Mars Telecommunications Constellation

15 May, 2019 MAE 5630 Erik Gustafson (eog5)

1 Abstract

The Mars Telecommunications Constellation is a proposed mission to establish a continuous ground coverage relay network over Mars capable of transmitting high data rates ($\geq$ 32 mega bits per second). The total constellation will be complete and operational for a total of 5 years, with the potential to operate indefinitely with additional launches. The proposed architecture is a constellation of 32 12U cubeSats, each equipped with an X-Band communications system for short range communication with assets at Mars, an optical communications system for communications with assets at Earth, and a solar sail as the sole source of on-board propulsion.

To establish the feasibility of this mission, 3 main aspects of the mission design and architecture were evaluated and developed. 1: The capabilities of optical communications systems, and their ability to fit within the mass and volume constraints of a cubeSat. 2: The capabilities of a cubeSat equipped with a solar sail, and its ability to navigate from Earth orbit to Mars Orbit. 3: Constellation design, with station keeping performed by a solar sail. In addition, the major cubeSat subsystems, with particular attention toward reducing cost through the use of commercial off-the-shelf (COTS) components, were developed, taking inspiration from the Mars Cube One (MarCo) architecture.

2 Introduction

Mars has been one of the centers of attention for planetary science missions for many years. While there are currently 8 science assets in orbit around or on the surface of Mars [1] , each of these assets is designed to handling its own communications with Earth, through the Deep Space Network (DSN). Some surface assets increase their data rate by relaying through the orbiting science assets, but these satellites are not in orbits conducive to continuously relaying information. One proposed solution to this problem was the Mars Telecommunications Orbiter (MTO), which would use optical communication to increase the available bandwidth by acting as a relay for Mars assets. While MTO was cancelled in 2005 due to budget constraints and the need to fund a Hubble servicing mission, the need for increased bandwidth to and from Mars is still present, and will only increase as Mars becomes the target of more robotic and human missions.

One solution to this problem is to deploy a constellation of 12U cubeSats, each equipped to serve as a communications relay. Additionally, these cubeSats could be equipped with solar sails, allowing them to navigate from Earth orbit to Mars, vastly reducing the launch cost for the mission. The Mars Telecommunications Constellation is a proposed mission to provide Mars with global communications relay coverage capable of transmitting at least 32 megabits per second. Producing a constellation of smallSats is cost effective due to the economy of scale, which opens the opportunity for redundant units to be launched, reducing the impact of any one satellite failing.

An emphasis on the use of COTS hardware will reduce the development time and cost for this mission. This will allow the mission to use hardware that has only recently been proven at a high tech readiness level (TRL). Such hardware would be off limits for a larger, more expensive satellite, that has a longer development timeline. Additionally, since this mission will feature redundant satellites, a higher risk of failure can be tolerated for any one satellite. This may allow higher risk, lower TRL hardware to be flown than would be tolerated on a flagship mission.

3 Mission Success Criteria

The main goals of this mission are as follows:

1. To navigate from Earth orbit to Mars orbit with solar sails.
2. To demonstrate high data-rate ($> 32Mbps$) communication from Mars orbit with an optical communication system.
3. To serve as a high data-rate ($> 32Mbps$) relay for a single target on the surface of Mars.
4. To provide continuous relay coverage for the entire surface of Mars for 5 years.

The first three objectives can be completed by a single satellite. The last objective can only be achieved by a constellation of satellites with global coverage.

4 Definition of Terms

G : Gravitational Constant
m_x :mass of object x
$\mu = Gm$: Gravitational Parameter
G_{sc} : Solar Irradiance at 1 AU
c : Speed of Light
n_x: mean motion
a_x: semi-major axis
α : Solar Sail Pitch Angle

σ : Solar sail size to mass ratio
A : Solar Sail Area
$\hat{\mathbf{n}}$: Unit Vector Normal to Solar Sail
$\mathbf{r_{x/i}}$:Position vector of object x with respect to object i
$\mathbf{v_x}$:Velocity vector
$\mathbf{a_x}$:Acceleration vector
θ_{wc} : Worst Case Solar Panel Illumination Angle
I_d : Solar Cell Power Transfer Efficiency
P_0 : Solar Power Area Density at 1 AU
η_{cells} : Solar Cell Efficiency
L_d : Solar Cell Degradation
T_m : Mission Time in Years
d_y : Solar Cell Annual Degredation
σ_s : Solar Cell Area Density
A_s : Solar Cell Area
F : Focal Length
w_0 : Beam Waist Width
D_t : Transmitter Lens Diameter
D_r : Receiver Lens Diameter
η : Gain Efficiency
G_t : Transmitter Gain
G_r : Receiver Gain
$\frac{E_b}{N_0}$: Energy per Bit to Noise Power Spectral Density Ratio
L_a : Atmospheric Losses
D_L : Lens Diameter
z_r : Wavefront Curvature
z_x : Axial Position x
w_x : Beam width at position x
θ_D : Divergence Angle
θ_J : Jitter Angle
L_{pt} : Pointing Losses
P_t : Transmitter Power
R_b : Bitrate
k : Boltzmann Constant
T : Noise Temperature
R_{max} :Maximum Transmission Distance
P_{sa} : Power to Solar Arrays
T_d: Time Illuminated
T_e: Time Eclipsed
X_d: Power Path Efficiency while Illuminated
X_d: Power Path Efficiency while Eclipsed
P_{req}: Power Required

5 Background and Technology

The original concept for the Mars Telecommunications orbiter called for a large ($1,000kg$) spacecraft to orbit mars in a high altitude ($5,000km$) orbit [2]. This orbit, higher than other-science focused orbiters, would be above ground targets for longer periods of time, allowing more data to be relayed. The MTO was to be equipped with X-band and Ka-band antennas, that could move independently to communicate with Mars ground targets. In addition, it was to have an optical communications system capable of communicating at > 30 megabits per second. The MTO was to be the first demonstration of a deep space optical communications system, before the program was cancelled in 2005.

Though the MTO has been cancelled, the problem of communication with Mars still remains. The Mars Reconnaissance Orbiter and Mars Odyssey are relied on to act as communication relays to the science assets on the surface. Data rates for the rovers direct to Earth vary between 500 bits to 32 kilobits per second [3]. MRO and Mars Odyssey can transmit up to 2 megabits and 256 kilobits per second, respectively. However, due to the science payloads' orbits, ground assets can only communicate with them for 8 minutes per sol. While relaying data is technically possible with the existing assets, the frequency of commands sent to ground assets, and the amount of data that they send back could be greatly increased with a dedicated, high-rate communications infrastructure.

