INTERNATIONAL UNIVERSITY LIAISON INDONESIA - IULI

Constellation Design and Analysis of LEO Microsatellites for Air Traffic Monitoring and Surveillance in Indonesia

Presented to the Faculty of
Engineering

In Partial Fulfilment
Of the Requirements for the Degree
Bachelor of Sciences
In
Aviation Engineering

By
Giovanni HENGGAR Setyantoro

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"I can calculate the motion of heavenly bodies, but not the madness of people."

Sir Isaac Newton

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Abstract

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The main obstacles of the current implementation of the ground-based Automatic Dependant Surveillance (ADS) system in Indonesia arise from its geographical conditions. As an archipelago country stretching along equatorial arc that consists of thousand volcanic islands, it will require an immense capital investment in order to provide full coverage out of terrestrial ground stations. Utilizing the satellite-based ADS system will be one of the best solutions in order that should be explored to overcome the coverage limits, potential poor coverage, provided by the ground-based system. This research tried to design and analyze the implementation of LEO microsatellites constellation to be used for air traffic monitoring and surveillance system in Indonesia. The constellation design process was started from evaluating possible orbital types to be used, including the possibility of utilizing the specialized orbits. It was then continued to explore the full constellation configurations and finding the number of satellites. In this research, the physical modelling of two-body problem was used, while the perturbations considered in the age constellation analysis was the combination between the zonal gravitational J_2 and the atmospheric drag perturbations. This research found that the single plane equatorial constellation with 10 microsatellites were able to provide at minimum 90% of the coverage percentage continuously in the Indonesian region for more or less 2 years of service.

Keyword: *LEO, Constellation, Microsatellite, ADS*

Statement by The Author

I hereby declare that this submission is my own work and to the best of my knowledge, it contains no material previously published or written by another person, nor material which to a substantial extent has been accepted for the award of any other degree or diploma at any educational institution, except where due acknowledgement is made in the thesis.

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List of Abbreviations

ADS	**A**utomatic **D**ependent **S**urveillance
ADS-B	**A**utomatic **D**ependent **S**urveillance **B**roadcast
ADS-C	**A**utomatic **D**ependent **S**urveillance **C**ontract
GNSS	**G**lobal **S**atellite **N**avigation **S**ystem
GPS	**G**lobal **P**ositioning **S**ystem
GLONASS	**G**lobal’naya **N**avigatsionnaya **S**putnikovaya **S**istema
UAT	**U**niversal **A**ccess **T**ransceiver
ES	**E**xtended **S**quitter
ATC	**A**ir **T**raffic **C**ontroller
SAR	**S**earch **A**nd **R**escue
PSR	**P**rimary **S**urveillance **R**adar
SSR	**S**econdary **S**urveillance **R**adar
RAAN	**R**ight **A**scension of the **A**scending **N**ode
ECI	**E**arth **C**entered **I**nertial
LEO	**L**ow **E**arth **O**rbit
MEO	**M**edium **E**arth **O**rbit
HEO	**H**igh **E**arth **O**rbit
GEO	**GE**osynchronous **O**rbit
ICO	**I**ntermediate **C**ircular **O**rbit
ERS	**E**arth **R**eferenced **S**atellite
DMCii	**D**isaster **M**onitoring **C**onstellation for **I**nternational **I**maging
ODE	**O**rder **D**ifferential **E**quation

Physical Constants

Gravitational Constant	$\mu = 3.986 \times 10^6\,\mathrm{km^3\,s^{-2}}$
Radius of Earth	$Re = 6371\,\mathrm{km}$
Second Zonal Harmonic	$J_2 = 1.082\,63 \times 10^{-3}$
Gravitational Acceleration	$g = 9.806\,65\,\mathrm{m\,s^{-2}}$

List of Symbols

F_s	force	N
M	mass of the Earth	kg
M_s	mass of the satellite	kg
r	distance from the center of the Earth to the satellite	km
a	semi-major axis	km
e	eccentricity	
i	inclination	
r_p	radius of perigee	km
r_a	radius of apogee	km
v	speed of satellite	m s^{-2}
h	angular momentum	kgm^2 s^{-1}
T	period	s
p	semi-latus rectum	km
C_D	drag coefficient	
A	cross-sectional area of the satellite	m^2
t	number of satellite	
p	number of equally spaced planes	
f	relative spacing between satellites in adjacent planes	
d	coverage radius of the satellite	m
S	surface coverage area of the satellite	m^2
h	altitude of the satellite	m
ν, θ	true anomaly	$^\circ$
Ω	right ascension of the ascending node	$^\circ$
ω	argument of perigee	$^\circ$
ρ	mass density of the atmosphere	kg m^{-3}
ε	elevation angle	$^\circ$
α	nadir angle	$^\circ$
β	central angle	$^\circ$

For my parents

Chapter 1

Introduction

1.1 Background

1.1.1 ADS-B overview and protocol

An Automatic Dependent Surveillance (ADS) system is an effort to utilize non-primary traditional radar-based surveillance systems in order to detect and locating relative state condition of an aircraft using the pre-installed communicative on-board unit. There are two types of ADS system currently being implemented: the broadcast type (ADS-B) and the contact type (ADS-C). The word based on the ADS-B scheme comes from its dependence on the correct functioning of the on-board machinery which determines the position and velocity vector and the accessibility of the installed sending scheme [1].

ADS-B Implementation's primary purpose is to enable all users to access extremely precise data on the location and flight route of an aircraft. The ADS-B will essentially distribute the information to anyone listening. Other aircraft with ADS-B installed can freely receive the broadcasted signal and process it to improve its own situational awareness, and determine the separation radius to other aircraft, which in turn, allows for safer operations. The Automatic Dependant Surveillance-Broadcast relies on the capability of the aircraft or other possible vehicles to broadcast and provide their surveillance data enabling the vehicle to be tracked. This surveillance information mainly consists of satellite based (e.g. GPS, GLONAS, GNSS): vector position and velocity of the aircraft. In addition to position and velocity information, ADS-B signal could also contain additional information, such as, meteorological data that could also be useful for navigational purpose [2, 3].

The ADS-B surveillance services are composed of two large devices, the 1090 Extended Squitter (1090ES) operating on the 1090 MHz base frequency and the Universal Access Transceiver (UAT) operating on the 978 MHz base frequency. Both signals are emitted from the Mode-S transponder of the aircraft. The 1090ES system is commercially and globally adopted, however the UAT is only regional specifically implemented in the U.S. and mandatory only with flying latitude less than 18,000 ft. The data set of aircraft information will periodically be transmitted automatically, at least once every second to be more specific, without the help and any intervention of the flight crew or operator input. Other parameters that may be transmitted by the ADS-B system are preselected and static during the flight [4].

There are two segmentations of the implementation of the ADS-B, the ground and air segment. The air segment consists of two types of ADS-B that can be installed on-board the aircraft, ADS-B In and ADS-B Out. According to FAA, ADS-B Out type is the one that is mandated to be installed by January 1, 2020. ADS-B Out is the part that will broadcast the whole system where later be continuously transmitting aircraft data set to the ground receivers. The ADS-B transmitter, either the 1090 MHz Mode S transponder or the 978 MHz UAT, is the minimum equipment required for the ADS-B Out capability. ADS-B In is the receiver that allows any equipped aircraft to receive and interpret the ADS-B Out data on an electronic flight bag installed in the cockpit from other participating aircraft. The functionality of the ADS-B needs an authorized scheme of ADS-B Out [5].

1.1.2 The current implementation of ADS-B system using the ground station

The signals coming from the ADS-B currently will be tracked by ground-based receivers called ground stations, this is the ground segment of the ADS-B system. In most cases, the output data of these ground stations will later be sent to the Surveillance Data Processing and distribution systems attached with the inputs from other surveillance sensors and receivers, such as radars and multilateration, in order to establish a Traffic Situation Picture for the users [6].

Altitude, distance from the stations' site, and obstructing terrain are some of the constraints that affect the ability of the ground stations to receive the signals coming from ADS-B transponders. The optimum range of coverage for a ground station is about 200 nautical miles, where it can be extended to a maximum of 250 nautical miles. Due to this limitation, the general implementation of terrestrial based air traffic control, either radar based or the ADS system, will obviously encounter the geographical issues such as low quality of coverage in secluded and inaccessible regions such as the highlands and at the territorial sea where ground stations and towers aren't feasible or excessively expensive to be installed [7].

