Deep Space CubeSat Constellation System

CubeSats provide a low-cost alternative to static landers and mobile platforms in deep space, which allows for greater launch accessibility and scientific return. Their use in constellation systems allows for complete coverage over a planet or moon, which helps to reduce costs while performing large-scale research for supporting missions. This report shows the design process of a sample deep space CubeSat constellation system, starting with developing subsystem requirements and constraints using existing data from other constellations. Results of design optimization and simulations are shown using set design parameters. Calculations performed in this report can be replicated for other deep space CubeSat constellation designs.

Keywords: cubesat, constellation, deep-space, spacecraft, nanosatellite

Introduction

Until recently, two types of spacecraft have been considered for deep space missions: static landers and mobile platforms. Now nanosatellite technology, especially CubeSats, provide a low-cost alternative through greater launch accessibility and scientific return. The ability to carry small instruments through multiple small, disposable spacecraft, as well as access to a variety of terrains, make CubeSats ideal for such missions (Klesh & Castillo-Roez, 2012).

The same applies for CubeSat constellation systems, as well as the need for zonal or global coverage of a body such as the Earth. Traditional constellation design methods incorporate continuous or discontinuous coverage of a specific geographic region (Ulybyshev, 2008). Usually, the objectives behind using these systems include providing connectivity across a specific region or performing large-scale scientific research in weather patterns.

Though Earth-orbiting constellation systems are becoming more common, there are currently no existing deep space constellation systems. However, deep space nanosatellites are currently in development and are capable of performing such high-risk and high-science return missions (Klesh & Castillo-Roez, 2012). These missions can include communicating with rovers or other larger spacecraft on other planets or moons. However, these types of missions still do not cover as much area over a planet or moon due to the various spacecraft constraints such as size or speed.

Using CubeSat constellation systems in deep space missions can help cover larger regions of planets or moons. One CubeSat would act as a mother ship, while the others would act as daughter ships and cover different areas around the planet or moon (Wong, Kegege, Schaire, Bussey, & Altunc, 2016). The more satellites used in the system, the more the propellant mass in each satellite can be reduced, thus reducing the cost-to-orbit (Singh et al., 2020).
The objective of this project is to develop a constellation of CubeSats to augment a deep space mission, such as a mission to the outer solar system.

### Literature Review

This section will cover information on small satellites, including CubeSats, and their deep space missions. Additionally, it will cover various types of constellation systems and their respective missions.

#### Small Satellites Overview

Small satellites (smallsats) in general are classified as satellites with mass less than 180 kg, or the size of a large kitchen fridge. Table 1 shows the five main classes of smallsats:

<table>
<thead>
<tr>
<th>Class</th>
<th>Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Minisatellite</td>
<td>100 – 180</td>
</tr>
<tr>
<td>Microsatellite</td>
<td>10 – 100</td>
</tr>
<tr>
<td>Nanosatellite</td>
<td>1 – 10</td>
</tr>
<tr>
<td>Picosatellite</td>
<td>0.01 – 1</td>
</tr>
<tr>
<td>Femtosatellite</td>
<td>0.001 – 0.01</td>
</tr>
</tbody>
</table>

*Source: “What are SmallSats and CubeSats?” 2015.*

One example of a smallsat mission is Janus SIMPLEx, a Lockheed Martin mission which will use a dual small deep space spacecraft system to visit near-earth binary asteroids (“Small Spacecraft, Big Universe: Lockheed Martin Selected For The Next Phase Of A Small Spacecraft Mission,” 2019). The system consists of two small satellites weighing around 40 kg each and carrying visible and infrared cameras. One key mission requirement is to meet the planetary launch window, since it is a deep space mission. Some subsystem requirements are that their power systems must handle a range of Sun distances and that their telecommunications systems need to be able to transmit over long distances and be compatible with the Deep Space Network (“Small Spacecraft, Big Universe: Lockheed Martin Selected For The Next Phase Of A Small Spacecraft Mission,” 2019).

Nanosatellites are used for extremely low-cost missions and can be used to support larger missions to increase science return. Rather than using static landers or mobile platforms, they allow for an alternative approach for in-situ and close-proximity observations in a variety of terrains through a distribution of multiple small, disposable spacecraft across the surface. Until recently, they were only used in low-Earth orbit missions but can be used as instruments part of larger missions, particularly as interplanetary CubeSats (Klesh & Castillo-Rogez, 2012).
CubeSats Overview

CubeSats are especially popular because their fixed design makes launching easier and allows for a viable market for common components. They also have a low cost of entry into development and a low-cost of failure. One standard form factor is 1U, or a 10x10x10 cm cube weighing 1 or 2 kg. Table 2 shows the most common factors along with their mass, volume (before deployment), and power parameters:

<table>
<thead>
<tr>
<th>Standard Form Factor</th>
<th>Mass (kg)</th>
<th>Volume (cm³)</th>
<th>Power, fixed (W)</th>
<th>Power, deployed (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1U</td>
<td>1.5</td>
<td>10x10x10</td>
<td>~3</td>
<td>-</td>
</tr>
<tr>
<td>3U</td>
<td>4.5</td>
<td>10x10x30</td>
<td>~8</td>
<td>~25</td>
</tr>
<tr>
<td>6U</td>
<td>10</td>
<td>10x20x30</td>
<td>~14</td>
<td>~35</td>
</tr>
</tbody>
</table>


Typical CubeSat characteristics include low mass or inertia and low risk to the primary spacecraft. CubeSats can support science experiments, such as imaging and remote sensing and direct surface sampling, and they are often driven by science applications. Because of these characteristics, they can enable both alternatives and augmentations to larger missions to low Earth orbit, geostationary orbit, and deep space. Other parameters for CubeSat design include the following (Klesh & Castillo-Rogez, 2012):

- Solar Arrays (fixed and/or deployable)
- Battery
- Antenna (monopole/dipole, turnstile, patch, 1U dish)
- Communications (UHF/VHF, S-Band)
- Data Rates
- Attitude Control (passive, torquer, drag control, etc.)
- Command & Data Handling (software)
- Propulsion
- Deployable Instruments (antenna, panels, tethers, boom, solar sail)
- Demonstrated Lifetime
- Payload Instruments (camera, telescopes, etc.)

The parameters listed above depend on the size of the CubeSat. For example, a 1U CubeSat does not have a propulsion system, but a 3U or 6U CubeSat has a cold gas thruster, electric propulsion, and solar sail (Klesh & Castillo-Rogez, 2012). If the CubeSat is larger, then it has greater performance and can hold more instruments in its payload. A larger CubeSat can also carry a higher-powered battery and antenna.

However, CubeSats still have low technological readiness level (TRL) in deep space (Klesh & Castillo-Rogez, 2012; Nieto-Peroy & Emami, 2019). Key
technological developments are being made to support interplanetary missions, such as radiation-tolerant operation strategies, deep-space deployment, and nanosat surface mobility.

Deep Space CubeSat Mission Example

The NASA probe InSight traveled with two CubeSats, together called Mars Cube One (MarCO), in 2018. These CubeSats, each with a mass of 13.5 kg, represented the first mission to test CubeSats in deep space. The purpose was to serve as communications relays during InSight’s landing and to send data from the probe to the Mars Reconnaissance orbiter (Scharf, 2018).

The CubeSats carried two main scientific instruments: UHF Radio Receiver and X Band Radio Transmitter. Additionally, the CubeSats used a “reflectarray,” a flat antenna patterned to mimic a parabolic dish to help focus transmissions towards Earth (“MarCO (Mars Cube One),” 2019; Scharf, 2018). Critical spare parts, such as experimental radios, antennas, and propulsion systems, will be used in future deep space CubeSat missions (“MarCO (Mars Cube One),” 2019).

