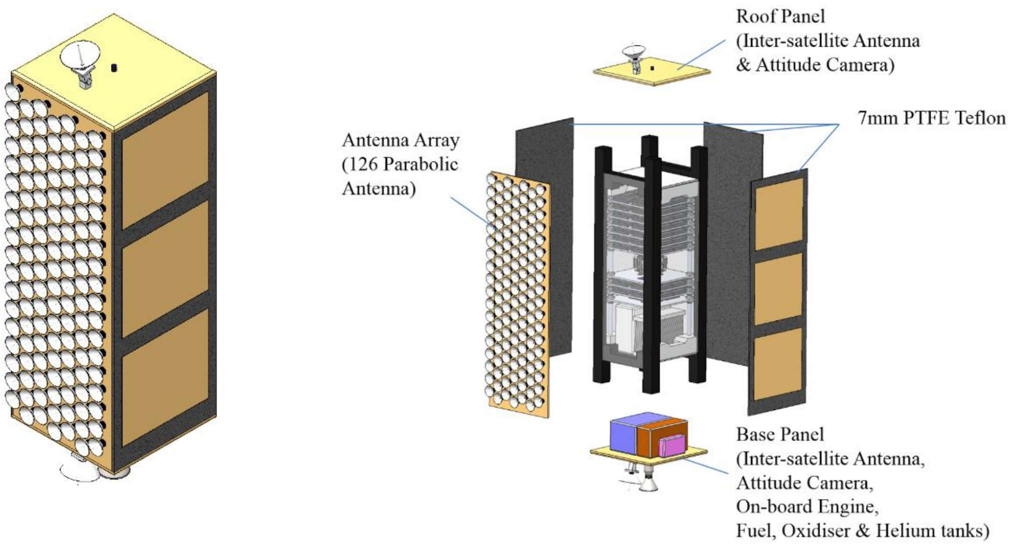


Figure 12: a) CAD of Main Frame assembled b) Exploded view.

The complete arrangement for the main body of the satellite is shown in figure 12; figure 12b highlights the positions of key external components. The patches of golden coating here are included to increase reflection. As shown previously, the satellite is in “day-time” for the majority of its orbit and is in need of cooling.