Indonesia is known as an archipelago with a lot of small islands. It is located along the equator with wideness more than 5000 kilometres. High terrain and oceanic area are the main characteristics of the Indonesian region that makes it harder to have full coverage using traditional ground-based radar. Because of the limitation on the ground station ADS-B instalment, a large number of ground stations are needed to have full coverage in Indonesia. There are currently 31 ground stations installed across the Indonesia area. The development phase that has been started by the Indonesian government since 2007 cost around 15 billion rupiahs, with comparison to imported ADS-B ground station instalment around 10 billion rupiahs each [8].

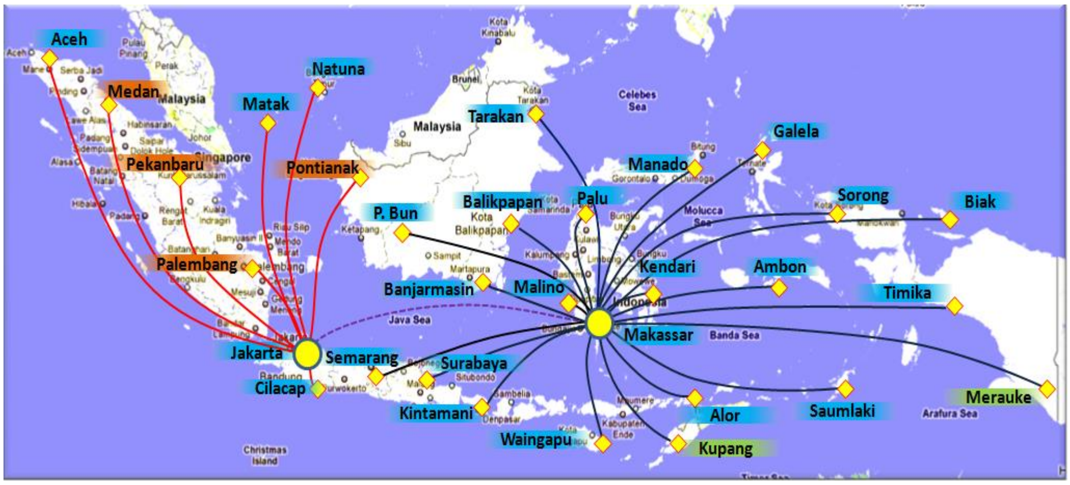


FIGURE 1.1: ADS-B Integration in Indonesia [9].

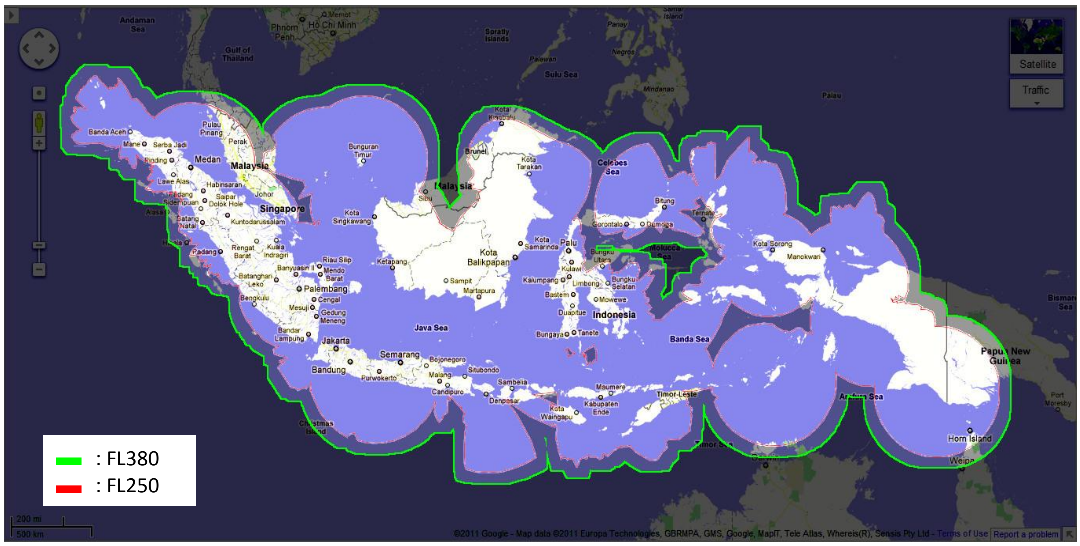


FIGURE 1.2: ADS-B Coverage Simulation in Indonesia [9].

The absence of real-time aircraft stating vectors (location) data in distant regions will considerably hamper ideal planning and control procedures as bigger lateral and vertical minimum separations are needed until the aircraft reaches airspace controlled radar or ground stations. Eventually, it will become a bottleneck problem for the search and rescue process in the remote and uncontrolled areas. The mishap that happened with the MH-370 is the perfect example of the importance of real-time aircraft positioning and situational awareness to enhance the flight safety and SAR assistance of aircraft in remote areas [4, 10].

1.1.3 Air Traffic Monitoring and Surveillance with the aid of satellite communication

The conventional surveillance method used by the ATC utilized the ground radar-based system. This system usually consists of the primary surveillance radar (PSR) and the secondary surveillance radar (SSR), which complemented the surveillance of the radio system. Increasing air traffic in the coverage region of the radar directly affects the radar saturation. Due to the enormous increase in air traffic, especially in the Indonesian region, the radar systems are getting saturated. Additionally, the radar based sensing has drawbacks, such as, its loudness and susceptibility to signal interference and deteriorating signal integrity [6].

Since the transceiver of ADS systems can recognize two aircraft at the same geographical position and location, ADS systems will saturate much slower than conventional radars. ADS systems is also able to interrogate single aircraft transponder discreetly. Compared to radar system, the signal from ADS systems propagates superiorly in the Earth's atmosphere compared to radar signals. Consequently this improves the effectiveness, thereby improving the determination of aircraft status vectors [10].

The ability to enhance the aircraft location determination provided by ADS system can also be supported by the satellite constellations even further by relaying the ADS information of the aircraft to other receivers without significantly limited by geographical barriers or constraints. Communication between satellites allows the link redundancy to be higher. Should an ATC is unable to function properly, for example due to natural disaster, the information collected the constellation are easily relayed to other ATC. The satellite-based ADS-B communication system is indeed a necessary solution strategy in distant and oceanic regions to relay aircraft signals and data.

In sum, by improving the network connectivity between the aircraft and the satellites that form a constellation, allows the performance of ATC system to be enhanced both in speed and in reliability. Further improvements are still also possible by augmenting the communication protocol and the transponder between aircraft and by designing an optimum constellation [11, 12].

1.2 Problem Statement

This research is primarily aimed to design the constellation of Low Earth Orbit (LEO) microsatellites for air traffic monitoring and surveillance in Indonesia.

Although the satellite constellation could also be extended to cover other purpose such as maritime traffics or land-based transportation, but here the subject of this thesis is limited as stated above.

1.3 Research Purpose

Based on the problem statement above, then the purposes of this thesis are:

- Constellation configuration: type and number of orbital planes,
- Constellation size: number of member of each orbital plane,
- Age of constellations: under the zonal gravitational perturbation (under the influence of earth nonsphericity) and atmospheric drag with more than 90% coverage percentage.

1.4 Research Scope

This research will have the initial assumption as follows:

- The area of coverage for the constellation design is the Indonesian region
- Current satellite-based ADS-B receiver protocol and characteristics will be used as the benchmark to design and evaluate the constellation.
- Orbit perturbation that will be considered is the zonal gravitational perturbation up to J_2 or the second order.

The limitation of this research includes:

- Orbit determination excluding the constellation deployment method and cost analysis.
- Age of constellation estimation without the presence of orbit maintenance.

1.5 Research Approach

The approach that will be taken into account in this research is by utilizing numerical analysis and simulation. The simulation tools and algorithm will be constructed using Python, while the orbital system that will be considered is the two-body problem.

Chapter 2

Theoretical Background

2.1 Overview of Satellite Orbit

2.1.1 Orbit Definition

An orbit is a periodic and repeated route taken around another by an object in space. The specific object running in the orbit is called a satellite, where the satellite itself can be natural, such as the moon as the Earth's satellite, or man-made, such as the International Space Station, etc. Most planets have moons orbiting them.