Constellation Systems Overview

Satellite constellation systems are used for communication, navigation, and remote sensing applications (Ulybyshev, 2008). Communication and navigation applications include Internet connectivity services and Global Positioning Systems (GPS), respectively. Remote sensing applications include research-quality science in systems such as the following:

- Space Technology 5 (ST5) Project
- the NASA Time-Resolved Observations of Precipitation Structure and Storm Intensity with a Constellation of SmallSats (TROPICS) mission
- NASA Hyperspectral CubeSat Constellation

These systems are ideal for objectives that require coverage for a large geographic region, since they consist of more than one satellites working together as a system. Of the three listed above, ST5 and TROPICS are actual missions; as of 2015, the hyperspectral CubeSat constellation is still a concept presented by NASA scientists at a conference.

ST5 Project Overview

The ST5 Project, launched in March of 2006 and decommissioned after 90 days, was a constellation of three microsatellites in a “string of pearls” orientation. Each spacecraft was around 25 kg in mass and 53 cm in diameter by 48 cm in height, and included the subsystems and components shown in Figure 1 below:
Figure 1. System decomposition of ST5 Project

Several components, such as the nutation damper from the guidance, navigation, and control (GN&C) subsystem and cold gas micro-thruster from the propulsion subsystem, are connected to those in other subsystems through interdisciplinary coupling. For example, both the nutation damper and cold gas micro-thruster are used for transportation purposes. The power subsystem, which operates under low voltage, is responsible for powering all of the components and other subsystems in each of the satellites; therefore, it is the most important subsystem in this constellation system (Carlisle & Webb, 2007).

The objective of the ST5 mission was to show that a constellation of microsatellites could perform research-quality science. In this case, the spacecraft observed changes in the Earth’s magnetic field, and in doing so, it met its objective. Instruments such as the battery, magnetometer, and ground system components were used in future missions.

Source: Carlisle & Webb 2007

NASA TROPICS Overview

The NASA TROPICS mission is a constellation system of six 3U CubeSats in three different low-Earth orbital planes (Braun et al., 2018). Each CubeSat included the subsystems and components shown in Figure 2 below:
Figure 2. System decomposition of NASA TROPICS mission.

The CubeSat bus houses all the other components within the unit, save for the solar panels. The motor controller and the instruments used for the GN&C subsystem help guide each CubeSat to gather data over a specific area using the radiometer payload cube (Braun et al., 2018). Therefore, in this case, the most important subsystem in this constellation system is the GN&C subsystem.

The main goal of the TROPICS mission is to provide weather observations of 3D atmospheric temperature and humidity, cloud ice and precipitation horizontal structure, and storm intensity (Braun et al., 2018). These observations will help analyze relationships between the different types of data and various weather patterns to provide more insight into storm processes such as hurricanes (“Time-Resolved Observations of Precipitation Structure and Storm Intensity with a Constellation of Smallsats,” 2015). Currently, the mission is under development.

NASA Hyperspectral CubeSat Constellation Overview

The main goal for a hyperspectral CubeSat constellation is to provide daily or diurnal coverage at any point on the Earth cost-effectively (Mandl et al., 2015). This concept was developed by team members of the Earth Observing 1 (EO-1) mission, which carried an imaging spectrometer to capture images of natural hazards for more than 15 years. Eventually, they needed a way to take the images daily, which proved impossible using EO-1.
The constellation system has two possible configurations. The first one would contain three CubeSats at one time, angled at 30 degrees each. The other would have three sets of 15 CubeSats observing select spots on any part of the Earth every 46 minutes. Each CubeSat would include the subsystems and off-the-shelf components shown in Figure 3 below:

Figure 3. System decomposition of NASA hyperspectral mission

![System decomposition diagram](source)

Source: Mandl et al. 2015

The CubeSat system presented in Figure 3 is similar to the two shown in Figures 1 and 2. All three contain a battery and EPS as part of its power subsystem, and all three have similar structural components (bus/box and solar panels). The performance of the nano-hyperspectrometer depends on the location of the CubeSats and data relay. Therefore, the two most important subsystems in this concept are communications and GN&C.

Components for the propulsion subsystem were not included in the conceptual design. However, each CubeSat could have a cold gas thruster due to its 6U size.

Constellation System Design and Coverage

There are two main types of constellation design methods: continuous and discontinuous global coverage. To design a constellation system, the following assumptions are made (Ulybyshev, 2008):

1. The Earth is considered a round body.
2. All satellites in a constellation will be at the same altitude with the same number of satellites in each orbit plane.
3. All orbit planes in a constellation will have the same orbit inclination.
In simple continuous global coverage, at least one satellite is visible above a minimum elevation angle at every point on the Earth’s surface. Complex continuous coverage can either be full or partial region coverage, useful for telecommunications and navigation purposes. Theoretical designs such as the Draim or Walker constellations can achieve ideal continuous global coverage, which use four satellites and circular orbits, respectively (Singh et al., 2020; Ulybyshev, 2008). The Draim constellation design, especially, can incorporate perturbations only if there is a common period and similar inclination between the satellites.

On the other hand, simple discontinuous coverage implies that every point on the surface is viewed with a predefined revisit time, which ranges from a few minutes to a few hours. Similarly, complex discontinuous coverage can either be full or partial region coverage with revisit times. These types of constellation systems have become more prevalent only within the last decade, but they have proved to be especially useful for remote sensing applications or weather observations (Sarno, Graziano, & D’Errico, 2016; Ulybyshev, 2008).

Methodology

The methodology is as follows:

- Determine mission and subsystem requirements based on literature review presented, analysis, and interdisciplinary coupling between subsystems
- Develop high level designs of the top five subsystems based on the requirements already determined
- Perform design of experiments to identify key parameters and performance drivers
- Develop simulations of the constellation system

Design of CubeSat Subsystems

Based on system decompositions presented earlier, the common subsystems are as follows:

- Power
- Communications
- GN&C
- Structures
- Payload

Therefore, a deep space CubeSat constellation should include them. Coincidentally, those subsystems are all the ones listed for the two CubeSat constellation systems; the ST5 system also contained the propulsion and
thermal subsystems. Table 3 shows the different components for each subsystem in each CubeSat constellation presented earlier:

**Table 3. Comparison of subsystems in different CubeSat constellations.**

<table>
<thead>
<tr>
<th>Constellation System Components</th>
<th>ST5 System</th>
<th>NASA Tropics</th>
<th>NASA Hyperspectral</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Subsystem</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Power</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Battery</td>
<td>Battery</td>
<td>Battery</td>
<td>Battery</td>
</tr>
<tr>
<td>Solar array</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>CULPRiT chip</td>
<td></td>
<td>EPS (not including battery)</td>
<td>EPS (not including battery)</td>
</tr>
<tr>
<td>Electronics card</td>
<td></td>
<td>EPS (not including battery)</td>
<td></td>
</tr>
<tr>
<td><strong>Communications</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>X-band antenna</td>
<td>UHF monopole antenna</td>
<td>S-band radio</td>
<td></td>
</tr>
<tr>
<td>X-band transponder</td>
<td>RF reflector antenna system</td>
<td>CHREC space processor</td>
<td></td>
</tr>
<tr>
<td>Amplifier</td>
<td>Radio</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>GN&amp;C</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Nutation damper</td>
<td>Avionics interface board</td>
<td>GPS receiver</td>
<td></td>
</tr>
<tr>
<td>Mini sun spinning sensor</td>
<td>ADCS reaction wheel system</td>
<td>Attitude determination system</td>
<td>ADCS interface board</td>
</tr>
<tr>
<td><strong>Structures</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Satellite bus</td>
<td>2U bus</td>
<td>6U box</td>
<td></td>
</tr>
<tr>
<td>Solar array</td>
<td>Solar panels</td>
<td>Solar panels</td>
<td></td>
</tr>
<tr>
<td><strong>Payload</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Mini magnetometer</td>
<td>Radiometer payload cube</td>
<td>Nano-hyperspectrometer</td>
<td></td>
</tr>
<tr>
<td>Deployment boom</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Other instruments</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

The subsystem components, especially the payload instruments, depend on the scope of the mission. Since the ST5 system comprises microsatellites, each satellite contains far more components in each subsystem than a CubeSat. However, there are some similarities between the components used in the three different systems. For example, all three contain the following:

- Battery
- Satellite bus/box
- Solar array

Additionally, the communications and GN&C subsystems had similar functions, even though the components themselves were different. The NASA Tropics and Hyperspectral systems especially have very similar components in their GN&C subsystem, such as an attitude determination system. Nevertheless, these are not part of deep space missions. Since deep space CubeSats require more instruments and additional subsystems, their size would be at least 6U. Therefore, based on the information presented in the literature...
review, one example of a deep space CubeSat system decomposition can be as shown in Figure 4.