Since the cancellation of the MTO, three technologies have been developed will allow for a more capable telecommunications relay infrastructure to be established at a lower cost. The first technology has been cubeSats, which have recently been proven to be capable of successfully operating on long-duration deep space missions. The second is solar sails, many of which are currently under development, which will greatly expand the trajectory modification capability of satellites with little to no propellant mass. The third is optical communications, which have been proven to be capable of transmitting high data rates.

5.1 Mars Cube One

The Mars InSight mission was accompanied by two cubeSats, which were designed to relay telemetry and other data from the InSight mission during entry, descent, and landing (EDL) [5]. Each cubeSat was 6U, and capable of independently flying to mars, performing deep space communication, navigation, and trajectory correction. The MarCO program served as a demonstration of the capability of nanoSats, and their ability to carry out deep space relay missions. Their architecture will serve as the foundation for the Mars Telecommunications Constellation.

5.2 Solar Sail Demonstrations

Solar sails are an essential part of this mission architecture, as they are used as both a cost reduction method, and a method to augment mission life. Relying on solar sails to propel the spacecraft from Earth orbit to Mars allows cubeSats to be deployed as secondary or ride share payloads, significantly reducing the cost to launch. Additionally, relying on solar sails for station keeping and trajectory control at Mars removes propellant as a mission life constraint.

Solar sail deployment methods for nanoSats are currently being investigated and developed by several groups of researchers, including the LightSail project, directed by the Planetary Society [6]. While no nanoSat has successfully navigated with a deployable solar sail, JAXA has successfully flown IKAROS, a 310 kg satellite that has seen 400 m/s ΔV from its solar sails [7]. NASA currently lists nanoSat deployable solar sails at TRL 6 [9]. The TRL level of nanoSat deployable solar sails is critical to this mission architecture, as it provides the only means of propulsion for each satellite. The launch of LightSail II, which will demonstrate cubeSat solar sail deployment and navigation, is scheduled for 2019 [10].

5.3 Optical Communications Demonstrations

Optical communication systems (OCS) are capable of higher data rates, with less mass and power than a radio frequency (RF) communications system [12]. While higher data rates are important for increasing the commands that can be sent and science that can be received from assets on Mars, the other features of OCS make it especially desirable for smallSats or nanoSats, which have even more stringent mass and power requirements than their larger counterparts. NASA has recently demonstrated the capability of nanosats with OCS, where a 1.5U cubeSat achieved a 100 megabits per second data rate [13]. A scaled version of this same technology should be capable of very high data rates from Mars orbit.

While deep space optical communications systems have not been tested, the Laser Communications Relay Demonstration (LCRD) is set to launch in 2019. This mission is predicted to be able to transmit over 1 giga-bits per second [14], and has been identified as a candidate technology for future future Mars science missions [17].

Optical communications is a key technology for this mission architecture. While optical communications systems have been flown, a cubeSat sized optical communications system capable of high data rates in deep space must be demonstrated during the pre-phase A studies before this architecture can be pursued any further. This may require the development of a cubeSat to fly a modified version of the LCRD system in Earth Orbit.

6 Technical Approach

The proposed architecture is a constellation of cubeSats, each equipped with an optical communications system. The constellation will act as a relay for both ground and orbiting assets at Mars. To reduce launch cost and increase station keeping and trajectory modification capabilities, each spacecraft would be equipped with a solar sail, and navigate to Mars from Earth orbit.

6.1 Solar Sail

Much of the sizing of the mission depends on the volume and mass of the solar sail. Each cubeSat was modeled to be 12U, and to be 24 kg. A simulation of a solar sail spacecraft was created to determine how solar sail size affects the amount of time required to escape from Earth's orbit, and the amount of time required for the transfer to Mars.

A circular restricted, 4 body simulation was created, where the cubeSat was subject to the gravity of the Sun, the Earth, and Mars. The sun was modeled as an inertially fixed point, and the Earth and Mars were fixed on circular orbits, with their positions calculated with Equation 1.

$$\mathbf{r_x} = a_x cos(n_x t)\hat{\mathbf{i}} + a_x sin(n_x t)\hat{\mathbf{j}} \tag{1}$$

For this analysis, Mars and Earth were assumed to have co-planar, zero eccentricity orbits. CubeSats were initially placed on a Geosynchronous orbit about Earth, and were subject to the dynamics described by Equation 2.

$$\mathbf{a_{sat}} = \sum -\frac{G m_i r_{sat/i}}{\|r_{sat/i}\|^3} + F_{sail} \tag{2}$$

Where the summation is over the Earth, Mars, and the Sun. The solar sail was modeled to be an ideal reflecting solar sail, whose specific force is described by Equation 3.

$$F_{sail} = 2\frac{G_{sc} cos(\alpha)^2 \|a_{Earth}\|^2}{c\|r_{sat/sun}\|^2}\frac{A_{sail}}{m_{sat}}\hat{\mathbf{n}} \tag{3}$$

The solar sail was pointed to achieve optimal energy gain, based on the work performed by MacDonald et al [11]. This formulation was adapted to the planar restricted motion developed for this simulation.

For this problem, three reference frames were used. The first is an inertial reference frame $(\hat{i}, \hat{j}, \hat{k}, 0)$, with its origin at the Sun. The inertial frame's X-Y plane is orbital plane, which all motion is constrained to. Its Z axis is consistent with the right hand rule.

The second frame rotates with the Earth's. Its x axis ($\hat{\mathbf{x}}_{\mathbf{Earth}}$) is parallel to $\mathbf{r_{Earth/Sun}}$. Its z axis ($\hat{\mathbf{z}}_{\mathbf{Earth}}$) is parallel to the inertial Z axis. Its y axis ($\hat{\mathbf{y}}_{\mathbf{Earth}}$) is consistent with the right hand rule.

The third fame is attached to the cubeSat. Its x axis ($\hat{\mathbf{x}}_{\mathbf{sat}}$) is parallel to $\mathbf{r_{Sat/Sun}}$. Its z axis ($\hat{\mathbf{z}}_{\mathbf{sat}}$) is parallel to the inertial Z axis. Its y axis ($\hat{\mathbf{y}}_{\mathbf{sat}}$) is consistent with the right hand rule. α is defined as the angle between ($\hat{\mathbf{x}}_{\mathbf{sat}}$) and ($\hat{\mathbf{n}}$). θ is definied as the angle between $\hat{i}$ and ($\hat{\mathbf{x}}_{\mathbf{sat}}$).

To find the optimal $\hat{\mathbf{n}}$, first the optimal pitch angle (α) must be found. Solving the optimal control control problem yields Equation 4.