The Earth and any other planets in the solar system, as well as comets, asteroids and other items, are also satellites orbiting the sun. Most of those natural objects moving along or near an imaginary flat surface (ecliptic plane) while orbiting the sun [13].

2.1.2 Satellite Equations of Motion

According to [14] the motion of planets and any other space objects, under two-body problem, will follow cone section orbits based on the Newtonian mechanics. In deriving the equation motion under two body problem, basically we combine the Newton's second law of motion, with the Newton's law of universal gravitation. When investigating the motion of the Earth's satellites, at first we can approach the two bodies, the Earth, as the primary body as a point mass, and the mass of the satellite, compared with the Earth is negligible. The force due to potential gravity according to Newton can be expresses as:

$$\vec{F_g} = -Gm \sum_i \frac{M_i}{r_i^3} \vec{r_i} \tag{2.1}$$

Since the focus of interest is the satellites in orbits about single planets, eventually the equation 2.1 will reduce to the two bodies (planet and satellite of interest) form. Consequently, the equation for the force magnitude induced by gravity finally becomes:

$$\vec{F_s} = -G \frac{Mm_s}{r^3} \vec{r_s} \tag{2.2}$$

$$\vec{F_s} = -\frac{\mu m_s}{r^3}\vec{r_s} \tag{2.3}$$

where F_s is the magnitude of the force induced by gravity, G is the universal gravity standard, M is the Earth's mass, m_s is the satellite's mass, r is the distance from the Earth's core to the satellite, and μ = GM is the Earth's gravity constant with the value is approximately 398,600.5 $\frac{km^3}{s^2}$

By combining Eq. 2.4) with Eq. 2.2), we can derive the acceleration of the satellite as a function of position explicitly and implicitly also a function of time. The derivation steps are as shown:

$$\sum \vec{F_s} = m_s \frac{d^2\vec{r}}{dt^2} \tag{2.4}$$

$$\sum \vec{F_s} = m_s \frac{d^2\vec{r}}{dt^2} = -G\frac{Mm_s}{r^3}\vec{r_s} \tag{2.5}$$

$$\frac{d^2\vec{r}}{dt^2} + G\frac{M}{r^3}\vec{r_s} = 0 \tag{2.6}$$

The equation 2.6 is well known as equation of motion of restricted two-body problem. This equation is the comparative equation of a satellite position vector's movement as the Earth orbits the satellite. The early step derivation starts with the assumption that gravity is the only force acting and the shape of the Earth is spherically symmetric and as stated previously, the mass of the satellite is negligible compared to the mass of the primary body and no other third bodies' gravitational pull are included in the consideration.

The solution of the two-body equation of motion for any satellites orbiting the Earth is the polar equation of a conic section, which means the orbit trajectory would be in patched conic section.A conic section can be defined as a curve created by a plane crossing a correct circular cone. This equation will eventually define the magnitude of the position vector as to where the orbit is located.,

$$r = a\frac{(1-e^2)}{(1+e\cos\nu)} \tag{2.7}$$

where a is the semi-major axis, e is the eccentricity, and ν is the polar angle or true anomaly. More details on orbital elements will be given in the next section.

2.1.3 Classical Orbital Elements

Analytical solution to the restricted two-body problem can be parameterized into a set of constants. This set of constants are also known as Keplerian elements or classical orbital elements. The Keplerian elements provide geometrical representation: size and orientation of an orbit, and without perturbation, the values are constant.

As shown in figure 2.1, the classical orbital elements consist of six parameters that define the specification of an orbit. Those elements are Semi-major axis (a), Eccentricity (e), Inclination (i), Longitude of the ascending node (Ω) or Right ascension of the ascending node (*RAAN*), Argument of perigee (ω), and True anomaly (ν, θ, or f).

The semi-major axis (a) defines the size of the orbit. It is half the sum of an orbit's apogee and perigee distances. For a circular orbit, the semi-major axis is the distance between the centres of the bodies. The eccentricity (e) defines the shape of the orbit, where zero value is a circular shape and eventually goes ellipse, parabolic and hyperbolic shapes as the value of the eccentricity going higher.

The governing equations are:

$$a = \frac{1}{2}(rp + ra) \tag{2.8}$$

$$e = \sqrt{\mathbf{e} \cdot \mathbf{e}} \tag{2.9}$$

$$i = \cos^{-1}(\frac{h_Z}{h}) \tag{2.10}$$

$$\Omega = \cos^{-1}(N_X/N) \tag{2.11}$$

$$\omega = \cos^{-1}(\mathbf{N} \cdot \mathbf{e}/N_e) \tag{2.12}$$

$$\theta = \cos^{-1}\left(\frac{\mathbf{e} \cdot \mathbf{r}}{er}\right) \tag{2.13}$$

The inclination (i) is the vertical tilt of the orbital plane with respect to the equatorial plane as the reference plane. The reference plane and the orbital plane are defined in three-dimensional space as two-dimensional objects. The longitude of the ascending node (Ω) or right ascension of the ascending node (*RAAN*) is the angle to the ascending node from the vernal equinox. The ascending node itself is the point at which the satellite moves from south to north through the equatorial plane. This right ascension is measured as a rotation law on the pole on the right-handed rule. Both elements define the orientation of the orbital plane.

The argument of perigee (ω) provides the location of the perigee of the orbit with respect the ascension node at the direction of the satellite motion. The true anomaly ($\theta, f, \text{or}, \nu$), is the distance angle of the satellite from the perigee; this is equivalent with the angle formed by the eccentricity vector with the vector position of the satellite.

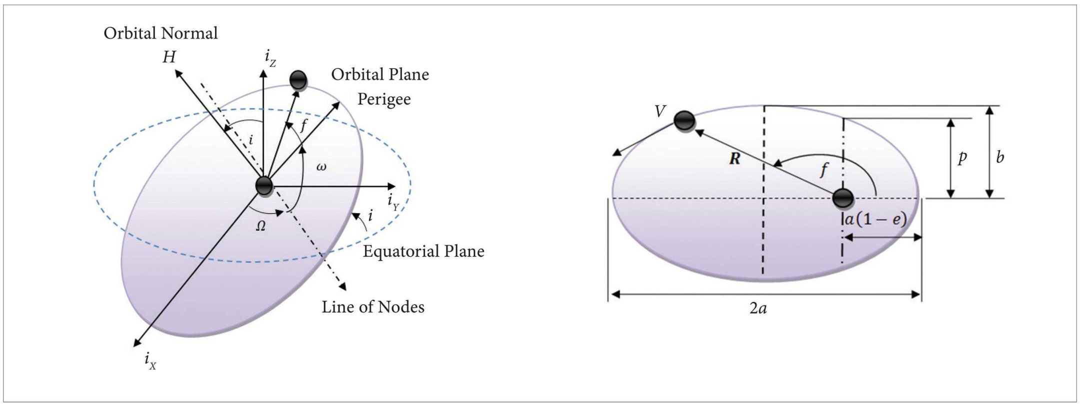


FIGURE 2.1: Classical orbital elements [15].

2.2 Orbital Elements and The State Vector

In order to uniquely determine the trajectory of satellites in space, defining satellites orbit state vectors will help. State vectors or orbital state vectors of an orbit are the Cartesian vector of position (**r**) and velocity (**v**). State vectors are defined for multiple reference frames, usually an inertial reference frame, where the Earth Centered Inertial (ECI) frame is the most common reference frame to be used. The position vector (**r**) describes the position of satellites, while the velocity vector (**v**) describes the velocity of the satellites both in the reference frame chosen and at the same time.

State vectors for satellites may be helpful in calculating classical or Keplerian orbital elements for satellites and vice versa. As mentioned in the previous chapter, the classical orbital elements will help describe the size, shape, and orientation of the satellite orbit and can be used to estimate the satellite status at any arbitrary time given its movement that is accurately modelled by a two-body problem assumption with only minor disturbance. On the other side, the state vectors will be helpful for numerical integration that accounts for important, arbitrary, time-varying disturbances such as drag as well as main body gravitational disturbance. Hence, knowing both state vectors and orbital elements representation will have its own advantages.