Figure 4. System decomposition for deep space CubeSats

The system decomposition in Figure 4 is similar to those shown earlier, save for their payload subsystems. Therefore, it shall be used as the system decomposition for this mission. The N2 diagram shown in Figure 5 shows the relationship between the subsystems.

Figure 5. N2 diagram of CubeSat constellation system
Figure 5 above shows the inputs and outputs for the subsystems. For example, one input for the structures subsystem is the power in daylight, which is an output from the power subsystem.

**Power Subsystem**

The power subsystem contains the battery and solar arrays. The solar arrays provide power generation, and the battery provides power storage. Together, along with wires for power conversion and distribution, they form an EPS.

**Power Generation from Solar Arrays**

There are two main types of solar arrays: deployed and fixed. The power generation from the solar panels depends on the amount of sunlight received. Equation 1 is used to determine the amount of power needed from the solar array during daylight to power the spacecraft in orbit. Assume that battery charging losses are excluded.

\[
P_{sa} = \left( \frac{P_e T_e X_e}{X_d} + \frac{P_d T_d X_d}{X_d} \right) \frac{1}{T_d} (1)
\]

Inputs: \( P_e \) = Spacecraft’s power requirement during eclipse, \( P_d \) = Spacecraft’s power requirement during daylight, \( T_e \) = Period length of eclipse orbit, \( T_d \) = Period length of daylight orbit, \( X_e \) = Transmission efficiency during eclipse, \( X_d \) = Transmission efficiency during daylight

Output: \( P_{sa} \) = Power needed from solar array during daylight

The efficiencies \( X_e \) and \( X_d \) depend on the type of power regulation, shown in Table 4.

<table>
<thead>
<tr>
<th>Type of Power Regulation</th>
<th>Transmission Efficiencies</th>
<th>Direct Energy Transfer</th>
<th>Peak-Power Tracking</th>
</tr>
</thead>
<tbody>
<tr>
<td>X_e</td>
<td>0.65</td>
<td>0.6</td>
<td></td>
</tr>
<tr>
<td>X_d</td>
<td>0.85</td>
<td>0.8</td>
<td></td>
</tr>
</tbody>
</table>


Efficiencies corresponding to direct energy transfer are larger because peak-power tracking requires a power converter between the solar arrays and the loads.

The beginning-of-life power per unit area \( P_{BOL} \) is calculated using Equation 2.
\[ P_{BOL} = P_O l_d \cos(\theta) \]  

(2)

Inputs: \( P_O \) = Ideal solar cell performance,  
\( l_d \) = Inherent degradation, 0.77 (nominal),  
\( \theta \) = Sun incidence angle, 23.5° for worst case, 90° for maximum power  
Output: \( P_{BOL} \) = Beginning-of-life power per unit area  
The ideal solar cell performance \( P_O \) depends on the type of solar cell used.  

By multiplying the solar flux with the efficiency value corresponding to the desired cell type in Table 5, the value for \( P_O \) can be found.  

Table 5. Efficiencies for different types of solar cells

<table>
<thead>
<tr>
<th>Cell Type</th>
<th>Efficiencies</th>
</tr>
</thead>
<tbody>
<tr>
<td>Silicon</td>
<td>0.148</td>
</tr>
<tr>
<td>Gallium Arsenide (GaAs)</td>
<td>0.185</td>
</tr>
<tr>
<td>Indium Phosphide</td>
<td>0.18</td>
</tr>
<tr>
<td>Multijunction GaInP/GaAs</td>
<td>0.22</td>
</tr>
</tbody>
</table>


The life degradation of the solar array \( L_d \) can be found using Equation 3.

\[ L_d = (1 - \text{degradation})^{\text{satellite life}} \]  

(3)

Inputs: Degradation = solar cell degradation/year, 0.375 for silicon, 0.275 for GaAs,  
Satellite life = usually length of the mission  
Output: \( L_d \) = life degradation of the solar array  
The end-of-life power per unit area \( P_{EOL} \) is calculated with Equation 4.

\[ P_{EOL} = P_{BOL} L_d \]  

(4)

Inputs: \( P_{BOL} \) = Beginning-of-life power per unit area,  
\( L_d \) = life degradation of the solar array  
Output: \( P_{EOL} \) = End-of-life power per unit area  

Power Storage from Batteries

There are two types of battery cells: primary and secondary. Primary batteries are useful for short missions and tasks that use very little power. Secondary batteries are useful for long-term missions and can provide power during eclipse periods. Therefore, the CubeSats in this project’s constellation system will use secondary batteries. The required battery capacity \( C_r \) for each spacecraft can be estimated using Equation 5.

\[ C_r = \frac{P_e \tau_e}{(DOD)N_n} \]  

(5)
Inputs: $P_e =$ Spacecraft’s power requirement during eclipse,  
$\tau_e =$ Eclipse duration  
DOD = depth of discharge, ranges from .20 to .80  
$N =$ number of batteries, between 2 and 5  
n = transmission efficiency of the battery  

Output: $C_r =$ required battery capacity

The mass of the batteries can be calculated by dividing the battery capacity $C_r$ by the specific energy density. Table 6 shows the specific energy densities for the different types of secondary battery couples.

### Table 6. Specific Energy Densities for Different Types of Secondary Batteries

<table>
<thead>
<tr>
<th>Secondary Battery Type</th>
<th>Specific Energy Density (W-hr/kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nickel-Cadmium (NiCd)</td>
<td>25 - 30</td>
</tr>
<tr>
<td>Nickel-Hydrogen, individual pressure vessel design (NiH)</td>
<td>35 - 43</td>
</tr>
<tr>
<td>Nickel-Hydrogen, common pressure vessel design (NiH)</td>
<td>40 - 56</td>
</tr>
<tr>
<td>Nickel-Hydrogen, common pressure vessel design (NiH)</td>
<td>43 - 57</td>
</tr>
<tr>
<td>Lithium-Ion (Li-Ion)</td>
<td>70 - 110</td>
</tr>
<tr>
<td>Sodium-Sulfur (NaS)</td>
<td>140 - 210</td>
</tr>
</tbody>
</table>


Communications Subsystem

The communications subsystem primarily includes an antenna system (antenna, transmitter, and receiver), as well as software to transmit data between the CubeSats and to and from Earth. The transmitter sends downlink signals to the other satellites and the ground station. The receiver gets uplink signals from other satellites and the ground station.

The amount of data can be calculated using Equation 6.

$$DQ = \frac{DR \times (F \times T_{max} - T_{initiate})}{M} \quad (6)$$

Inputs: DR = data rate,  
F = fractional reduction in viewing time, greater than 0.5  
$T_{max} =$ maximum time in view,  
$T_{initiate} =$ time required to initiate a communications pass, 2 minutes  
M = margin needed to account for missed passes, estimated at 2 to 3

Output: DQ = quantity of data

Equation 7 can be used to size a data link.

$$\frac{E_b}{N_o} = 10 \log \left( \frac{P \cdot G_t \cdot G_r \cdot L_d \cdot L_a}{kT_0(DR)} \right) \quad (7)$$

Inputs: P = transmitter power,
L₁ = transmitter-to-antenna line loss,
G₁ = transmit antenna gain,
Lₛ = space loss,
Lₐ = transmission path loss,
Gᵣ = receive antenna gain,
k = Boltzmann’s constant, 1.380 x 10⁻²³ J/K,
Tₛ = system noise temperature,
DR = data rate

Output: Eᵦ/No = ratio of received energy-per-bit to noise density, usually 50-100

The expression PL₁G₁ can also be expressed as the effective isotropic radiated power EIRP, which is related to the power flux density Wᵣ. For an S-band antenna system, Wᵣ is -137/4 kHz with the following uplink and downlink frequency ranges (Larson & Wertz, 1999):

- Uplink: 2.65 – 2.69 GHz
- Downlink: 2.5 – 2.54 GHz

Likewise, for an X-band antenna system, Wᵣ is -142/4 kHz with the following uplink and downlink frequency ranges (Larson & Wertz, 1999):

- Uplink: 7.9 – 8.4 GHz
- Downlink: 7.25 – 7.75 GHz

For all of the uplink and downlink frequencies shown above, the system noise temperatures Tₛ are as follows (Larson & Wertz, 1999):

- Uplink: 614 K
- Downlink: 135 K

Also, based on the frequencies, the transmission path loss Lₐ is -0.3 dB (Larson & Wertz, 1999).