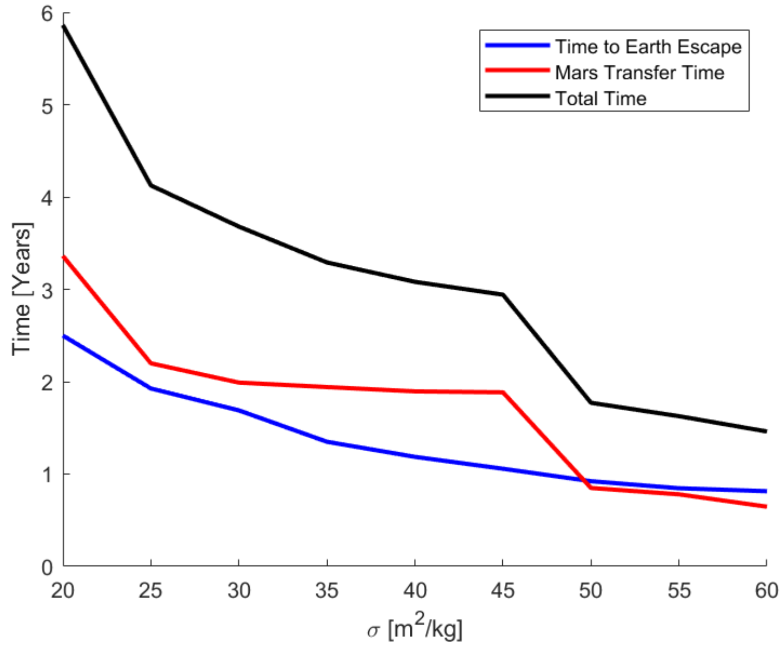

Figure 1: This figure shows how changing the sail size to mass ratio (σ) influences time required for three key mission events: Time to escape from Earth, Transit Time from Earth to Mars, and Total Transfer Time.

$$\alpha = arctan\left(\frac{3v_x}{4v_y} + \sqrt{\frac{1}{2} + \left(\frac{3v_x}{4v_y}\right)^2}\right) \tag{4}$$

Where $v_x = \mathbf{v_{sat}} \cdot \mathbf{\hat{x}_{sat}}$ and $v_y = \mathbf{v_{sat}} \cdot \mathbf{\hat{y}_{sat}}$. The reference frame definitions, and this angle, can easily be used to find $\mathbf{\hat{n}}$.

Equation 2 was used to to numerically integrate the planar trajectory of a cubeSat, from Geostationary orbit, until it crossed Mars' orbital radius. This was simulated with a fixed time-step leapfrog integrator. Simulations were run for 12U cubeSats with solar sails ranging from 300 to 2300 m^2. In each simulation, the time to escape Earth and Mars transfer time was was recorded (1).

Time to Earth escape was defined as the amount of time from the beginning of the simulation, where the cubeSat was in a geosynchronous parking orbit, to when the cubeSat left the Earth's sphere of influence. Mars transfer time was defined as the amount of time between the cubeSat leaving the Earth's sphere of influence, and it crossing Mars' orbit ($\|r_{sat/sun}\| > a_{mars}$).

While the time to Earth escape varies smoothly as sail size is increased, the Mars transfer time had unexpected variations. This is largely due variation in the direction each cubeSat left Earth's orbit. Additional control needs to be implemented to optimize that part of the trajectory.

These simulations demonstrated that it is possible for a cubeSat with a small solar sail to reach Mars Orbit. Next, the influence of solar sail size on the volume and mass budget was analyzed. The solar sail is assumed to made of Mylar or Kapton, with an area density of $6g/m^2$ [15]. The boom, which span the diagonals of a square sail. are assumed to have similar properties to the booms developed for the "Advanced Composites-Based Solar Sail System [16], which have a linear density of 46 g/m. The sail and deployer are assumed to have a volume proportional to the NEA Scout solar sail drums, which has a volume to sail area ratio of 7.5 cm^3/m^2. Once the sail mass and boom mass were calculated, it was assumed that the entire system would be no more than 1.3 times the mass of the boom and the sail. These values were used to estimate the mass and volume of a solar sail based on its area (Figure 2).

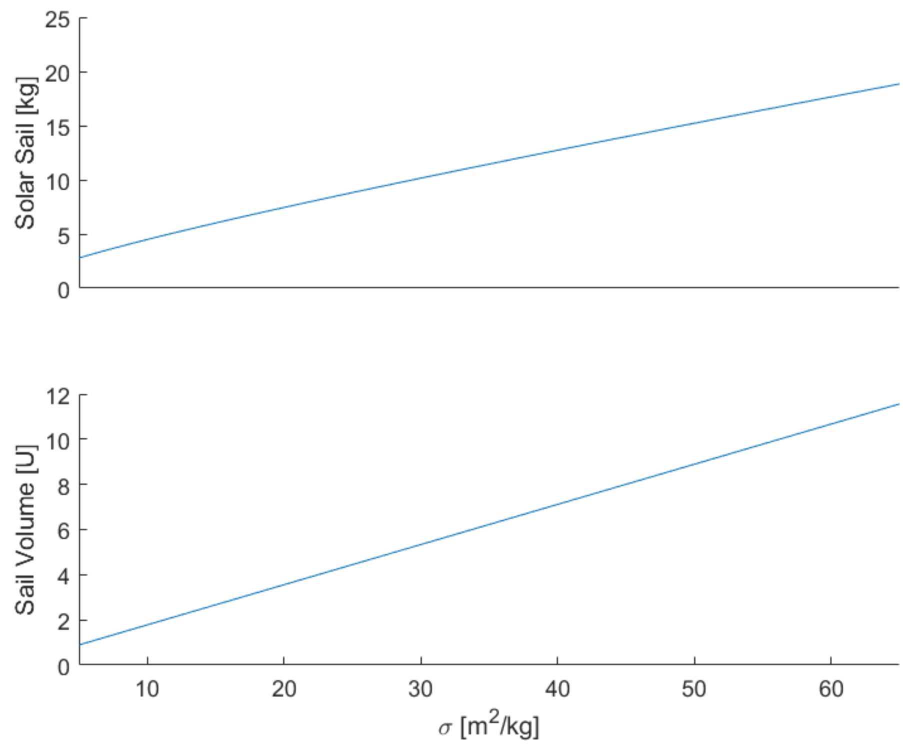

Figure 2: This figure shows the relationship between σ and the mass and volume taken up by the solar sail. This plot is for a 12U cubeSat with a total mass of 24kg.

This data, in combination with the relationship between σ and total time was used to pick $\sigma = 40m^2/kg$, as it leaves considerable mass and volume for the other critical subsystems, while keeping the total time to Mars under 4 years.