2.2.1 Transforming Satellites State Vectors into Classical Orbital Elements

Knowing a satellite's position vector (**r**) and velocity vector (**v**), there are several steps in order to obtain classical orbital elements as well as some other useful parameters. The algorithm outlines are:

1. Calculating the range,

$$r = \sqrt{\mathbf{r}\cdot\mathbf{r}} = \sqrt{X^2 + Y^2 + Z^2} \tag{2.14}$$

2. Calculating the magnitude of the velocity,

$$v = \sqrt{\mathbf{v} \cdot \mathbf{v}} = \sqrt{v_X^2 + v_Y^2 + v_Z^2} \tag{2.15}$$

3. Calculating the radial component of the velocity,

$$\boldsymbol{v_r} = \mathbf{r} \cdot \mathbf{v}/r = (Xv_X + Yv_Y + Zv_Z)/r \tag{2.16}$$

4. Calculating the specific angular momentum,

$$\mathbf{h} = \mathbf{r} \times \mathbf{v} = \begin{bmatrix} \hat{\mathbf{I}} & \hat{\mathbf{J}} & \hat{\mathbf{K}} \\ X & Y & Z \\ v_X & v_Y & v_Z \end{bmatrix} \tag{2.17}$$

5. Calculating the magnitude of the specific angular momentum,

$$h = \sqrt{\mathbf{h} \cdot \mathbf{h}} \tag{2.18}$$

6. Calculating the inclination angle,

$$i = \cos^{-1}(\frac{h_Z}{h}) \tag{2.19}$$

7. Calculating the vector of node line,

$$\mathbf{N} = \hat{\mathbf{K}} \times \mathbf{h} = \begin{bmatrix} \hat{\mathbf{I}} & \hat{\mathbf{J}} & \hat{\mathbf{K}} \\ 0 & 0 & 1 \\ h_X & h_Y & h_Z \end{bmatrix} \tag{2.20}$$

8. Calculating the magnitude of the node line,

$$N = \sqrt{\mathbf{N} \cdot \mathbf{N}} \tag{2.21}$$

9. Calculating the right ascension of the ascending node,

$$\Omega = \cos^{-1}(N_X/N) \tag{2.22}$$

10. Calculating the eccentricity vector,

$$\mathbf{e} = \frac{1}{\mu}\left[\mathbf{v} \times \mathbf{h} - \mu\frac{\mathbf{r}}{r}\right] = \frac{1}{\mu}\left[\mathbf{v} \times (\mathbf{r} \times \mathbf{v}) - \mu\frac{\mathbf{r}}{r}\right] = \frac{1}{\mu}\left[\mathbf{r}v^2 - \mathbf{v}(\mathbf{r} \cdot \mathbf{v}) - \mu\frac{\mathbf{r}}{r}\right] \tag{2.23}$$

so that,

$$\mathbf{e} = \frac{1}{\mu}\left[(v^2 - \frac{\mu}{r})\mathbf{r} - rv_r\mathbf{v}\right] \tag{2.24}$$

11. Calculating the magnitude of the eccentricity,

$$e = \sqrt{\mathbf{e} \cdot \mathbf{e}} \tag{2.25}$$

12. Calculating the argument of perigee,

$$\omega = \cos^{-1}(\mathbf{N} \cdot \mathbf{e}/N_e) \tag{2.26}$$

13. Calculating the true anomaly,

$$\theta = \cos^{-1}\left(\frac{\mathbf{e} \cdot \mathbf{r}}{er}\right) \tag{2.27}$$

14. Calculating the period of the orbit,

$$T = \frac{2\pi}{\sqrt{\mu}} a^{\frac{3}{2}} \tag{2.28}$$

15. Calculating the perigee and apogee radii

$$r_p = \frac{h^2}{\mu} \frac{1}{1 + e\cos(0)} \tag{2.29}$$

$$r_a = \frac{h^2}{\mu} \frac{1}{1 + e\cos(180°)} \tag{2.30}$$

16. Calculating the semi-major axis.

$$a = \frac{1}{2}(rp + ra) \tag{2.31}$$

Hence from all the numerical calculations, we can define each of the six orbital elements.

2.3 Transformation Between Geocentric Equatorial and Perifocal Frames

The perifocal frame is the satellite orbit's natural frame. It's centered at the orbit's focus. Its $\bar{x}$ and $\bar{y}$ plane shows the plane of the orbit where the x-axis is directed from the focus through the periapse/perigee to the eccentricity vector. The figures 2.2 will illustrate the relationship between perifocal and geocentric equatorial frames. It is so essential to convert the orbital elements of Kepler into a geocentric vector of the

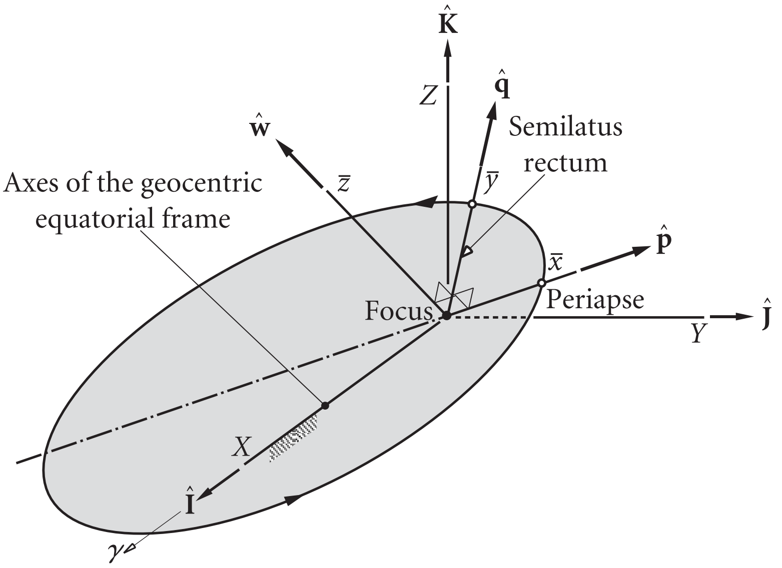


FIGURE 2.2: The relationships between Perifocal and Geocentric frame [16].

equatorial frame state. Since the orbit lies in the $\bar{x}$ and $\bar{y}$ plane, the state vector components of a body associated with its perifocal reference are:

$$\{\mathbf{r}\}_{\bar{x}} = \frac{h^2}{\mu}\frac{1}{1+e\cos\theta}\begin{Bmatrix}\cos\theta\\ \sin\theta\\ 0\end{Bmatrix} \tag{2.32}$$

$$\{\mathbf{v}\}_{\bar{x}} = \frac{\mu}{h}\begin{Bmatrix}-\sin\theta\\ e+\cos\theta\\ 0\end{Bmatrix} \tag{2.33}$$

As shown by Fig. 2.3, we can transform any vector from the geocentric equatorial frame into the perifocal form by using typical Eulerian angle of transformation. The standard sequence transformation can use **K** unit axis as the first rotation. This rotation angle is the ascending node Ω, and the matrix transformation for this can be obtained as:

$$[\mathbf{R}_3(\Omega)] = \begin{bmatrix}\cos\Omega & \sin\Omega & 0\\ -\sin\Omega & \cos\Omega & 0\\ 0 & 0 & 1\end{bmatrix} \tag{2.34}$$

The second rotation uses the inclination angle (i) and the transformation matrix is obtained as:

$$[\mathbf{R}_1(i)] = \begin{bmatrix}1 & 0 & 0\\ 0 & \cos i & \sin i\\ 0 & -\sin i & \cos i\end{bmatrix} \tag{2.35}$$