The space loss Lₛ, antenna pointing loss L₀, and antenna half-power beamwidth θ can be found using Equations 8, 9, and 10.

\[ Lₛ = 20 \log \left( \frac{c}{\pi S f} \right) \]  \hspace{1cm} (8)
\[ L₀ = -12 \left( \frac{c}{\theta} \right)^2 \]  \hspace{1cm} (9)
\[ \theta = \frac{21}{f GH₄ D} \]  \hspace{1cm} (10)

Inputs: c = speed of light, 3 x 10⁸ m/s
S = path length,
f = frequency in GHz,
e = pointing error,
D = antenna diameter

Outputs: Ls = space loss,  
Lθ = antenna pointing loss,  
θ = antenna half-power beamwidth

**GN&C Subsystem**

The GN&C subsystem mainly contains an ADCS system and a navigation system. The ADCS system contains sensors and actuators such as reaction wheels or torquers. The navigation system, similar to the communications subsystem, includes a receiver and transmitter. This section will focus more on the ADCS system design.

Each spacecraft will use a spin stabilization control type; the entire spacecraft rotates while the angular momentum vector remains fixed in space. Reaction wheels in each ADCS system will act as the actuators to adjust orientation. They spin in either direction, while providing one axis of control for each wheel. The following shows the typical specifications for reaction wheels (Larson & Wertz, 1999).

- Weight: 2 – 20 kg
- Power: 10 – 110 W
- Performance Range: 0.01 to 1 N-m (max torque)

The sizing of the reaction wheels can be determined in two different ways. One way is to account for disturbance rejection in Equation 11.

\[
T_{RW} = T_{D}(MF)
\]  

Inputs: T_D = worst-case disturbance torque,  
MF = margin factor  
Output: T_{RW} = reaction-wheel torque due to disturbance rejection

The disturbance torque T_D is caused by gravity gradients, solar radiation, magnetic fields, and aerodynamic disturbance. This is shown in Equation 12 (Larson & Wertz, 1999).

\[
T_D = T_G + T_{sp} + T_m + T_a
\]  

Inputs: T_G = maximum gravity torque,  
T_{sp} = solar radiation torque,  
T_m = magnetic torque,  
T_a = aerodynamic torque

Output: T_D = worst-case disturbance torque  
Another way to size the reaction wheels is to find the slew torque T_{slew} using Equation 13.

\[
T_{slew} = \frac{4\theta I}{t^2}
\]  

Inputs: \(\theta\) = slewing angle, 30 degrees,  
I = moment of inertia of CubeSat,  
t = minimum maneuver time

Output: T_{slew} = slew torque
The wheel momentum $h$ can be estimated using Equation 14.

$$h = \frac{T_D \tau}{4\sqrt{2}}$$  \hspace{1cm} (14)  

Inputs: $T_D =$ disturbance torque,  
$\tau =$ orbit period  
Output: $h =$ wheel momentum

Both Sun sensors and gyroscopes (inertial sensors) will also be used to supplement the CubeSats’ navigation system. Sun sensors use visible light to measure angles between their mounting base and incident sunlight. Gyroscopes are used to measure the speed or angle of rotation from an initial reference. The following shows characteristics for both (Larson & Wertz, 1999).

- Weight Range: 1 to 15 kg (gyro), 0.1 to 2 kg (Sun sensor)
- Power: 10 to 200 W (gyro), 0 to 3 (Sun sensor)
- Performance: Gyro drift rate 0.003 to 1 deg/hr, sun sensor accuracy 0.005 to 3 deg

Propulsion Subsystem

The main component in the propulsion subsystem is the cold gas thruster due to its low cost and simplicity. The propellant (N2, NH3, Freon, He) performance characteristics are as follows (Larson & Wertz, 1999):

- Vacuum specific impulse ($I_{sp}$): 50-75 seconds
- Thrust range ($F$): 0.05-200 N
- Average bulk density: 0.28, 0.60, and 0.96 g/cm$^3$

This subsystem would only maintain attitude control and orbit maintenance and maneuvering. Therefore, staging will not be considered for this project. Since a single thruster has such a low specific impulse, multiple will need to be used. Figure 6 shows a schematic of a single cold gas thruster propulsion system.

Figure 6. Schematic of Cold Gas Propulsion System

Source: Anis 2012.
The propulsion system consists of a gas tank, gas feed line, valve, and thruster. The gas tank is either cylindrical or spherical in shape. The thruster itself is a convergent-divergent nozzle. Key assumptions for the design include:

- The flow is isentropic, quasi, and one-dimensional
- The propellant is an ideally, thermally, calorically perfect gas

Equation 15 is used to find the thrust generated by the system.

\[ F = \dot{m}V_e + A_e P_e \]  

Inputs: \( \dot{m} = \) propellant mass flow rate, 
\( V_e = \) propellant exhaust velocity, 
\( A_e = \) nozzle exit area, 
\( P_e = \) gas pressure at nozzle exit 
Output: \( F = \) thrust

The thrust is used to find the specific impulse, shown in Equation 16.

\[ I_{sp} = \frac{F}{\dot{m}g} \]  

Inputs: \( F = \) thrust, 
\( \dot{m} = \) propellant mass flow rate, 
\( g = \) acceleration due to gravity, 9.81 m/s² 
Output: \( I_{sp} = \) specific impulse

The propellant exhaust velocity \( V_e \) can be found using Equation 17.

\[ V_e = \sqrt{\frac{2\gamma}{\gamma-1} \frac{R_A}{M_W} T_c \left( 1 - \left( \frac{P_e}{P_c} \right)^{\gamma-1} \right)} \]  

Inputs: \( \gamma = \) specific heat ratio for gas, 
\( R_A = \) universal gas constant, 8.314 J/K-mol, 
\( M_W = \) molecular weight of gas, 
\( T_c = \) chamber temperature, 
\( P_e = \) gas pressure at nozzle exit 
\( P_c = \) chamber pressure 
Output: \( V_e = \) propellant exhaust velocity

The nozzle’s area expansion ratio \( \varepsilon \) can be found using Equation 18.

\[ \varepsilon = \frac{A_e}{A_t} \]  

Inputs: \( A_e = \) nozzle exit area, 
\( A_t = \) nozzle throat area 
Output: \( \varepsilon = \) area expansion ratio

The characteristic velocity \( c* \) and thrust coefficient \( c_f \) can be found using Equations 19 and 20. They are used to evaluate the propulsion system performance.

\[ c^* = \frac{P_c A_t}{\dot{m}} \]  

\[ c_f = \frac{F}{P_c A_t} \]  

Inputs: \( P_c = \) chamber pressure,
At = nozzle throat area,
\( m \) = propellant mass flow rate
\( F \) = thrust,
\( P_c \) = chamber pressure,
\( A_t \) = nozzle throat area
Outputs: \( c^* \) = characteristic velocity,
\( c_r \) = thrust coefficient

Structures Subsystem
Basic analysis for the structures subsystem will mainly be done on the CubeSat box, cold gas thruster tank, and solar arrays.

CubeSat Box
The CubeSat box will have a monocoque structure and will be made of aluminum 7075-T73 sheets. This type of material is most often used for spacecraft due to its easy machinability, ductility, low density, and high strength versus weight. The following material properties will be useful in conducting structural analysis for the main CubeSat box (Larson & Wertz, 1999).