6.2 Configuration and Sizing

The architecture for this mission will be heavily based on the flight-proven architecture used on MarCO. Additional sizing and hardware selection is drawn from the architecture proposed by Robert Staehle.

The cubeSat requirements state that a 12U cubeSat is to have a mass no greater than $24kg$, and have dimensions of $229mm$ x $239mm$ x $366mm$, allowing for a total volume of $0.02m^3$ [19]. Mass and volume will be the primary constraints on the subsystem sizing performed for this mission.

6.2.1 Interplanetary cubeSats

Robert Staehle et al. proposes a 6U cubeSat (StaehleSat), equipped with a solar sail and optical communications system. This architecture is designed for interplanetary science missions, and is capable of carrying 2U of payload. The key problems identified by Staehle are survivability of solar panels in the deep space radiation environment, and data rates for the small optical communications system. Using a similar architecture to StaehleSat for a communications relay satellite, but increasing the satellite to a 12U cubeSat, would allow additional mass and volume to be dedicated to critical subsystems, such as the power system, the solar sail, or the optical communications system.

6.2.2 Power

A simple power system architecture would consist of solar panels and batteries, as these components are already commonly used on cubeSats. Additionally, a common solar panel deployment scheme is single-deployable solar panels, where the four largest faces of the cubeSat are covered with panels. For a 12U cubeSat, that allows for $A = 0.3426m^2$ of solar panels.

Current solar cells typically used on cubeSats have an efficiency of $\eta_{cells} = .29$, and an area density of $\sigma_s = 3.6kg/m^2$ [20]. The solar cell power transfer efficiency is assumed to be $I_d = 0.77$. Assuming an annual efficiency degradation of $d_y = .03$, the total solar cell degradation factor can be calculated with Equation 5.

$$L_d = \eta_{cells} * (1 - d_y)^{T_m} = 0.7374 \tag{5}$$

With this value set, the end of life power, at Mars orbit, can be calculated with Equation 6.

$$P_{Mars,eol} = \eta_s P_0 I_d L_d cos(\theta_{wc}) A_s \frac{a_{Earth}^2}{a_{Mars}^2} = 22.58W \tag{6}$$

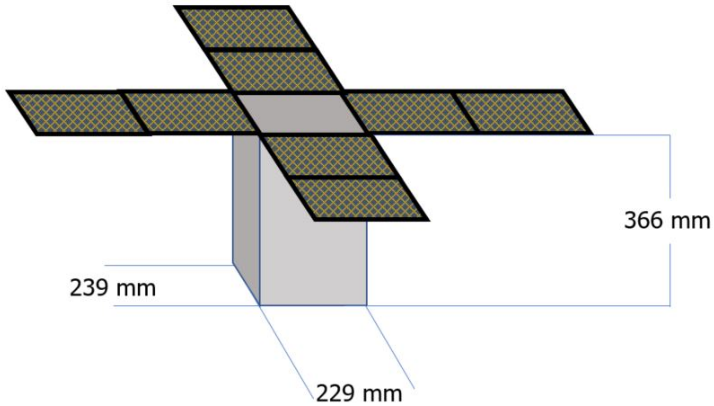

Figure 3: This figure shows single delpoyable panels, after deployment. They come from the four largest faces of the 12U bus.

This value for $P_{Mars,eol}$ represents the power limit for this architecture. While it would be possible to get more power by using double deployable panels, the single deployable architecture should be used unless its power output is found to be deficient.

The power consumption for the Optical Communications System, and the RF communications system, are discussed in Sections 6.2.3 and 6.2.4. Assuming the other subsystems (Attitude Determination and Control, Command & Data Handling, and Guidance Navigation & Control) have a power draw of 5 W, the total required power is $P_{req} = 19.87W$.

The power budget was closed by iterating the altitude until the time eclipsed, and the RF power consumption, created a closed power budget. This altitude was 500 km, which gives a maximum time eclipsed of 18.62 minutes, with an orbital period of 195.96 minutes.

Assuming a path efficiency $X_d = .85$ while the panels are under illumination and $X_e = 0.65$ while the panels are eclipsed, the required power to the panels during the illuminated portions of the orbit can be determined with Equation 7, and was found to be $P_{sa} = 21.4W$, which is within the end of life power limit.

$$P_{sa} = \frac{\frac{X_d P_{req}}{T_d} + \frac{X_e P_{req}}{T_e}}{T_d} \tag{7}$$

6.2.3 Optical Communications

The optical communications system must be sized such that its major dimensions are not larger than the dimensions of a 12U cubeSat. The transmitter power was arbitrarily set to be $P_t = .5P_{Mars,eol} = 11.29W$.

Another goal for the communications relay is to be able to achieve a Bit Error Rate (BER) less than 1e-8 for BPSK, QPSK, 8PSK, or 16QAM encoding. This is achieved if $\frac{E_b}{N_0} > 16dB$ [22].

A laser with a wavelength of $\lambda = 1060nm$ was selected due to the fact that sensors for this wavelength are relatively common [21].

Additional assumptions were made, including setting the atmospheric losses to $L_a = -.5dB$. The jitter angle was assumed to be the satellite pointing accuracy, which was set to $\theta_j = .1deg$. The laser was assumed to be focused to a waist with a lens where $D_l = 0.01m$ and $F = .005m$. The receiver noise temperature was assumed to be $T = 200K$. The gain efficiency was assumed to be $\eta = 0.55$. The diameter of the receiver lens was assumed to be $D_r = 5m$, and the diameter of the transmitting lens was initially set to be $D_t = 0.18m$.

The waist diameter was determined modeling the lens as ideal, with 8.

$$2w_0 = \frac{4\lambda}{\pi} \frac{F}{D_l} \tag{8}$$

This gives $w_0 = 3.37e - 07m$. This was used to determine the distance between the laser and the transmitting lens (Equation 10), the divergence angle (Equation 11), and the pointing losses (Equation 12).

$$z_r = \pi \frac{w_0^2}{\lambda} = \tag{9}$$

$$z = z_r \sqrt{\frac{w_z}{w_0}^2 - 1} \tag{10}$$

The distance between the laser and the transmitting lens was taken to be $2z = 0.16m$ where z was calculated with $w_z = D_t/2$. This distance is sufficiently small to fit within the satellite bus.

$$\theta_D = \frac{\lambda}{\pi w_0} \tag{11}$$

$$L_{pt} = e^{-8\frac{\theta_j^2}{\theta_D^2}} \tag{12}$$

$\theta_D = 57.3deg$ was calculated, and used to find the pointing loss, $L_{pt} = 0dB$. This low pointing loss is due $\theta_D >> \theta_j$. Next, G_t and G_r were calculated with Equation 13. All of the preceding values were used in Equation 14 [23] to calculate the maximum bit-rate possible when the Earth and Mars are the furthest from one another, $R_{max} = a_{Earth} + a_{Mars}$. This yielded a bit rate of $R_b = 8.64e + 07bps = 86.4Mbps$, which fulfills Mission Success Criteria 2.