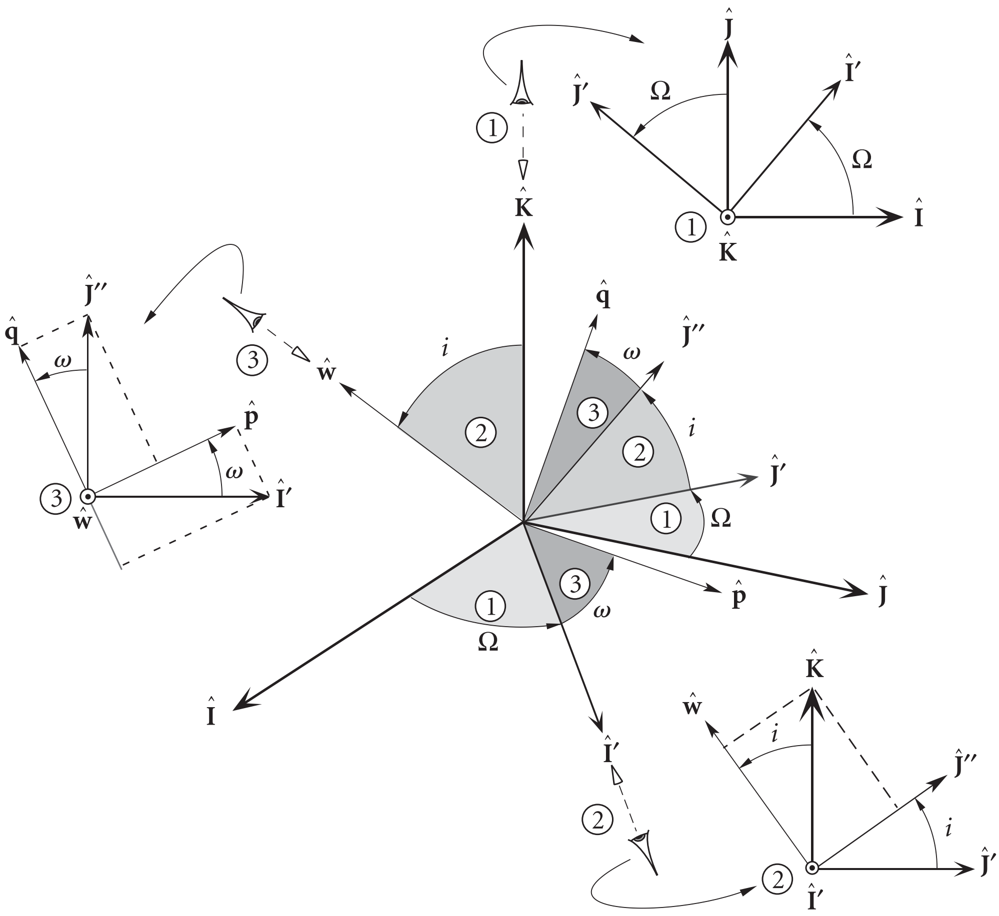


FIGURE 2.3: The transformation rotation sequence [16].

For the last rotation, the argument of perigee is used (ω), and the transformation matrix is:

$$[\mathbf{R}_3(\omega)] = \begin{bmatrix} \cos\omega & \sin\omega & 0 \\ -\sin\omega & \cos\omega & 0 \\ 0 & 0 & 1 \end{bmatrix} \tag{2.36}$$

The combined transformation from the geocentric equatorial frame into the perifocal frame can be obtained as product of all the Eulerian sequence of transformation. Here the matrix transformation is denoted as $[\mathbf{Q}]_{X\bar{x}}$, and the elements of the matrix is given by Eq. 2.37.

$$[\mathbf{Q}]_{X\bar{x}} = [\mathbf{R}_3(\omega)][\mathbf{R}_1(i)][\mathbf{R}_3(\Omega)] \tag{2.37}$$

The transformation matrix is orthogonal so the relationship between the inverse with the transpose is basically equivalence. This means that $([\mathbf{Q}]_{X\bar{x}})^{\mathbf{T}} = [\mathbf{Q}]_{\bar{x}X}$, hence:

$$[\mathbf{Q}]_{\bar{x}X} = \begin{bmatrix} \cos\Omega\cos\omega - \sin\Omega\sin\omega\cos i & -\cos\Omega\sin\omega - \sin\Omega\cos i\cos\omega & \sin\Omega\sin i \\ \sin\Omega\cos\Omega + \cos\Omega\cos i\sin\omega & -\sin\Omega\sin\omega + \cos\Omega\cos i\cos\omega & -\cos\Omega\sin i \\ \sin i\sin\omega & \sin i\cos\omega & \cos i \end{bmatrix} \tag{2.38}$$

If we want to simply transform from the geocentric to perifocal only the position part then [16]:

$$\left\{\mathbf{r}\right\}_X = [\mathbf{Q}]_{\bar{x}X}\left\{\mathbf{r}\right\}_{\bar{x}} \tag{2.39}$$

Likewise if we want to only transform the velocity part, then it can be done as:

$$\left\{\mathbf{v}\right\}_X = [\mathbf{Q}]_{\bar{x}X}\left\{\mathbf{v}\right\}_{\bar{x}} \tag{2.40}$$

2.4 Orbit Types and Classifications

2.4.1 Orbit Type based on the Altitude

Table 2.1 shows the classification of an orbit based on its altitude and will further be discussed in this section.

TABLE 2.1: The classification of an orbit based on its altitude.

Orbit Type	**Altitude**
Low Earth Orbit (LEO)	160-2000 km
Medium Earth Orbit (MEO)	2000-35790 km
High Earth Orbit (HEO)	above 35790 km

Low Earth Orbit

Low Earth Orbit (LEO) is typically a circular or elliptical orbit, and technically objects or satellites in this specific orbit will be above the Earth's surface at an altitude between 160 and 2000 km (99 to 1200 mi). Beyond the 2000 km altitude, the orbit type will swift into Middle Earth Orbit (MEO). The satellites will also suffer from orbital decay below the 160 km altitude and thus sink quickly into the Earth's atmosphere.

The low objects altitude in this orbit makes the orbital period become much shorter than any other types of orbit. Normally objects in this orbit will have an orbital period of between 88 to 127 minutes. Also, low earth orbit satellites will rapidly alter its position in relation to the ground position.

A large number of LEO satellites are necessary for missions requiring uninterrupted connectivity and coverage since the ground coverage radius will be limited for every orbital plane, even for local or regional applications. Because of this particular reason, LEO satellites are often part of satellite constellations.

The main reason for the utilization of the LEO satellites is because the cost to launch into orbit is much less expensive compared to geostationary satellites, as well as its development cost. Also, due to its proximity to the ground, LEO satellites will only require relatively low signal strength.

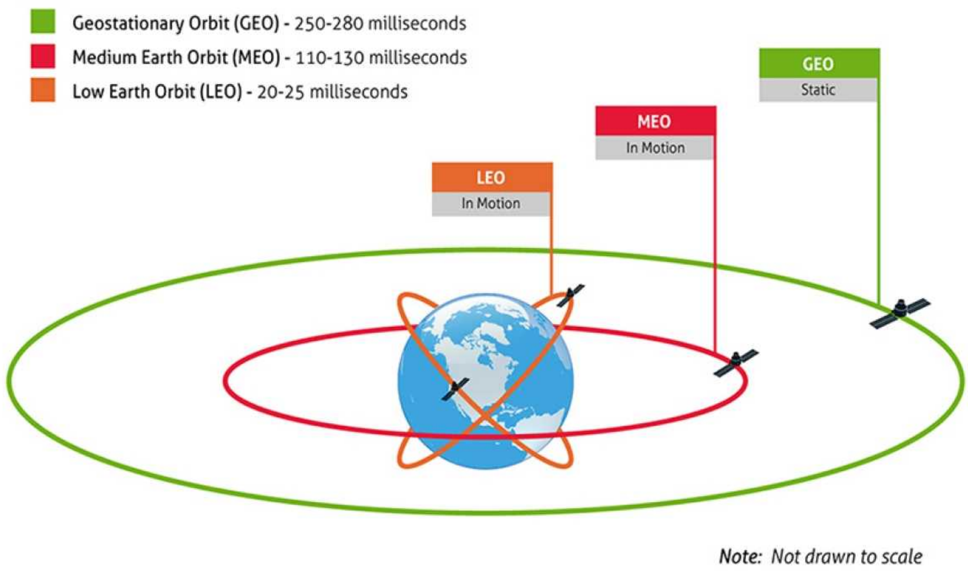


FIGURE 2.4: Types of orbit based on orbital altitude [17].

Medium Earth Orbit

Medium Earth Orbit (MEO), sometimes referred to as Intermediate Circular Orbit (ICO), is a satellite orbit whose altitude is between Low Earth Orbit and Geosynchronous Orbit. The altitude is between 2000 km up to an altitude of about 35790 km above Earth's surface.

Using this orbit is frequently used for artificial satellites with a particular navigation, communication, and geodetic / space environment scientific mission. The prevalent altitude to be used in this region is usually about 20,000 km, resulting in an orbital period of about 12 hours as used by the Global Positioning System (GPS). Another instance of the satellites using MEO is the 19,100 km altitude GLONASS and the 23,222 km altitude Galileo. Also placed in the MEO are the communications satellites covering the North and South pole.