- Young’s Modulus: \( E = 71 \times 10^9 \) N/m²
- Poisson’s Ratio: \( \nu = 0.33 \)
- Density: \( \rho = 2.8 \times 10^3 \) kg/m³
- Ultimate Tensile Strength: \( F_{tu} = 524 \times 10^8 \) N/m²
- Yield Tensile Strength: \( F_{ty} = 448 \times 10^6 \) N/m²

Assuming uniform thickness, Equations 21 and 22 can be used to find the axial frequency \( f_{nat,a} \) and lateral frequency \( f_{nat,l} \).

\[
\begin{align*}
    f_{nat,a} &= 0.250 \frac{AE}{m_B L} \\
    f_{nat,l} &= 0.560 \frac{EI}{m_B L^3}
\end{align*}
\]

Inputs: \( A \) = cross-sectional area of CubeSat box,
\( E \) = Young’s Modulus,
\( m_B \) = estimated distributed mass of CubeSat box,
\( L \) = length of CubeSat box,
\( I \) = area moment of inertia
Outputs: \( f_{nat,a} \) = axial frequency,
\( f_{nat,l} \) = lateral frequency

The estimated distributed mass of the CubeSat box \( m_B \) includes the box itself, as well as the components located inside.
The area moment of inertia for the CubeSat box can be found using Equation 23.
\[ I = \frac{bh^3}{12} \]  

(23)

The area moment of inertia, \( I \), can be found using the formula above. The limit load, or the equivalent axial load \( P_{eq} \), can be found using Equation 24.

\[ P_{eq} = m_B g (LF) + \frac{4M}{b} \]  

(24)

Inputs: \( m_B \) = estimated distributed mass of CubeSat box,  
\( g \) = acceleration due to gravity, \( 9.81 \text{ m/s}^2 \),  
\( LF \) = loading factor,  
\( M \) = bending moment,  
\( b \) = base of cross section

Output: \( P_{eq} \) = equivalent axial load

The bending moment used in Equation 24 can be calculated using Equation 25.

\[ M = m_B g y \]  

(25)

Inputs: \( m_B \) = estimated distributed mass of CubeSat box,  
\( g \) = acceleration due to gravity, \( 9.81 \text{ m/s}^2 \),  
\( y \) = moment arm, or distance from center of mass

Output: \( M \) = bending moment

The ultimate and yield equivalent axial loads (\( P_{eq,u} \) and \( P_{eq,y} \), respectively) can be found using their respective factors of safety \( FS_u \) and \( FS_y \), shown in Equation 26.

\[ P_{eq,u} \text{ or } P_{eq,y} = P_{eq} \times (FS_u \text{ or } FS_y) \]  

(26)

Inputs: \( P_{eq} \) = equivalent axial load,  
\( FS_u \) or \( FS_y \) = ultimate or yield factor of safety

Output: \( P_{eq,u} \) or \( P_{eq,y} \) = ultimate or yield equivalent axial load

The factors of safety depend on the type of structure being designed. Since the CubeSat box is being designed for strength and will be affected by temperature changes during the mission, the factors of safety will be as follows (Larson & Wertz, 1999):

- Ultimate factor of safety \( FS_u = 1.25 \)
- Yield factor of safety \( FS_y = 1.1 \)

The critical buckling load \( P_{cr} \) for the structure is calculated using Equation 27.

\[ P_{cr} = \frac{\pi^2 E I}{4 L^2} \]  

(27)

Inputs: \( E \) = Young’s Modulus,  
\( I \) = area moment of inertia,  
\( L \) = length of CubeSat box
Output: \( P_{cr} \) = critical buckling load

Both the ultimate equivalent axial load \( P_{eq} \) and critical buckling load \( P_{cr} \) are used in Equation 28 to determine the stability of the structure.

\[
MS = \frac{P_{cr}}{P_{eq}} - 1
\]  

(28)

Inputs: \( P_{cr} \) = critical buckling load,
\( P_{eq} \) = equivalent axial load

Output: MS = margin of safety

If the margin of safety MS is less than zero, then the structure is not adequate for the mission. The inputs to calculate the applied loads will need to be modified until the margin of safety MS is greater than or equal to zero.

The mass of the box’s shell \( m_{box} \), not including fasteners or attachments, is calculated using Equation 29.

\[
m_{box} = \rho V
\]  

(29)

Inputs:
\( \rho \) = density of material,
\( V \) = volume of CubeSat box only

Output: \( m_{box} \) = mass of CubeSat box shell only

Gas Tank

The structural design of the tank depends on its shape. As mentioned earlier, a tank for a cold gas thruster system is either cylindrical or spherical in shape. Equation 30 is used to find the spherical tank stress.

\[
\sigma = \frac{Pr}{2t}
\]  

(30)

Inputs: \( P \) = maximum expected operating pressure,
\( r \) = radius,
\( t \) = thickness

Output: \( \sigma \) = allowable stress

Cylindrical pressure vessels depend on both the longitudinal and hoop stresses. Equation 30 can be used to find the longitudinal stress of a cylindrical pressure vessel, and Equation 31 can be used to find the hoop stress.

\[
\sigma = \frac{Pr}{t}
\]  

(31)

Inputs: \( P \) = maximum expected operating pressure,
\( r \) = radius,
\( t \) = thickness

Output: \( \sigma \) = allowable stress
**Solar Arrays**

The size of the solar arrays can be found using values calculated for the power subsystem. Equation 32 is used to calculate the solar-array area $A_{sa}$ needed to support each spacecraft’s power requirement.

$$A_{sa} = \frac{P_{sa}}{P_{EOL}}$$  

Inputs: $P_{sa} =$ Power needed from solar array during daylight,  
$P_{EOL} =$ End-of-life power per unit area  
Output: $A_{sa} =$ required solar-array area  

The mass of the solar arrays can also be found using Equation 33.

$$m_a = 0.04P_{sa}$$  

Input: $P_{sa} =$ Power needed from solar array during daylight  
Output: $m_a =$ mass of solar arrays

**Payload Subsystem**

The payload subsystem is made up of different instruments and depends on the type of mission and mission objective. Therefore, it drives the mission requirements and specifications. For a deep space, remote sensing mission, the instruments would be required to perform imaging, intensity measurement, and topographic mapping. Therefore, the payload would include an imager/camera, radiometer, and altimeter used for analysis in deep space (Larson & Wertz, 1999).

The data rate $DR$ used as an input for the communications subsystem can be calculated using Equation 34.

$$DR = Z_C Z_A B$$  

Inputs: $Z_C =$ Number of cross-track pixels,  
$Z_A =$ Number of swaths in one second  
$B =$ Number of bits per pixel  
Output: $DR =$ data rate  

The number of cross-track pixels $Z_C$ and number of swaths $Z_A$ can be calculated using Equations 35 and 36.

$$Z_C = \frac{2\eta}{IFOV}$$  

$$Z_A = \frac{V_g \times 1 sec}{IFOV \times H \times \frac{\pi}{180°}}$$  

Inputs: $\eta =$ maximum sensor look angle,  
$V_g =$ spacecraft ground-track velocity,  
$IFOV =$ Instantaneous field of view,  
$H =$ orbit altitude  
Output: $Z_C =$ Number of cross-track pixels,  
$Z_A =$ Number of swaths in one second  

The instantaneous field of view $IFOV$ can be calculated using Equation 37.

$$IFOV = \frac{Y_{max} \left(\frac{180°}{\pi}\right)}{R_s}$$  

Inputs: $Y_{max} =$ maximum along-track ground sampling distance,  
$R_s =$ slant range  
Output: $IFOV =$ Instantaneous field of view
Orbital Mechanics and Constellation Design

In designing a constellation system, several key design factors need to be considered:

- Number of satellites
- Constellation pattern
- Altitude
- Number of orbit planes
- Minimum elevation angle
- Inclination
- Eccentricity
- Spacing between satellites

Of those listed above, the three most important are the altitude, inclination, and minimum elevation angle. The altitude and minimum elevation angle can be assumed as design variables, while the inclination depends on either analytical calculations or constellation pattern.