3.8.2. Solar Array and Deployment

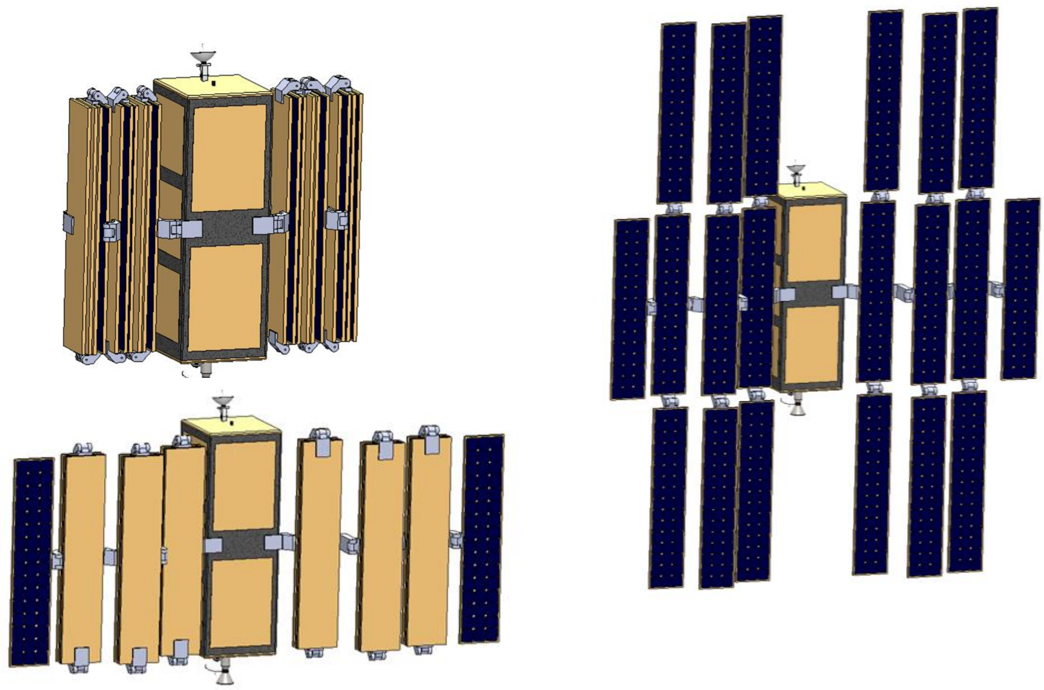


Figure 13: Satellite shown with deployed solar array in three phases

Following the work conducted on the number of panels required to deliver the required power, an appropriate arrangement had to be identified. As multiple satellites are required to be launched together, packing was of primary consideration. The kinematics of the deployment system have been established here but the mechanism has not been found. It is thought that due to the purpose of the satellites, the panels should only be deployed once in a satellite’s lifetime. Hence, it is thought that some sprung loaded hinges may be appropriate rather than including a motor at each hinge. However, if each hinge could be individually controlled, the optimum angle of incidence for the panels could be tracked. The array and its deployment is certainly an area for future development.

4. Production Schedule

4.1. Development Schedule

4.1.1. Sub-System Testing

As all the major components used in the satellites are off-the-shelf models, little individual component testing will be required. The first development stage would therefore be procurement of all the required communication and control components, for individual performance verification and then complete satellite system testing. Testing of the satellite systems would be conducted by simulating inputs to the hardware covering the full range of situations the satellites may encounter throughout their lifetimes, including simulated component failures.

In parallel to the communication and control system testing, the propulsion systems would be tested to provide data on their performance and assess the mating of the fuel delivery systems to the bought-in engines. The structure would also be built and analysed to ensure that it is capable of withstanding the forces it is subject to during launch and orbital insertion. Fatigue testing will also be conducted to simulate the structures response to the low amplitude low cycle loading resulting from station-keeping.

4.1.2. Prototype

Having completed satellite sub-system testing, initially three prototype satellites (to facilitate test repeatability) will be manufactured for full systems testing, including vacuum chamber and radiation exposure trials.

Cubesat projects often take 9-24 months to go from design to launch [14], a far shorter time period than most commercial or government satellites due to their prolific use of off-the-shelf components. The satellite design proposed in this project utilises the same philosophy as cubesats in utilising off the shelf components, however, as this project also aims to provide a reliable and robust service to its commercial customers, it will be assumed that the development phase will take 24 months.

4.2. Ground Segment Schedule

120 ground stations are required across the globe to meet the data capacity required of the system. A project to build 10 ground stations, similar to those required, across Australia was predicted to take 2 years [15]. This project requires considerably more stations to be across the globe, however, as work can be carried out simultaneously on these stations, a ground segment lead time of 4 years will be assumed.

4.3. Space Segment Schedule

4.3.1. Procurement

The below table (table 12) details the lead times for the components required for a single satellite as quoted by potential suppliers or, where no lead time is provided, based upon conservative estimates. As can be seen the maximum procurement lead time is 24 weeks.

Table 12: Table detailing procurement stages of the satellite components.

Item	**Lead Time (weeks)**
Liquid Fuel, Oxidiser and Helium Tanks	12
Attitude Sensor	4
Command and Data	4
Communications Transceiver	2
Rocket Motor	24

Solar Panels	16
Reaction Wheels	24
Heating Element	4
Chassis Materials	4
Solar Array Deployment Mechanism	8
Battery	4
Antennas	8
Hydrogen Fuel Cell	8
Hydrogen Generator	8

4.3.2. Manufacturing

The sub-assemblies and their sub-components required to manufacture a single satellite are shown in the table below. Assemblies relate to the design sections shown in section 3. Assembly times are estimated based on the complexity of the operation required.

Table 13: Details of the assemblies required to build a complete satellite. Level 1 is the top level finished unit.