The minimum power required to fulfill this mission success criteria is 8.36 W. This value was used for sizing the power system.

$$G_x = \eta \left(\frac{\pi D_x}{\lambda} \right)^2 \tag{13}$$

$$\frac{E_b}{n_0} = \frac{P_t L_a L_p t G t G r \lambda^2}{R_b k T (4\pi R_{max})^2} \tag{14}$$

6.2.4 RF Communications

Each cubeSat will be equipped with an X-Band transmitter for short range communication with ground assets. The sizing for this communications system will be based on the communications capabilities of the Mars Science Laboratory (MSL) [24].

The MSL has a high gain antenna, which transmits at 8400 MHz ($\lambda = 35.7mm$), has a gain of 25 dB, and has a 15W amplifier.

The cubeSats will be equipped with a deployable X-band antenna, with a diameter of $D_t = 100cm$.

Some additional assumptions were made about the RF system. The atmospheric losses were assumed to be $L_a = -.5dB$. Pointing losses were assumed to be $L_{pt} = 0dB$. The altitude for the constellation was adjusted iterative until the power budget closed. With an altitude of 500 km, the maximum communication distance was calculated with Equation 15 to be $R_{max} = 1910km$.

$$R_{max} = \sqrt{(R_{Mars} + Altitude)^2 - R_{Mars}} \tag{15}$$

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With these assumptions, and BER and bit rate requirements identical to the optical system, the required power for the RF communications system was found with equation 14 to be $P_t = 2.65W$. This represents a maximum power consumption, at the longest range, with a minimum consumption of $P_t = .182W$ at a range of $R = 500km$.

6.2.5 Attitude Determination and Control

The attitude control system will consist of reaction wheels and a small cold gas thruster system for momentum dumping. It will be sized to meet the pointing requirements imposed by the communications system. To achieve the pointing requirements imposed by the optical communications system, these cubeSats will require star trackers, which are unusual for satellites of this size.

7 Mission Profile

7.1 Timeline

The overall mission will consist of the deployment of 33 identical 12U cubeSats. The first cubeSat (DemoSat) will be launched in advance of the other 32 units, and will complete the first three mission success criteria before the launch of the remaining units. The remaining 32 units will be launched within 2 years of one another, and will operate concurrently for 5 years. Assuming $\sigma = 40m^2/kg$, that corresponds to a 4 year total time from launch to Mars insertion, and a 11.5 year total lifespan for each cubeSat.

Sat1-Sat8 and Sat9-sat16 will consist of satellites in the $I = 45°, 135°$ orbits. These two orbits can provide coverage four times a day for the entire Martian surface, except for the polar regions (Figure 12). Sat17-Sat24 will consist of satellites in the $I = 0°$ orbit. Which will result in constant coverage for the entire surface, except for the poles. Sat25-32 will deploy in the $I = 90°$ orbit, and will provide constant coverage for the polar regions. Additional information on the constellation design can be found in Section 7.5.

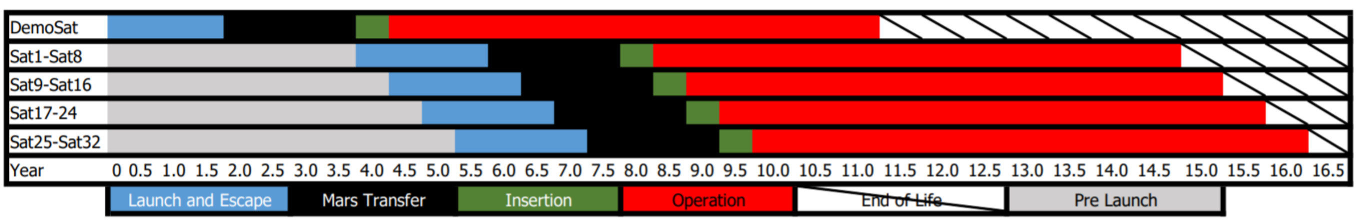

Figure 4: This figure shows the mission timeline, outlining the major stages of the cubeSat's lifespan: Launch and Earth Escape, Transfer to Mars and Orbit Insertion, Operation, and End of Life.

7.2 Launch

CubeSats will be launched as secondary or ride-share payloads to Geosynchronous orbit. Once in an orbit at geo, they will deploy their solar sails, and begin maneuvering to escape Earth. The first cubeSat will be the only telecommunications cubeSat on its launch, while the remaining 32 launches can be deployed on the same launch if there is adequate spare payload. While this mission requires launches to GEO, the low mass of each unit makes it likely that there will be many available launch opportunities for one or more cubeSat. It is assumed that all 32 launches can be accomodated in a 2 year span.

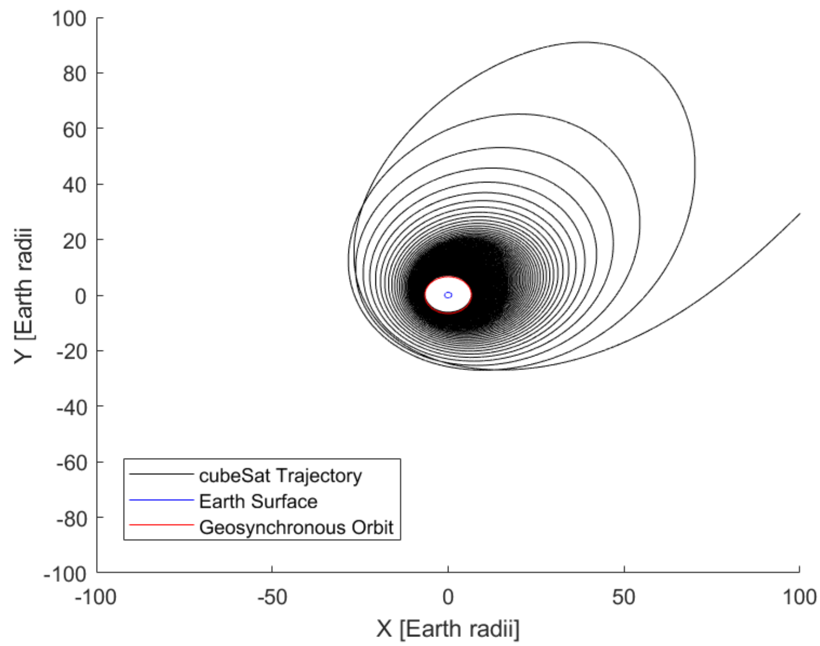

Figure 5: Earth escape trajectory of a cubeSat with $\sigma = 40m^2/kg$.