In designing the constellation structure, all assumptions listed in the literature review will be considered. Therefore, the following will be true for this project:

1. Since the planet or moon will be considered a round body, no oblate body perturbations will be incorporated. However, other perturbations such as solar radiation or finite burns may be incorporated depending on the mission.
2. All satellites will be at the same altitude, and each orbit plane will contain the same number of satellites.
3. All orbit planes will have the same inclination, either determined through analytical calculations or the chosen constellation pattern.

Forces and Perturbations

To better simulate reality, the following outside forces or perturbations must be considered in the simulations.

- Finite burns
- Atmospheric drag
- Solar radiation

Solar pressure and atmospheric drag can affect the values of the orbits’ Keplerian elements. Finite burns, however, have low impact on orbital changes but should still be considered (Curtis, 2014; Larson & Wertz, 1999).

Keplerian Elements

The satellites’ orbits are generally defined by the following Keplerian elements (Curtis, 2014):
Length of semi-major axis, $a$

- Eccentricity, $e$
- Argument of periapsis, $\omega$
- True anomaly, $\theta$
- Inclination, $i$
- Right ascension of ascending node, $\Omega$

The latter four elements are illustrated in Figure 7. Note that the equatorial plane shown below is of the Earth.

**Figure 7. Orientation of Orbit Plane with respect to Equatorial Plane**

For both types of coverage, continuous or discontinuous, the orbits can be assumed as circular, due to its more widespread use than elliptical ones (Ulybyshev, 2008). Therefore, $a$ is equal to the radius of the orbit, which is the sum of the altitude and the radius of the planet or moon. For circular orbits, $e$ is equal to zero.

$\theta$ and $\omega$ can be combined to obtain the argument of latitude $u$, which can range from 0 to 360 degrees (“Basic Concepts of Manned Spacecraft Design: Describing Orbits,” 2018). The orbital period $\tau$, shown in Equation 38, can be considered as another Keplerian element (Curtis, 2014).

$$\tau = 2\pi \sqrt{\frac{a^3}{GM}}$$  \hspace{1cm} (38)

Inputs: $a =$ length of semi-major axis,
$G =$ gravitational constant, $6.67 \times 10^{-20}$ km$^3$/kg·s$^2$,
$M =$ mass of the planet or moon

Output: $\tau =$ orbital period
The remaining two Keplerian elements depend on the type of constellation structure. The inclination also depends on the angular momentum, which is found using Equation 39.

\[ h = a \times v_{satellite} \]  

Inputs: \( a \) = length of semi-major axis, \( v_{satellite} \) = velocity of the satellite

Output: \( h \) = angular momentum of the satellite

### Constellation Structure

The structure of a constellation system is symmetric; therefore, there should be an equal number of satellites in each orbit plane. Usually, constellations with one or two planes are more responsive to changes based on user needs than those with more than two (Larson & Wertz, 1999). Therefore, it is better to place more satellites in a smaller number of planes rather than placing less satellites in a larger number of planes. Nevertheless, the structure and number of constellation planes should allow for maximum coverage with the smallest possible number of satellites.

There are two common types of constellation structures that use circular orbit patterns:

- Streets of Coverage pattern
- Walker-Delta pattern

Figure 8 shows an example of a Streets of Coverage pattern.

**Figure 8. Streets of Coverage Pattern, \( i = 90^\circ \)**

![Streets of Coverage Pattern](image)


Planes in the Streets of Coverage pattern all have the same inclination at 90 degrees, since it is a polar constellation (Hu, Maral, & Ferro, 2002). One main advantage is that most of the orbital planes in this pattern are evenly spaced, so they have similar \( \Omega \) values. The example shown in Figure 8 uses five planes in the pattern, all intersecting at the poles. The image shows the main disadvantage of this pattern; coverage is more concentrated in areas closer to the poles than the equator (Hu et al., 2002; Larson & Wertz, 1999).
Figure 9 shows an example of a Walker-Delta pattern.

**Figure 9.** Walker-Delta pattern, $i = 65^\circ$

Planes in the Walker-Delta pattern all have the same inclination relative to the equatorial plane, which allows for equally distributed coverage everywhere over the surface. The example shown in Figure 9 uses 15 satellites in total, distributed across five planes. Each plane is at an inclination of 65 degrees with respect to the equator.

However, one main disadvantage of the Walker-Delta pattern is its design constraints; both the orbital planes and satellites have to be equally spaced from each other, which adds more difficulty in optimizing the design (Hu et al., 2002). Therefore, for maximum coverage and simplicity, the constellation system for this project will use the Streets of Coverage pattern. Figure 10 shows a view of the example shown in Figure 8 from the north pole.

**Figure 10.** Streets of Coverage Pattern View from North Pole

Satellites traveling over half of the planet or moon travel northward, while those over the other half travel southward. Equation 40 is used to calculate the maximum perpendicular separation between two adjacent orbital planes going in the same direction.
\[ D_{maxS} = \lambda_{street} + \lambda_{max} \]  

(40)

Inputs: \( \lambda_{street} \) = street of coverage angle,  
\( \lambda_{max} \) = maximum central angle  
Output: \( D_{maxS} \) = maximum perpendicular separation between two adjacent orbital planes going in the same direction

There is a seam between the two halves in Figure 10, which has much narrower spacing than other areas in the pattern. The \( \Omega \) values for those two planes will be different from those of the other planes. Equation 41 is used to calculate the maximum perpendicular separation between two adjacent orbital planes going in opposite directions.

\[ D_{maxO} = 2\lambda_{street} \]  

(41)

Input: \( \lambda_{street} \) = street of coverage angle  
Output: \( D_{maxO} \) = maximum perpendicular separation between two adjacent orbital planes going in opposite directions

The “street of coverage” is an area or swath with continuous coverage that is centered on the ground track. The street of coverage angle, half of a swath, depends on both the space between satellites in each plane and the maximum central angle. This relationship is shown in Equation 42.

\[ \cos(\lambda_{street}) = \frac{\cos(\lambda_{max})}{\cos(\frac{\pi}{2})} \]  

(42)

Inputs: \( \lambda_{max} \) = maximum central angle,  
\( S \) = space between satellites in each plane  
Output: \( \lambda_{street} \) = street of coverage angle

The relationships between both the maximum central angle and space between satellites help determine whether coverage is continuous or discontinuous, which are as follows:

- If \( S < 2\lambda_{max} \), then coverage in the swath is continuous.  
- If \( S > 2\lambda_{max} \), then coverage in the swath is discontinuous.

Using the number of satellites per plane \( N_p \), Equation 43 can be used to find the revisit time \( t_{rev} \) for discontinuous coverage (Singh et al., 2020).

\[ t_{rev} = \frac{\tau}{N_p} \]  

(43)

Inputs: \( \tau \) = orbital period,  
\( N_p \) = number of satellites per plane  
Output: \( t_{rev} \) = revisit time

Results

The equations presented earlier will be used to determine key design parameters, which will help determine the best design scenarios for the system.
Additionally, the parameters of the constellation system will be used to generate simulations in Systems Tool Kit (STK). Results from sample calculations, which use Mars as the central body, will also be discussed.

**Analysis of CubeSat Subsystems**

**Power Subsystem Analysis**

The power subsystem depends on the spacecraft life, solar array characteristics, and orbit characteristics. Each CubeSat can be assumed to have a lifespan of two to four years to function in deep space. If the lifespan is higher, then both $L_d$ and $P_{BOL}$ are lower. Both $P_e$ and $P_d$ can be assumed to be a fixed power of 14 W, taken from literature review.

For all solar array calculations, peak-power tracking was used for power regulation. Therefore, $P_{sa}$ is higher than if direct energy transfer was used. Both $P_{BOL}$ and $P_{EOL}$ depend on the type of solar cell used. GaAs solar cells have the best efficiency and the lowest degradation, and they contribute to the best $P_{BOL}$ and $P_{EOL}$ results.