High Earth Orbit

High Earth Orbit is a geo-centered orbit above the geosynchronous orbit (which will be further explained in the special orbit subsection) or more than 35,790 km. The orbital periods of the objects in this orbit are usually greater than 24 hours, so that the satellites have a retrograde motion (opposite of Earth rotation), and even if they are in a prograde orbit (as the Earth rotation), their orbital velocity is much smaller than the rotational velocity of the Earth, causing their ground track to move west on the surface of the Earth [18, 19].

2.4.2 Orbit Shapes

If we consider two gravitational bodies which interact with each orbit, their orbit characteristic will follow a conic section. The orbit can be an open orbit which indicates that the object will never return and is closed orbit which indicates the returning object. This classification is dependent on the system's complete energy. In the case of an open orbit, the velocity of the object at any position in the orbit is at least the velocity of escape for that position, while the velocity of escape for a closed orbit is always lower than the velocity of escape.

TABLE 2.2: The orbit shapes based on its eccentricity.

Orbit Shape	**Eccentricity**
Circular orbit	e = 0
Elliptic orbit	0 < e < 1
Parabolic orbit	e = 1
Hyperbolic orbit	e > 1

For an open orbit, when the velocity of the object in the orbit is exactly the same as escape velocity then the orbit will have a parabolic shape. But when the object's velocity is greater than the escape velocity then the orbit will be a hyperbola. When bodies that have escape velocity or higher approach each other, at the moment of their closest approach they will shortly curve around each other and then be forever divided.

For all closed orbit, they will have an ellipse shape. The circular orbit is a special case, and these shapes will be determined by its orbit eccentricity. The eccentricity price for a circular orbit is 0, whereas the eccentricity value for an ellipse form is between 0 and 1. For a parabolic shape, it has eccentricity value of 1 and hyperbolic has value more than 1. This overall classification is shown in the table 2.2

2.4.3 Special Orbit Type

The Polar Orbit is the first sort of unique orbit, but in fact the right word is near-polar orbit. This particular orbit will have an inclination close to 90°. The advantage of this sort of orbit is to enable the satellite to see nearly every portion of the Earth as the Earth rotates below this trajectory of orbit. The period of the orbit is more or less 90 minutes for a full rotation. A polar orbit is usually used for measuring atmospheric elements and characteristic mission.

The second form is the Sun Synchronous orbit, enabling a satellite to cross a segment of the Earth at the same moment of a day. Since there are 365 days in a year and 360° in a circle, this orbit's satellite has to alter its orbit by about one degree a day. The altitude of this orbit usually lays in between 700 to 800 km. The satellites in this orbit utilize the fact of the oblateness of the Earth, where the additional gravitational force acting on the satellite is taken into account. This will render either proceeding or receding the satellites orbit.

This orbit is generally used for missions requiring a steady quantity of sunlight. Satellites with an Earth imaging mission, for instance, would operate best with bright sunlight, while satellites with long-wave radiation mission job best in full darkness.

The next form is called the orbit of Geosynchronous or also known as the orbit of geostationary. The satellites in this orbit will circle the Earth at about the same pace of rotation on the Earth. The altitude is about 35,790 km above the Earth surface to make these satellites have the same duration as the Earth. Considering the Earth-shaped bulge, the satellites in this orbit will be situated at the equator as there is a steady gravitational force from all directions at this latitude place. While the bulge situation at the center of the Earth will take the satellite nearer to the Earth for other latitudes.

The advantage of using this orbit type is to allow Earth's near-full hemisphere to be observed by satellites in this orbit. Usually these satellites are used to study weather phenomena like hurricanes or cyclones on a big scale. The other utilization is for communication satellites. However, since the satellites in this orbit are located very far away, usually these satellites will have a very poor resolution. The other drawback is that there will be bad coverage of these satellites around the pole region [20].

2.5 Orbit Perturbation

The orbital elements mentioned at the beginning of this chapter provide the best method in order to define satellites orbit. However, there are other forces affecting the orbit of the satellites that draw it away from its normal orbit. Thus, these disturbance forces will cause the orbital elements to vary, they can be categorized based on how they impact the orbital elements of Kepler or classical.

The first classification is secular variations representing a linear variation of the orbital elements, while the second is short-period variations that periodically affect the orbital elements with a period less than the satellite's own orbital period. The last classification is long-term differences with a period longer than the satellite's orbital period. Since secular differences have long-term impacts on orbit determination where the rate of change of the impacted orbital elements will always increase or reduce, our study will be focused and further discussed.

2.5.1 Perturbations due to Non-spherical Earth

When defining the orbit of a satellite, the most simpler way to define the motion of arbitrary particles, subject only to a few restrictions, is by using a two body problem

TABLE 2.3: The list of planetary oblateness and second zonal harmonics.

Planet	**Oblateness**	**J2**
Mercury	0.000	60 x 10^{-6}
Venus	0.000	4.458 x 10^{-6}
Earth	0.003353	1.08263 x 10^{-3}
Mars	0.00648	1.96045 x 10^{-3}
Jupiter	0.06497	14.736 x 10^{-3}
Saturn	0.09796	16.298 x 10^{-3}
Uranus	0.02293	3.34343 x 10^{-3}
Neptune	0.01708	3.411 x 10^{-3}
Pluto	0.000	-
(moon)	0.0012	202.7 x 10^{-6}

approach. We could initially assume that the Earth is spherically symmetrical with homogeneous mass when developing the two-body equations of motion, in this case the satellite and the Earth. But the Earth is actually neither spherical nor homogeneous. The Earth's most dominant characteristics of form are the equator bulge, a slight pear-random shape, and the Earth's poles flattening portion. We can identify the acceleration of the satellites for a prospective Earth function by taking the gradient of the potential function. While the geopotential function's most commonly used form relies on the Jn latitude and geopotential coefficient or we call it zonal coefficients.

Periodic differences in all Keplerian or classical orbital elements are caused by the potential produced by the non-spherical Earth disturbance. The dominant impacts of this disturbance are the secular differences in the longitude of the ascending node ($RAAN$) and argument of perigee (ω) due to the oblateness form of the Earth, represented by the dimensionless parameter J_2 that quantifies the significant impact of oblateness on orbit and geopotential expansion. The rates of change of the dominant variations of and due to J_2 are:

$$\dot{\Omega} = -\frac{3}{2} n J_2 \left(\frac{RE}{p}\right)^2 \cos(i) \tag{2.41}$$

$$\dot{\omega} = \frac{3}{4} n J_2 \left(\frac{RE}{p}\right)^2 (5\cos^2(i) - 1) \tag{2.42}$$

where n is the mean motion in degrees/day, J_2 has the value 0.00108263, RE is the Earth's equatorial radius, p is the semi-latus rectum in kilometres, i is the inclination in degrees/day.

The primary reason we consider the disturbance due to non-spherical Earth up to J_2, apart from its secular variation, is because the disturbances of J_2 will dominate for the satellites in GEO and below, while the disturbances of the Sun and Moon will dominate for the satellites above GEO [14, 16].

2.5.2 Evaluating the Effects of The Earth's Oblateness

As explained in the previous subsection about the potential perturbation generated by the oblateness of the Earth, the J_2 coefficient is about 1000 times greater than the next biggest aspheric coefficient and is therefore very crucial when describing the motion of a satellite around the Earth. Now we will take into account this dominant perturbation force to get the potential arising due to J_2 disturbance as shown in the equation 2.48.