7.3 Earth Escape

Once at a geosynchronous orbit, cubeSats will use the optimal energy gain approach to achieve Earth escape, similar to the approach outlined in 6.1. This results in a trajectory that spirals out from Earth until escape velocity is achieved (Figure 5, 6).

7.4 Mars Transfer and Capture

Transfer to Mars will be achieved using the optimal energy gain approach (Figure 7, 8).

Once a cubeSat has reached a sufficiently large semi-major axis, it will need to switch from the optimal energy-gain control scheme to a Mars insertion control scheme. Optimal energy gain does not control eccentricity, which tends to prevent the cubeSat's specific energy from matching the energy required for insertion. Figure 9 shows the cubeSat's Mars relative velocity when they cross Mar's orbital radius in the simulations performed in Section 6.1. This mismatch is partially due to errors in eccentricity and semi-major axis.

For the purposes of this discussion, it will be assumed that launch dates and orbit phasing are appropriately chosen.

For a spacecraft with a solar sail, or any low thrust spacecraft, it becomes necessary to smoothly achieve the desired orbit. Therefore, an orbit-matching control scheme must be implemented before the cubeSat reaches Mars' orbit; the following

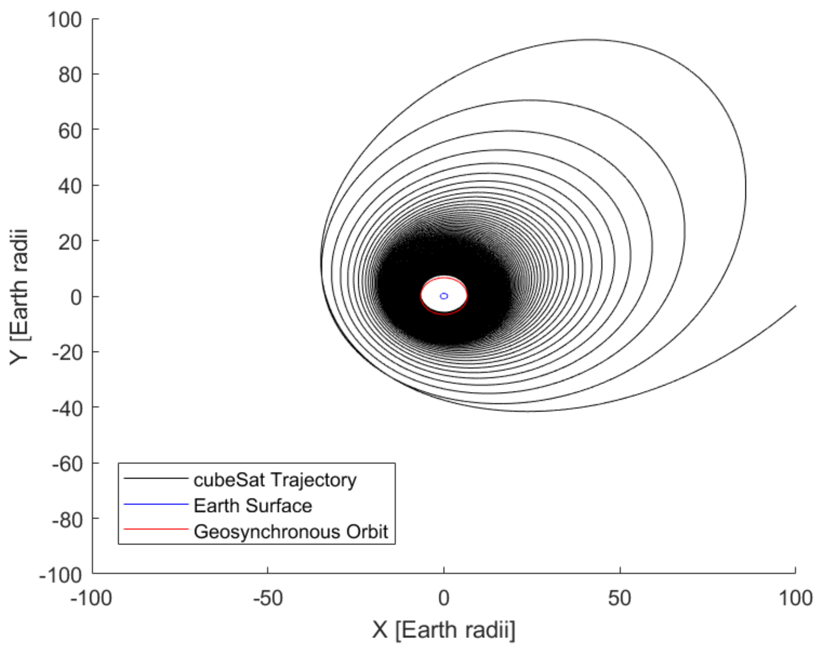

Figure 6: Earth escape trajectory of a cubeSat with $\sigma = 25m^2/kg$.

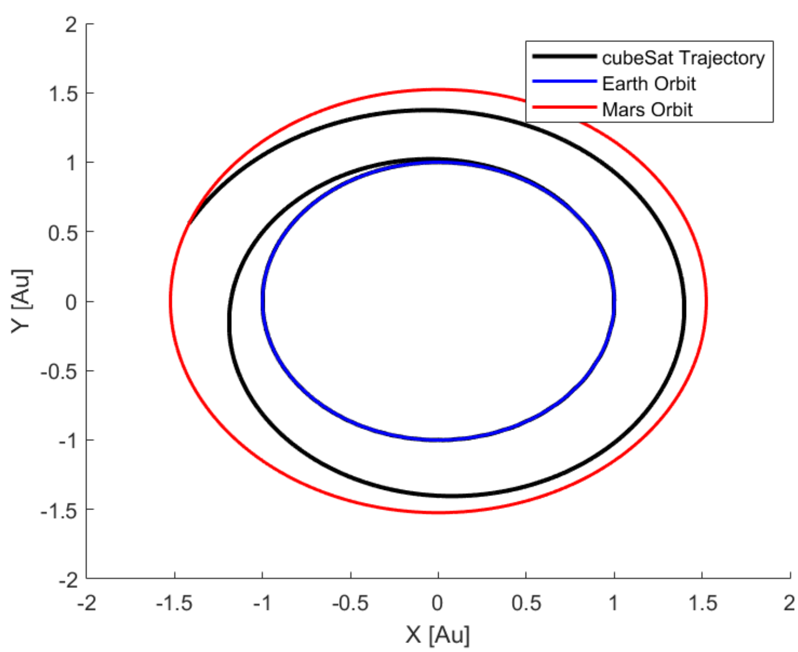

Figure 7: Mars transfer trajectory of a cubeSat with $\sigma = 40m^2/kg$.

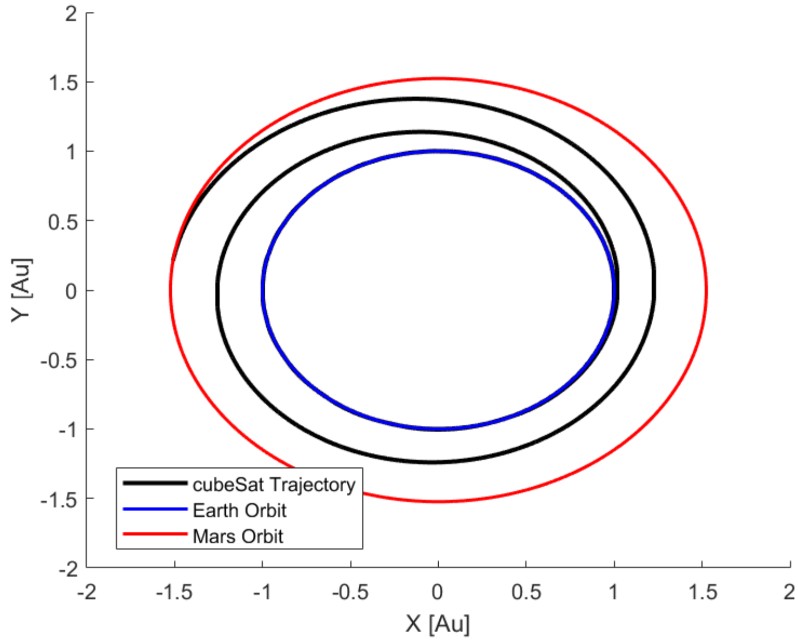

Figure 8: Mars transfer trajectory of a cubeSat with $\sigma = 25m^2/kg$.

control scheme was implemented once the cubeSat had achieved a radius at aphelion equal to the semi-major axis of Mars.