The batteries used for the CubeSats in this project are based on those used on MarCO. Li-Ion batteries are predominantly used on CubeSats today because they are rechargeable, light, and have a high specific energy density. Each CubeSat will contain two Li-Ion batteries with a total battery mass of around 0.7 kg.

The orbit characteristics depend on the central body. If the central body has a smaller angular radius, then the eclipse duration is smaller. Additionally, if the central body is farther from the Sun, then the solar flux is smaller. These characteristics are determined concurrently with simulation results.

Table 7 shows results from the sample calculations along with explanations for each.

<table>
<thead>
<tr>
<th>Results</th>
<th>Values</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>$P_{sa}$</td>
<td>41.027 W</td>
<td>Orbit around Mars, peak-power tracking</td>
</tr>
<tr>
<td>$P_{BOL}$</td>
<td>77.075 W</td>
<td>Mars solar flux, GaAs solar cell type</td>
</tr>
<tr>
<td>$L_d$</td>
<td>0.381 year</td>
<td>CubeSat life of 3 years</td>
</tr>
<tr>
<td>$P_{EOL}$</td>
<td>29.372 W</td>
<td></td>
</tr>
<tr>
<td>$C_r$</td>
<td>191.49 W-hr</td>
<td>Battery capacity for each battery</td>
</tr>
<tr>
<td>Mass of batteries</td>
<td>0.723 kg/battery</td>
<td>Li-Ion battery type</td>
</tr>
</tbody>
</table>

**Communications Subsystem Analysis**

The communications subsystem indirectly depends on orbit characteristics, since one of the design parameters is the payload’s data rate. The size and performance in the communications instruments will be based on similar ones in other CubeSat missions.

An X-band antenna system has higher frequency ranges for both uplink and downlink, but those frequencies can contribute to a higher $L_d$ if there is a
pointing error. However, this type of antenna system was used on MarCO, so it will be used in this mission.

Table 8 shows the results using the desired antenna system and other parameters.

**Table 8. Communications Subsystem Sample Results**

<table>
<thead>
<tr>
<th>Results</th>
<th>Values</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>DQ</td>
<td>27871800 bits</td>
<td>DR = 31529 bps (see Section 4.1.6), F = 0.77, maximum time is 1 hour</td>
</tr>
<tr>
<td>E_b/N_0 (uplink)</td>
<td>225 dB</td>
<td>Frequency = 7.9 GHz</td>
</tr>
<tr>
<td>E_b/N_0 (downlink)</td>
<td>231 dB</td>
<td>Frequency = 7.25 GHz</td>
</tr>
<tr>
<td>L_θ (uplink)</td>
<td>-16982 dB</td>
<td>D = 1 mm, pointing error of 0.0001</td>
</tr>
<tr>
<td>L_θ (downlink)</td>
<td>-14303 dB</td>
<td></td>
</tr>
</tbody>
</table>

The E_b/N_0 ratio for both uplink and downlink is more than two times higher than the desired value, probably due to the data rate. Still, it is higher for the downlink than the uplink. The downlink L_θ is less than that of the uplink.

**GN&C Subsystem Analysis**

The orbit characteristics, especially those of the central body, greatly impact the sizing of the reaction wheels of the GN&C subsystem. As mentioned earlier, their sizing is determined by T_RW or T_slew. The wheel momentum is found through T_D and T_slew.

If the central body has little to no impact on the CubeSat’s magnetic or aerodynamic torque, then T_slew should be used as a design parameter. Otherwise, T_RW must be compared with the maximum torque (0.01 to 1 N-m) to determine the better design parameter. If T_RW is less than the maximum torque, then T_slew must be considered as the main design parameter for the reaction wheel torque.

Table 9 shows T_RW, T_slew, and the wheel momentum for a CubeSat orbiting Mars.

**Table 9. GN&C Subsystem Sample Results**

<table>
<thead>
<tr>
<th>Results</th>
<th>Values</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>T_RW</td>
<td>1.098 x 10^6 N-m</td>
<td>Mainly influenced by solar radiation and aerodynamic torques</td>
</tr>
<tr>
<td>T_slew</td>
<td>1.396 x 10^6 N-m</td>
<td>Min maneuver time of 5 seconds</td>
</tr>
<tr>
<td>Wheel momentum</td>
<td>0.001433 N-m-s</td>
<td></td>
</tr>
</tbody>
</table>

As seen in the table above, T_RW is less than the maximum torque and T_slew. Therefore, T_slew should be the main design parameter for the reaction wheel torque. Additionally, the wheel momentum is very small; therefore, the mass of the reaction wheels should be less than 2 kg.
Propulsion Subsystem Analysis

The performance of the propulsion system depends on the propellant used. Therefore, the major design variables are the thrust, specific impulse, molecular weight, and specific heat ratio. Out of the four propellants mentioned, N2 and NH3 produce the best results because of their light weight and lower specific heat ratio. NH3 has the best of the two, since it produces the highest exhaust velocity. Therefore, the propulsion system in each CubeSat will use NH3 as its propellant, which has the following characteristics:

- $\gamma = 1.32$
- MW = 0.01703 kg/mol

The size and other characteristics of the cold gas thruster is based on similar models used on other CubeSats. For this project, the area ratio of the nozzle will be 50. Since a cold gas thruster has lower performance than other propulsion systems, the chamber temperature will be relatively low (between 200 and 320 K). The higher the chamber temperature, the higher the exhaust velocity.

Table 10 shows results using the selected propellant.

<table>
<thead>
<tr>
<th>Results</th>
<th>Values</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>$V_e$</td>
<td>790.5 m/s</td>
<td>Assume chamber and exit conditions to be $T_c = 300 , \text{K}$, $P_c = 800 , \text{kPa}$, $P_e = 40 , \text{kPa}$</td>
</tr>
<tr>
<td>$m$</td>
<td>-2.53 kg/s</td>
<td>$F = 0.05 , \text{N}$, $I_{sp} = 50 , \text{seconds}$</td>
</tr>
<tr>
<td>$c^*$</td>
<td>-316.213 m/s</td>
<td>Assume $A_t = 0.001 , \text{m}^2$, $A_e = 0.05 , \text{m}^2$</td>
</tr>
<tr>
<td>$c_f$</td>
<td>0.167</td>
<td></td>
</tr>
</tbody>
</table>

Structures Subsystem Analysis

Parameters for CubeSat structure depend on $m_B$, including the mass of the propulsion system, batteries, and other instruments. The mass of the solar arrays is not included, since they are not stored inside the box. The mass of the box shell is around 16.8 kg, so the distributed mass should be assumed greater than that. For calculations, $m_B$ is assumed to be around 25 kg.

Given a maximum operating pressure and gas tank thickness, a spherical tank would have a lower allowable stress than a cylindrical tank’s hoop stress. However, a spherical tank is more structurally stable than a cylindrical one, since there are no edges. Therefore, the gas tank for the cold gas thruster system will be spherical. Using an average bulk density of 0.96 g/cm³ and volume of the tank, the propellant mass is found to be around 0.025 kg.

Both the area and the mass of the solar arrays strongly depend on the design parameters of the power subsystem. The area depends on the type of solar cells used; GaAs solar cells contribute to a smaller area, which still makes them the best choice for the solar arrays. The mass, on the other hand, depends on the power required and orbit characteristics. Using the given power requirement, the area and mass are less than 2 m² and 2 kg, respectively.
Payload Subsystem Analysis

Payload instruments are based on similar ones in other CubeSat missions due to the size, weight, and power requirements. Therefore, the most important calculation was the data rate, which depends on the orbit altitude and spacecraft speed. This also influences other parameters in the constellation system. A higher spacecraft speed gives a higher number of swaths in one second, which then leads to a higher data rate. Assuming a $Y_{\text{max}}$ of 50 m and a slant range of 1200 m, the data rate was found to be around 32327 bps with 8 bits per pixel.