Level	Assembly Name	Sub-Components	Assembly Time (hrs)
4	Chassis Frame	Chassis Materials	3
3	Equipped Chassis Frame	Chassis Frame Transceivers Reaction Wheel Heating Element Hydrogen Generator Hydrogen Fuel Cell Batteries	1
3	Base Panel	Chassis Materials Inter-Satellite Antenna	2

		Altitude Camera Rocket Motor Liquid Fuel Tank Oxidiser Fuel Tank Helium Fuel Tank	
3	Roof Panel	Chassis Materials Inter-Satellite Antenna Altitude Camera	1
3	Wall Panels	Chassis Materials Ground Antenna	2
2	Satellite Body	Equipped Chassis Frame Base Panel Roof Panel Wall Panels	2
2	Solar Cell Assemblies	Solar Panels Solar Array Deployment Mechanisms	3
1	Satellite	Satellite Body Solar Cell Assemblies	2

From table 13 the Gantt Chart for the assembly of a single satellite may be produced.

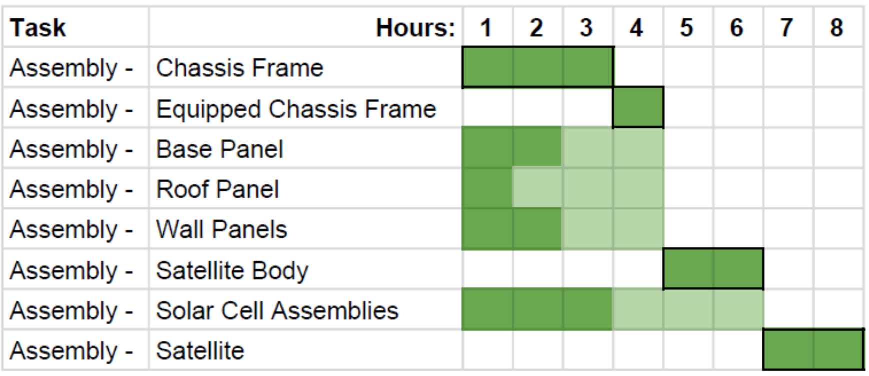


Figure 14: Gantt Chart of the manufacturing process for a single satellite. Dark green indicates the process time and light green any float available. The critical path is highlighted with outlines.

As can be seen from the Manufacturing Gantt Chart (figure 14), the minimum assembly time for a single satellite is estimated to be 8 hours, assuming a sufficiently large workforce is employed to allow simultaneous operations.

As 1392 satellites are required for the entire constellation, a flow production method will be employed, where each assembly operation has a dedicated workstation and the sub-assemblies are passed on to the next work station once completed. This method would allow a satellite to be produced every 3 hours based on the longest individual operation. To allow for a margin of error in the assembly operation times, it will be assumed that a satellite could be produced every 5 hours using the method described.

This is production rate is considerably shorter when compared to the Iridium satellite production rate of 4.5 days. However, Iridium satellites are considerably larger and more complex with an in-orbit mass of 689 kg and fewer off-the-shelf components [3], meaning the estimated production rate is reasonable.

Each launch will carry 203 satellites, which would have a total manufacturing time of 1015 hours. Assuming a working week of 40 hours, this would result in a manufacturing lead time of 26 weeks per launch.

4.3.3. Launch

From section 2, launch is to be carried out by SpaceX's Falcon Heavy carrying 203 satellites per launch. SpaceX aim to produce "a Falcon 9 first stage or Falcon Heavy side booster every week and an upper stage every two weeks" [16], giving the total production time of a Falcon Heavy as at least 5 weeks. Preparation for launch would take approximately further 3 weeks, which the payload of satellites would be required for.

4.3.4. Maintenance

The 29 satellites required per plane includes allowance for failures, meaning no additional launches would be required within the systems lifetime.

4.4. System Schedule

As the ground stations are required to make the data links for the satellites and not for control purposes, satellite launches can be carried out before the ground segment is fully completed.

4.4.1. Minimum Configuration

Due to the nature of the constellation structure discussed in section 2, the absolute minimum configuration for a functioning data link is a single orbital plane of 29 satellites. This would be unsuitable for customer usage as coverage would be minimal and extremely intermittent as Earth's rotation causes the area covered by the single orbital plane to change rapidly. However, a single orbital plane can be used for systems and performance checks before the full constellation is launched.