At this point, the cubeSats will begin to alternate between using the optimal energy gain control method, and pointing the sail in a passive direction, where it produces no thrust.

This scheme is tuned to avoid using the solar sail near aphelion and perihelion. When the sail is used to gain energy in these locations, it increases the eccentricity as well as the semi-major axis. This may result in an eccentricity that is greater than Mars' when the cubeSat's orbit matches Mars' semi-major axis. Since eccentricity is rapidly changed when the sail is used near aphelion and perihelion, it is better to use the solar sail when the cubeSat is far from these points.

A simple way of ensuring this is implementing control switching, as outlined in Equation 16. Here, the cubeSat only uses its solar sail if its true anomaly (ν) is at-least $\delta\nu = 63°$ away from either perihelion ($\nu = 0°$) or aphelion ($\nu = 180°$)

$$Control = \begin{cases} Passive, & \text{if } 63° > \nu \geq 0° \\ Energy\ Gain, & \text{if } 117° \geq \nu \geq 63° \\ Passive, & \text{if } 243° > \nu > 117° \\ Energy\ Gain, & \text{if } 297° \geq \nu \geq 243° \\ Passive, & \text{if } 360° \geq \nu > 297° \end{cases} \tag{16}$$

$\delta\nu$ can be adjusted depending on the cubeSat's initial trajectory in order to manipulate its encounter with Mars.

Figure 10 shows a trajectory generated using optimal energy gain until radius at aphelion was equal to Mars' semi-major axis, and then using control switching. This method produced a final eccentricity of 0.1044, and a Mars Relative velocity magnitude of $|V_{sat} - V_{Mars}| = 267m/s$, which is less than Mar's escape velocity at the hill radius, $280m/s$. This insertion maneuver takes 6 months, which has a negligible effect on the overall mission timeline.

7.5 Constellation Design and Operation

Constant global coverage is one of the key requirements for this mission, and it drives the choice of orbit for this mission. Orbits also drives communications and power system sizing, which are affected by the orbit's altitude and maximum time eclipsed.

While there are many types of constellations, the parameter that drives the number of satellites in the constellation is the altitude, as this dictates the coverage area for each satellite.

The constellation was iterative designed, to ensure the power budget closed. The final constellation calls for each cubeSat to be in a circular orbit, with an altitude of 500 km. 32 total cubeSats will be placed on 4 orbits. Each orbital plane will have 8 cubeSats equally spaced. All orbits will have a shared longitude of the ascending node, and will have inclinations of 0, 45, 90, and 135 degrees. Figure 11 shows the satellites positions in the constellation.

7.6 End of Life

The end of life plan is to use the solar sails to spiral into Mars, similar to the method used for Earth escape. The cubeSat will be destroyed on on entry into the Martian atmosphere.

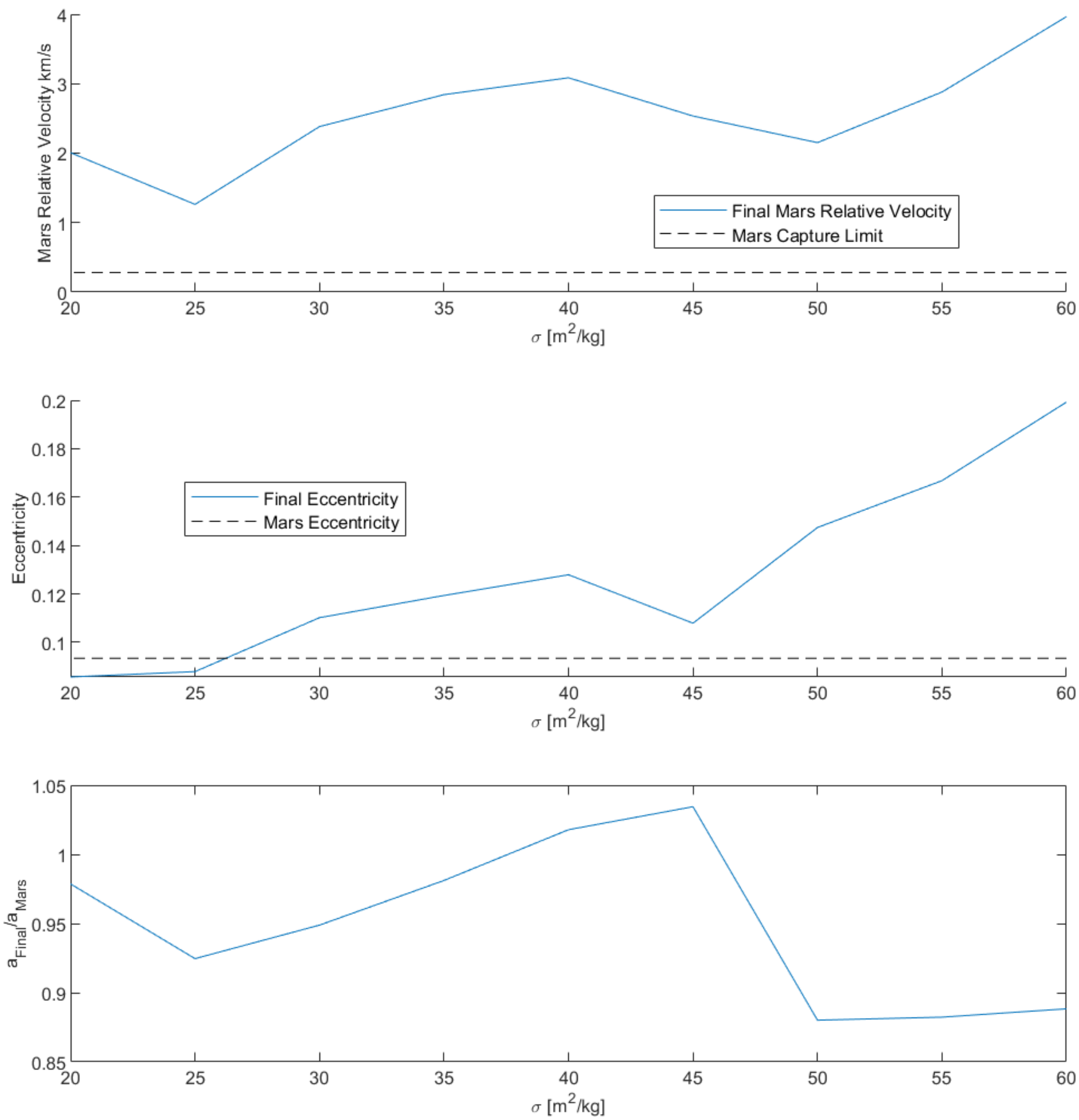

Figure 9: Plot of Mars Relative Velocity, eccentricity, and semi-major axis when the cubeSats cross Mars' orbit with optimal energy gain control.