Gravitational Potential with J_n disturbance:

$$U_{zonal} = -\frac{\mu}{r} J_n \left(\frac{Re}{r}\right)^n P_n[\cos(\phi_{gc_{sat}})] \tag{2.43}$$

for J_2 order:

$$U_{zonal} = -\frac{\mu}{r} J_2 \left(\frac{Re}{r}\right)^2 P_2[\cos(\phi_{gc_{sat}})] \tag{2.44}$$

where $P_2[\cos(\phi_{gc_{sat}})]$ is associated Legendre polynomial of J_2, and The co-latitude may be written as (Ginn, 2006):

$$\sin^2(\phi_{gc_{sat}}) = 1 - \frac{Z^2}{r^2} \tag{2.45}$$

The equation 2.44 can further be reduced to:

$$U_{zonal} = -\frac{\mu J_2 Re^2}{2r^3}(3\cos^2(\phi_{gc_{sat}}) - 1) \tag{2.46}$$

Finally, the J_2 acceleration in the ECI frame is calculated as the potential gradient

$$\nabla U_{zonal} = \bar{J_2} = \begin{bmatrix} \frac{\partial U_{zonal}}{\partial \vec{x}} \\ \frac{\partial U_{zonal}}{\partial \vec{y}} \\ \frac{\partial U_{zonal}}{\partial \vec{z}} \end{bmatrix} \tag{2.47}$$

So;

$$\nabla U_{zonal} = -\frac{3}{2}\frac{\mu J_2 Re^2}{r^4} \begin{bmatrix} \left(1 - 5\frac{\vec{Z}^2}{r^2}\right)\frac{\vec{X}}{r} \\ \left(1 - 5\frac{\vec{Z}^2}{r^2}\right)\frac{\vec{Y}}{r} \\ \left(3 - 5\frac{\vec{Z}^2}{r^2}\right)\frac{\vec{Y}}{r} \end{bmatrix} \tag{2.48}$$

2.5.3 Effect of Atmospheric Drag on Satellite Orbits

The satellites around the Earth will have direct contact with the upper atmosphere in the low altitude orbit. Although the air in the upper atmosphere has a very small mass density, when the exposures in the orbit between the upper atmosphere and the satellites are sufficiently long, this interaction can considerably alter the orbit of the satellite, causing it to decline over time. It is possible to model the force per unit mass the atmosphere exerts on an orbiting satellite as:

$$F = -\frac{1}{2}\frac{\rho C_D A}{m}\dot{r}\dot{\mathbf{r}} \tag{2.49}$$

where ρ is the mass density of the atmosphere (at the satellite's position), C_D the drag coefficient, A the cross-sectional area of the satellite perpendicular to its direction of motion, m the satellite mass, and $\dot{\mathbf{r}}$ the satellite velocity. It can be seen that the force is proportional to the square of the satellite velocity and is directed contrary to its instantaneous movement direction. However, the drag coefficient itself is a dimensionless constant of unity of order that depends mainly on the form of the satellite. Compared to the force of gravitational attraction between the Earth and the satellite, the atmospheric drag is low and can therefore be handled as disturbance although, once again, it is much smaller compared to the perturbations due to non-spherical Earth.

2.5.4 Orbit Maintenance

Most of them will not need extra orbit adjustment once the satellites are in their orbit. On the other side, when the perturbation forces have influenced them, mission demands may require orbit manoeuvre to correct their orbital elements. Two of the instances to be noted are satellites with repeated ground tracks and geostationary satellites as they require precise and accurate determination of the orbit.

Nevertheless, although the proposed plan of the constellation will eventually have repeating ground tracks, this research will only consider orbital design without any orbit maintenance and orbit adjustment. The rate of change of the orbital elements will be calculated and the age of the satellite orbit will then be estimated.

After the satellite's mission is already completed, several options of deorbiting options exist, depending on the type of the orbit. For the low earth orbit (LEO) satellites, it is possible to allow the satellites to decay and re-enter the atmosphere. Satellites can also be boosted into benign orbit at all altitudes to decrease the likelihood of collision with active payloads or satellites.

2.6 Types of Satellites

As already defined in the early part of this chapter, we understand there are two satellite types: natural or artificial. In terms of artificial or man-made satellites, it can be categorized into distinct mission- and size-based kinds. The various type of satellites is shown in the table 2.4 and 2.5

2.6.1 Weather satellites

Weather satellites are primarily used to monitor Earth's weather and climate circumstances. They are also renowned as satellites for meteorology. These satellites

TABLE 2.4: Types of satellites based on its mission.

Mission Based Satellites
Weather satellites
Communication satellites
Navigation satellites
Earth observation satellites
Astronomical satellites
Miniaturized satellites

continually evaluate the quantity of heat produced from the surface of the Earth. The meteorologist uses the data collected later to forecast the weather conditions. The weather satellites can capture the cloud mass in the sky that can be used to forecast the amount of precipitation. The weather satellites can also be used to predict critical weather conditions (e.g. storms, hurricanes).

Weather satellites are introduced into two orbit types that are either geostationary orbit or polar orbit. The geostationary orbit is used so that the weather satellites can easily capture the images utilizing its rest position referred to Earth's rotation while for polar weather satellites utilize their sun-synchronous characteristic to have a better imagery capability.

2.6.2 Communication satellites

The satellite of communication is a satellite designed primarily to transmits, for instance, lengthy distance communication signals, telephone, television, radio, and internet. The communication satellites utilize light signals or electromagnetic waves in order to transmit information wirelessly. However, after the Earth's curvature, light signals and electromagnetic waves can not bend. Multiple satellites are therefore required in a constellation to transmit the data or signal over lengthy distances by redirecting the signals between each satellite.

There are two kinds of satellites of communication: passive and active. Passive satellites obtain and redirect the electromagnetic signal from the source or transmitter of the signal to the receiver or destination. In contrast, an active satellite doesn't only provide relay function but also process, and if necessary amplify the signal it receives before transmitting it.

2.6.3 Navigation satellites

The basic function of navigation satellites is to provide transmitting signal that can be used by a receiver to find the its geographic location for example, ships and aircrafts. One of the most famous navigation satellite systems which provides global coverage as already explained earlier in this chapter is the Global Navigation Satellite System (GNSS).

2.6.4 Earth observation satellites

Observing the Earth condition from its orbit is the main purpose of the Earth observation satellites. These satellites will detect the changes in various characteristics of the Earth's surface including the Earth vegetation, ocean colour, radiation. The satellites of Earth observation can also be used as an imaging satellite to map the Earth's terrain. The different satellites for earth observation include proba-1, proba-2, ERS-1, ERS-2, and Environmental Satellite.

2.6.5 Astronomical satellites

The primary objective of astronomical satellites is to observe galaxies, stars, planets, natural satellites and other items in space. These satellites are frequently used with an astronomical telescope as the primary payload to discover fresh planets, stars, and galaxies. Hubble space telescope is one of the world's most popular and biggest human astronomical satellites ever. It is capable of capturing highly high-resolution space pictures.

2.6.6 Miniaturized satellites

Also commonly known as small satellites or smallsats, these satellites have a relatively low mass and small physical size. Usually the weight of small satellites is less than 600 kg. The reason for miniaturizing satellites is to reduce costs as heavier satellites require a larger and larger rocket thrust that requires higher financing costs. Based on its size, the miniaturized satellites can be categorized as shown in table 2.5.

TABLE 2.5: The classification of satellites based on its size.

Class of Satellites	**Size**
Small Satellites	100 - 500 kg
Microsatellites	10 - 100 kg
Nanosatellites	1 - 10 kg
Picosatellites	0.1 - 1 kg
Femtosatellites	below 100 gr

The term "small satellites" often referred to an artificial satellite with a wet mass of between 100 and 500 kg, which includes the satellite's fuel. But any satellite below 500-600 kg has come to mean this word. The microsatellites are the satellites which have a wet mass between 10 and 100 kg. However, microsatellites can also refer to satellites larger than 100 kg but still much lower than small satellites. The design purpose of microsatellites is usually to be installed in a constellation. The nanosatellites are the satellites with a wet mass between 1 and 10 kg. Like microsatellites, the nanosatellites design purpose is usually to be installed in a constellation. Both microsatellites and nanosatellites are actually able to be installed individually.

Nowadays, the advancement of technology increases the capability of nanosatellites to perform commercial missions that previously required microsatellites. The picosatellites and femtosatellites are the satellite with a wet mass in between 0.1 to 1 kg and below 100 gr respectively.

Specifically micro and nanosatellites also cover a unique type of satellite classification based on the shapes of the satellites. They are CubeSats, PocketQubes, TubeSats, SunCubes and non-standard picosatellites. CubeSats is the most common microsatellites configuration which has a wet mass between 0.2 and 40 kg and the dimension of 0.25 to 27 U. The U or Units in here is standard satellite dimensions of 10 cm x 10 cm x 10 cm [21, 22, 23].

2.7 Satellite Constellation Overview

2.7.1 The importance of Satellite Constellation in providing desired Earth Coverage mission

Coverage is one of the main parameters defining the mission requirement for any referenced task on Earth or planet. Calculating and estimating the coverage of certain mission can be so challenging since coverage is not uniform especially for Earth referenced mission.