Whilst the absolute minimum configuration would be a single orbital, each Falcon Heavy launch would carry the satellites for 7 orbital planes. Therefore, in reality the minimum configuration is 7 orbital planes. This is still only adequate for testing purposes.

Prerequisites for the minimum configuration launch are the development lead time of 104 weeks, the procurement lead time of 24 weeks, the manufacturing lead time of 26 weeks and the launch preparation time of 3 weeks. Considering these lead times, time taken to launch the initial configuration would be 157 weeks or just over 3 years. In this time, the majority of ground stations would have been completed, meaning testing could be carried out as soon as each satellite is on-station.

4.4.2. Full Configuration

Before the system is made available for customer use, each of the 29 orbital planes must be launched and in orbit, providing continuous ground coverage within the latitudes defined in section 2. Therefore, the aim must be to have the full constellation operational in a short time period so the system can start generating revenue.

Following the initial launch of the first 203 satellites, subsequent launches could be carried out every 26 weeks, as procurement can begin before the previous launch, and manufacture can begin as soon as the previous batch are completed (3 weeks before launch). Therefore, the full constellation of 1392 satellites could be in orbit 156 weeks after the first launch, or 313 weeks (just over 6 years) after the project start, by which time the ground segment will be fully functional. These values are based upon a number of assumptions, therefore a time of 6.5 years from project start to full service is anticipated.

5. Costing Plan

5.1 Satellite Hardware

In order to achieve the large quantity of satellites required to achieve the full constellation for the satellite hardware a large portion of off the shelf components were selected. Therefore, reducing the in house manufacturing to mainly production of the final satellite by constructing the sub systems. Thus, reducing the overall cost of the constellation.

Table 14: Cost breakdown for satellite components.

Subsystem	Item	Cost ($)	Quantity	Total Cost ($)
Power				
	SolarTech HF315-6-36b Solar Panel [17]	1512	1	2104704
	BA0x High Energy Density Battery Array [18]	5800	14	113030400
	Helium	200	1	278400
	Horizon 3000W PEM Fuel Cell [19]	15484	1	21553728
	Peak Precision Hydrogen generator [20]	7163	1	9970896
Thermal				
	Fralock Adhesive Heater [21]	10000	1	13920000

	Thermacore Aluminium Heat Pipe [22]	5000	1	6960000
Attitude control				
	Blue Canyon RWP100 Reaction wheel [23]	26956	4	150091008
	CubeSense Attitude Sensor [24]	5371	1	7476432
Chassis				
	Structure	10000	1	13920000
Communications				
	HackRF Software Defined Radio	368	126	64544256
	Antennas	200	2	556800
	Cube Computer Command & Data [25]	5036	1	7010112
Propulsion				
	Airbus 200N bi-propellant thruster [26]	6800	1	9465600
	N2O4 Oxidiser [27]	367.82	1	512004.89
	Hydrazine Hydrate [28]	6571.00	1	9146832.61

Note: Total cost is for 1392 satellites.

Where available current suppliers have been used for accurate cost estimates due to the large quantities to be ordered costs of these components may reduce when bought in bulk for large scale production.

For the thermal systems, structures and antennas pricing was unavailable so estimates were made based on the quantity required, materials and manufacturing needed.

5.2 Launches

Launches of the satellites was to be undertaken by Falcon Heavy via the capture and return method and as stated in section 3 the cost per satellite for launch is $367,350 and decommissioning is $395,610.

5.3 Ground Stations

Finally, 120 ground stations will be needed for global connectivity. This will require design, surveying and construction. A cost estimate was determined from NBN Co contract for Australian satellite ground stations [15], this was altered accounting for large scale construction and issues that can occur in worldwide connectivity. An estimate was determined of $1.2billion, in addition to maintenance costs per year of $100000 per ground station.

5.4 Total cost

Via cost analysis of the hardware and estimation of launch and ground station needs a total cost to complete full integration and decommission of $2.42 billion was found.