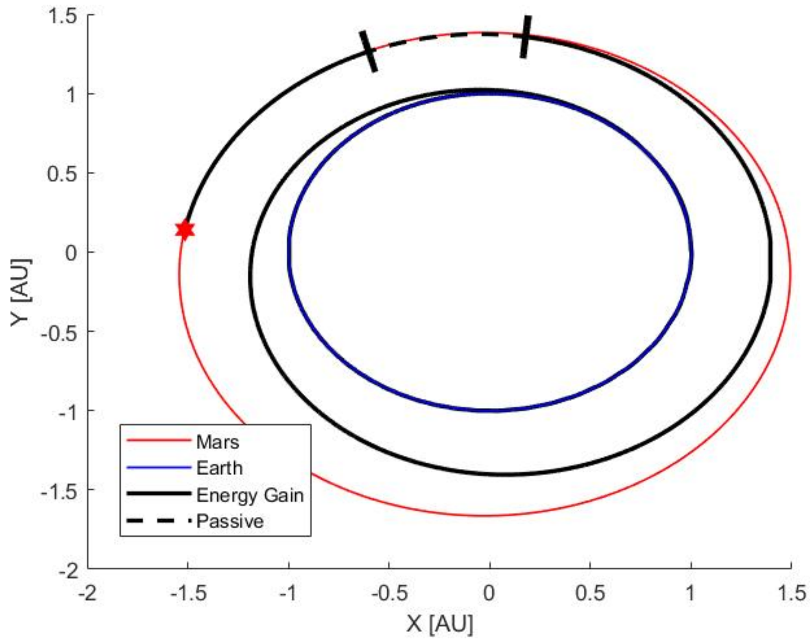

Figure 10: This figure shows a simulated trajectory for a $\sigma = 40m^2/kg$ cubeSat, taking it from Earth orbit to Mars capture. The dashed trajectory indicates where the cubeSat is getting no thrust from its solar sails, and the red star indicates where the Mars encounter occurs.

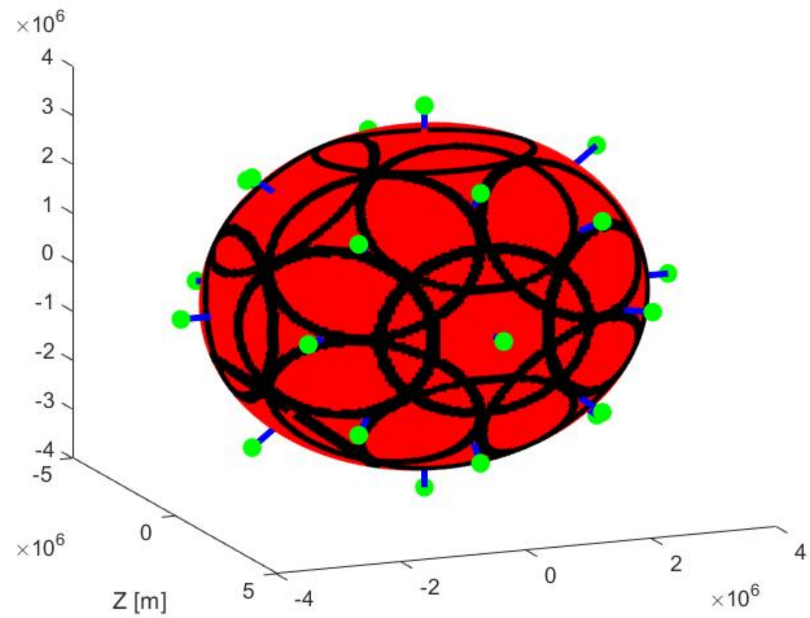

Figure 11: This figure shows the position of cubeSats in the constellation. The green circles are the satellites, the blue lines is their nadir pointing vector, and the black circles represent their coverage area. This constellation design ensures constant coverage of the entire surface.

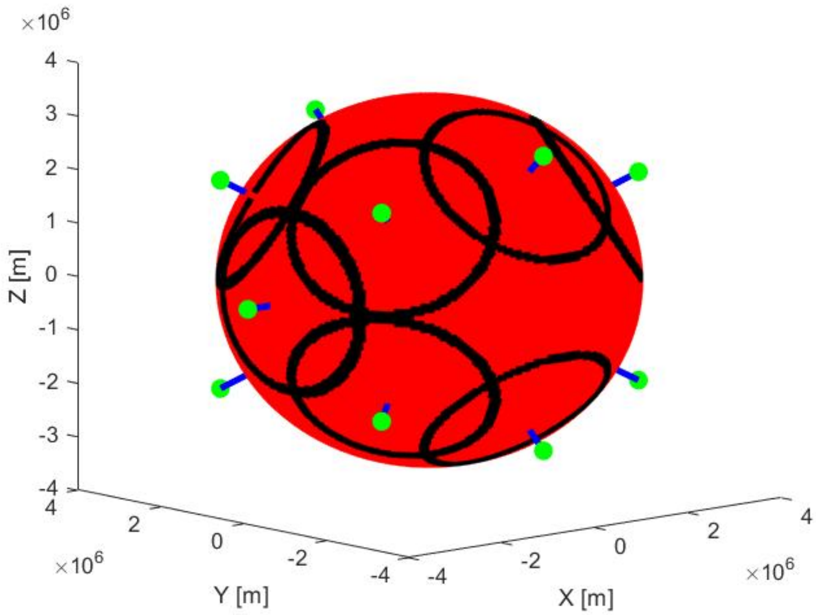

Figure 12: This figure shows the coverage provided by the first 16 cubeSats. Over the course of a Martian day, they provide coverage for the entire surface, except for very high and very low latitudes.

8 Conclusion

The Mars Telecommunications Constellation will take advantage of the emerging technologies of optical communications and solar sails, and package them in a cubeSat architecture. These new technologies, combined with commercial off-the-shelf components, allow this mission to achieve their objective of providing continuous high-data rate communications relays to the entire surface of Mars, while costing less than previously proposed MTO. This new capability will allow future missions to Mars greater freedom to use data-heavy instrumentation, or to implement missions that require constant communication.

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