Analysis of Constellation System

The velocity and orbital period of each spacecraft affect subsystems such as communications, GN&C, and the payload. The scalar angular momentum can be found using the velocity. The following orbit parameters were determined earlier:

- Eccentricity, $e = 0$
- Inclination, $i = 90$ degrees

Other orbit parameters, $u$ and $\Omega$, are determined based on the size of the swaths and the space between the satellites. Using Mars as the central body and an orbit altitude of 500 km, the following parameters are found:

- $\tau = 7386$ seconds
- $v_{\text{satellite}} = 3.314$ km/s
- $h = 12914$ km²/s

The constellation system will have between three and five planes for maximum possible coverage and simplicity. Constellation design parameters such as the number of satellites and space between satellites depend on the desired amount of coverage and the radius of the central body. Each plane should have a minimum of nine satellites per plane, so that coverage within a swath can be continuous. A higher number of satellites per plane leads to lower revisit time.

The sample calculations assume 3 planes with 18 satellites in each. Table 11 shows the resulting parameters for the simulation.
Table 11. Sample Constellation Parameters, \( N_P = 18, 3 \) Planes

<table>
<thead>
<tr>
<th>Parameters</th>
<th>Values</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>( S )</td>
<td>20 degrees</td>
<td>For satellites within each plane</td>
</tr>
<tr>
<td>( t_{rev} )</td>
<td>410.3 seconds</td>
<td>For discontinuous coverage</td>
</tr>
<tr>
<td>( \lambda_{\text{max}} )</td>
<td>20.9 degrees</td>
<td>Assume 10° elevation angle, Coverage in swath is continuous</td>
</tr>
<tr>
<td>( \lambda_{\text{street}} )</td>
<td>18.4 degrees</td>
<td></td>
</tr>
<tr>
<td>( D_{\text{max,} S} )</td>
<td>39.26 degrees</td>
<td>Size of swaths away from seam</td>
</tr>
<tr>
<td>( D_{\text{max,} O} )</td>
<td>36.8 degrees</td>
<td>Size of swaths near seam</td>
</tr>
</tbody>
</table>

The parameters above will be used to generate a simulation. Other simulations will also be done to show constellations with different number of satellites per plane.

Simulations

This section presents three different simulations, each with different \( \Omega \) and \( u \) values for each orbit. The first simulation uses data presented in Table 11, while the other two are shown with their respective constellation data. All other inputs (i.e. CubeSat communications parameters) are the same for all three simulations.

Sample Constellation System

Using the parameters shown in Table 11, Figure 11 shows a top view of the constellation system at the start of the simulation.

Figure 11. Constellation system over Mars, \( N_P = 18, 3 \) planes
The image above shows that this constellation is only useful if the mission requires coverage over a specific area during specific times of the day. The area labeled as Place1 is seen by at most six CubeSats for various durations over 24 hours. Place1 is not seen by any CubeSats for around one hour and 40 minutes, when it passes between plane 3 and plane 1. This is because the space between the two orbits is larger than $D_{\text{max},S}$ and $D_{\text{max},O}$.

The three-plane constellation system is especially useful for follow-up missions to places such as Mars or the Moon, where certain areas are already known. Keeping all other parameters the same, five planes are more useful for complete coverage over the central body.

Second Constellation System

Parameters for another three-plane constellation model are shown in Table 12.

<table>
<thead>
<tr>
<th>Parameters</th>
<th>Values</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>$S$</td>
<td>20 degrees</td>
<td>For satellites within each plane</td>
</tr>
<tr>
<td>$t_{rev}$</td>
<td>410.3 sec</td>
<td>For discontinuous coverage</td>
</tr>
<tr>
<td>$\lambda_{\text{max}}$</td>
<td>24.73 deg</td>
<td>Assume 5° elevation angle, Coverage in swath is continuous</td>
</tr>
<tr>
<td>$\lambda_{\text{street}}$</td>
<td>22.73 deg</td>
<td></td>
</tr>
<tr>
<td>$D_{\text{max},S}$</td>
<td>47.47 deg</td>
<td>Size of swaths away from seam</td>
</tr>
<tr>
<td>$D_{\text{max},O}$</td>
<td>45.5 deg</td>
<td>Size of swaths near seam</td>
</tr>
</tbody>
</table>

Additionally, the corresponding constellation system is shown in Figure 12 below.

**Figure 12. Second Constellation System over Mars, $N_P = 18$, 3 Planes**
For this constellation, the area labeled as Place1 is seen by at most seven satellites for various durations over a 24-hour period. Place1 is not seen by any CubeSats for about 25 minutes, when it passes between plane 3 and plane 1. This period of time is far less than that of the previous constellation system due to larger swath size, which is in turn due to the lower elevation angle. Lower elevation angles helps with using a lower number of planes to cover a lot of area. Still, keeping all other parameters the same for this system, one additional plane is needed for complete coverage over the central body.

Third Constellation System
Parameters for the third three-plane constellation are shown in Table 13.

### Table 13. Third Constellation Parameters, \( N_p = 18, 3 \) Planes

<table>
<thead>
<tr>
<th>Parameters</th>
<th>Values</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>( S )</td>
<td>20 degrees</td>
<td>For satellites within each plane</td>
</tr>
<tr>
<td>( t_{\text{rev}} )</td>
<td>410.3 seconds</td>
<td>For discontinuous coverage</td>
</tr>
<tr>
<td>( \lambda_{\text{max}} )</td>
<td>28.36 degrees</td>
<td>Assume 1° elevation angle, Coverage in swath is continuous</td>
</tr>
<tr>
<td>( \lambda_{\text{street}} )</td>
<td>26.68 degrees</td>
<td></td>
</tr>
<tr>
<td>( D_{\text{max},S} )</td>
<td>55.04 degrees</td>
<td>Size of swaths away from seam</td>
</tr>
<tr>
<td>( D_{\text{max},O} )</td>
<td>53.36 degrees</td>
<td>Size of swaths near seam</td>
</tr>
</tbody>
</table>

The corresponding constellation system is shown in Figure 13 below.

**Figure 13. Third Constellation System over Mars, \( N_p = 18, 3 \) Planes**
For this constellation, the area labeled as Place 1 is seen by at least one CubeSat throughout the entire 24-hour duration. Therefore, this constellation system can be considered as fully continuous.

Discussion

Constellations with only one or two planes are better responsive to user needs. However, for this project, three to five planes were considered to achieve better coverage of the central body. Of the three constellation systems presented in this chapter, the third one requires the fewest number of planes to cover the most possible area. Therefore, it is the most ideal system, provided that there are no restrictions on the elevation angle.

For communications satellites, the elevation angle should be at least 5 degrees (Larson & Wertz, 1999). In this case, a version of the second constellation system would be the most ideal, which would have four planes instead of three to achieve maximum possible coverage. A version of the sample constellation system, with five planes instead of three, would also work; however, it would not be as ideal because it is less responsive to user needs.

Note that the results only correspond to an altitude of 500 km over Mars. If the central body was a larger planet such as Venus, for example, the constellation system would require more planes to cover the entire area. A smaller body such as the Moon would require no more than two planes regardless of the elevation angle.

If the altitude was decreased over Mars, more planes would be needed since the CubeSats would cover a smaller portion than before. The opposite is true if the altitude was increased, since the CubeSats would cover more area.

Conclusion

Constellation systems depend on user needs and specific mission requirements, as well as the type of CubeSats used. Given the calculations and simulations presented in this paper, the number of planes and CubeSats in the constellation system depend on the central body and orbit characteristics. Larger bodies generally need more planes for complete coverage, as do constellation systems with smaller orbits.

The CubeSats themselves also depend on the user needs and mission requirements, which determine the type of instruments carried. Deep space CubeSats need to communicate with ground stations on Earth, so their communication instruments should allow them to transmit and receive data over long distances. They have different power requirements than those surrounding the Earth, and the size of the CubeSat box should be able to support the total mass of all the instruments. The payload depends on the
mission. For this project, the payload includes an imager/camera, altimeter, and radiometer.

A deep space CubeSat constellation system, such as those simulated in this project, provides coverage over large areas of a central body. Therefore, it would help gather data for future deep space missions. Using Mars as the central body for the simulations presented, the potential constellation systems could help determine where future missions could focus on exploring. Perhaps, these constellation systems can help determine the best areas for humans to physically explore next.

References