6. Risks

6.1 Physical Risks

6.1.1 Launch Failure

Launch failure is an eventuality that must be considered for any spacecraft. Though launch vehicles are generally increasingly reliable, there is always uncertainty, especially in untested technology. The SpaceX Falcon 9, similar to that suggested as a launch vehicle failed two minutes after take off in June 2015 [29] and in a pre launch test in 2016 [30], both resulting the loss of payload.

The SpaceX Falcon Heavy Launch vehicle is currently untested. Repeat failures are unlikely, though loss of just one launch will cost 203 satellites, approximately $255.5m. There is very little that can be done to mitigate the causes and effects of a launch failure, but it is highly unlikely. It may be financially preferable to insure the launches and operational life of the satellites, for example the insurance company XL Catlin can compensate up to $45 million per launch [31], though the cost of this specialised insurance is not available publically and would have to be negotiated with the insurance company.

6.1.2 Major Space Debris Strike

Currently 21,000 pieces of debris larger than 10 cm are being tracked by the United States Space Surveillance Network, much of this in LEO [32]. Meteoroids with a diameter greater than 5 cm can be tracked. Their trajectories are predictable and avoidance manoeuvres can be planned well in advance. Despite this, should collision occur with one of the satellites, the damage would be catastrophic and result in loss of the satellite. Due to the large amount of satellites in each plane and presence of redundant satellites, the disruption to coverage would be minimal, with adjacent satellites adjusting position in the plane to cover damaged satellites.

A single collision with a satellite produces more space debris which can be dangerous to other systems. For example, when an operational Iridium satellite collided with Kosmos-2251 and created 2000 further pieces of debris [33]. This cascading effect is particularly dangerous in a constellation, where 1392 spacecraft are at similar altitudes on the same plane.

As with launch failure, this could be covered by space insurance.

6.1.3 Minor Space Debris Strike

Meteoroids with a diameter of less than 5 cm cannot be tracked from earth and so present a randomly distributed risk to satellites. Any exposed part of the satellite should be designed to withstand these minor collisions as general wear and tear, though some collisions could disable subsystems. If possible, the satellite would then be disposed of as outlined in section 3.7.4, and the coverage distributed to the adjacent satellites. Despite the high probability of the system encountering minor strikes, the consequences are minimal and as such this is not a significant risk.

6.1.4 Loss of Ground Station

The system requires a high number of ground stations to support the bandwidths required. These will need to be spread across the globe and may need to be located in less diplomatically secure countries. Locations will be selected to minimise risk from political instability, infrastructure damage, and conflict. Redundant ground stations are expensive and so if a ground station is lost, or unavailable due to a power outage, adjacent ground stations should be able to handle the increased demand without failing.

6.1.5 Summary

Table 15: Summary risk assessment of design.

Risk	**Probability**	**Severity**	**Overall Risk to Constellation**
Launch failure	**Low**	**High**	**Medium**
Major Debris Strike	**Low**	**Medium**	**Low-Medium**
Minor Debris Strike	**High**	**Low**	**Low**
Ground Station Loss	**Medium**	**Low**	**Low-Medium**

6.2 Project Risks

6.2.1 Delays

Whilst unpredictable, delays are likely in both the testing and launch phases of the project. Minor delays of several weeks are unlikely to have large effects on the overall timescale of the project, but major delays of months or years could impact the efficacy of the technology. The constellation would not provide a reliable service if several planes are missing, so the launch phase is the most delicate. The first satellite will begin degrading as soon as it is placed in orbit so it is crucial that the last satellite is deployed as soon after this as possible.

6.2.2 Funding

The funding and business side of the system are not covered in this report, but it is noted that a potential risk to the system is the loss of funding at any time during the start-up and operation of the project.

7. Conclusion

To summarise, a satellite constellation of 1392 satellites was achieved. Via cost analysis and performance constellation the design is within budget and is close to desired 25 year lifetime by operating for 20 years. The constellation will be able to be implemented within 6.5 years to full coverage via launch scheduling.

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