A single satellite may be enough to provide full coverage for a certain region or local area. But, in order to achieve continuous coverage over a region, satellite constellation may provide better coverage performance and higher reliability as there are multiple satellites deployed for the mission. Constellation can also provide various circumstances necessary for the performance of a task, such as lighting variation for observation purposes and/or geometric variation for navigation and monitoring purposes.

2.7.2 Types of Constellation

There are vast options of constellations that may be used to satisfy a satellite's particular mission. Constellations are usually designed to have similar orbits, eccentricity and inclination for all satellites in the constellation so that the disturbances acting on each satellite are about the same. Because of this, the geometry can be maintained without unnecessary station-keeping so that fuel use will eventually be reduced and satellite life will be increased. Another consideration is that the phasing of each satellite in the orbital plane of the constellation retains adequate separation to prevent accidents and/or interference at junctions of the orbit plane. The sort of orbit frequently used in the constellation is circular orbits because then the satellite has to interact at a steady altitude requiring a steady force signal [24]. The types of constellations that are currently implemented are as follows:

2.7.3 Walker Delta Constellation

The most common used constellation design for global coverage of the Earth's surface is Walker Delta pattern constellation. This constellation is able to provide continuous coverage using the minimal number of satellites. A walker delta model includes complete t amount of satellites in p orbital planes in each orbital plane with $s = t/p$ satellites.

All orbital planes are presumed to be at the same angle of inclination in this specific constellation. The phase difference between satellites in an adjacent plane is defined as the angle of motion from the ascending node to the nearest satellite at a time when the satellite is in the westernmost plane with its ascending node. The notations of this constellation are:

$$i : t/p/f \tag{2.50}$$

where: i is the inclination; t is the total number of satellites; p is the number of equally spaced planes, and f is the relative spacing of adjacent planes between satellites. The swift in true anomaly (in degrees) in neighbouring satellite for equivalent satellites is $f * 360/t$.

Another part of this constellation's application is the near-polar walker star constellation that Iridium uses. The satellites in this orbit are roughly 180° in a near-polar circular orbit, travelling north on one side of the Earth and south on the other.

2.7.4 Disaster Monitoring Constellation

It is also commonly known as Disaster Monitoring Constellation for International Imaging (DMCii). There are a number of remote sensing satellites in this constellation that under the control of the International Charter for Space and Major Disasters provide emergency Earth imaging for disaster relief mission. Another utilization of this constellation is for a variety of civil applications by several countries. The spare imaging capacity available on a certain mission can also be sold under contract.

The DMC can provide much bigger imaging regions than generally developed public imaging satellites like Landsat. In order to leverage the knowledge and software of the big established remote sensing community used to operate with Landsat pictures, the DMC imagery was intended to compete with Landsat imagery capacity. DMC can deliver faster image results as it has numerous comparable satellites in orbit that are prepared to cross a point of interest and thus can generate much bigger pictures. Eventually, DMC will take up the responsiveness required for emergencies and catastrophe aid, providing pictures from the responsive satellites in the constellation over the Internet and the ground station of a member country within one day or less of a specific request [25].

2.7.5 Navigation Constellation

Navigation constellation is a sort of satellite constellation that offers signals to satellite navigation receivers that convey positioning and timing information. These information will then be used by the receivers to indicate the present place of specific objects. The constellation of navigation is called the Global Navigation Satellite System (GNSS). By definition, GNSS continually offers complete worldwide coverage. Examples of GNSS include Europe's Galileo, the USA's NAVSTAR Global Positioning System (GPS), Russia's Global'naya Navigatsionnaya Sputnikovaya Sistema (GLONASS) and China's BeiDou Navigation Satellite System.

GNSS performance is evaluated using four criteria:

1. Precision, defined as the difference between the measurement data of the receiver and the actual position, speed or time;
2. Integrity defined as the ability of a system to provide a confidence limit and in case of an anomaly in the positioning information;
3. Continuity identified as the capacity of a system to operate without disruption;
4. Availability defined as the proportion of moment a signal meets the above-mentioned precision, integrity and continuity requirements.

Nevertheless, GNSS is commonly known as the example walker delta constellation implementation to provide its global coverage which has notation 56°:24/3/1. This implies that in 3 planes there are 24 satellites with an inclination of 56°, spanning around the equator the 360°. The number 1 denotes phasing and spacing between the aircraft [26].

2.7.6 Evaluating Coverage Area of The Satellite

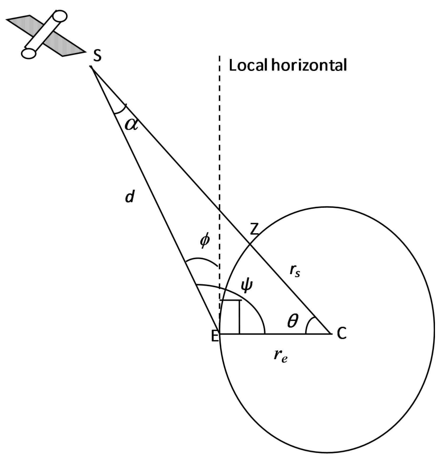


FIGURE 2.5: Ground station geometry [27].

The figure 2.5 will demonstrate how to analyze and evaluate satellite performance of a single satellite. The elements of the figure consists of Re as the radius of the Earth, h is the altitude of the satellite above the Earth surface, Rsat is the geocentric radius of the satellite found as the sum of Re and h, $\varphi$is the nadir angle, $\varepsilon$is the elevation angle, $\theta$is the central angle, slant range is the distance between ground station and the satellite denotes as D.

The minimum elevation angle takes a major role in the coverage performance analysis as it denotes the ability of the satellite to communicate with the ground stations. There are vast criteria to define the performance of satellite coverage. It may come from total coverage time over the region, coverage percentage, and coverage count. Overall coverage performance also needs to be defined as a percentage of the covered area over the whole Earth surface area [28, 29].

2.7.7 LEO Coverage Geometry

The basic geometry between a satellite and a ground station as already shown in figure 2.5 will give a good visualization about how we can evaluate the coverage area of a satellite. Let us define the three angles with new symbols as: ε is elevation angle, α is nadir angle, β is central angle. Once two quantities are known, the other equations can be found:

$$\varepsilon_0 + \alpha_0 + \beta_0 = 90 \tag{2.51}$$

$$d \cos \varepsilon_0 = r \sin \beta_0 \tag{2.52}$$

$$d \sin \alpha_0 = Re \sin \beta_0 \tag{2.53}$$

Besides the elevation, the slant range is also one of the most important parameters. It will later determine the relationship between the coverage area and the altitude of the satellite. Applying cosines law to analyse the coverage geometry triangle at is shown as:

$$r^2 = Re^2 + d^2 - 2Red \cos(90 + \varepsilon_0) \tag{2.54}$$

where the radius (r) is equal to the total of satellite's altitude (H) and radius of the Earth (Re).

Substituting equations 2.52 and 2.54 we will finally get the slant range as a function of elevation angle as shown:

$$d_(\varepsilon_0) = Re \left[\sqrt{\left(\frac{H + Re}{Re} \right)^2 - \cos^2 \varepsilon_0} - \sin^2 \varepsilon_0 \right] \tag{2.55}$$

We can define the wideness of coverage belt as twice of the largest coverage radius, expressed as follows:

$$d_{(\varepsilon_0=0)} = d_{max} = Re\left[\sqrt{\left(\frac{H+Re}{Re}\right)^2} - 1\right] \tag{2.56}$$

$$D_{BELT} = 2d_{max} \tag{2.57}$$

Also as shown in the figure 2.5 and applying sinus theorem yields out:

$$\frac{\sin\alpha_0}{Re} = \frac{\sin(90+\varepsilon_0)}{Re+H} \tag{2.58}$$

$$\sin\alpha_0 = \frac{Re}{Re+H}\cos\varepsilon_0 \tag{2.59}$$

We know that the maximal coverage is achieved at e = 0, thus for each altitude H known we can easily define the coverage angle for maximal coverage as:

$$(\sin\alpha_0)_{MAXH} = \frac{Re}{Re+H} \tag{2.60}$$

Now, for different elevations we can calculate α_0 from equation 2.58 and find β_0 from equation 2.51. Based on the angle β_0, now we can evaluate the surface of the coverage area as:

$$S_{Coverage} = 2\pi Re^2(1-\cos\beta_0) \tag{2.61}$$

It is generally expressed as a fraction of the Earth's region to calculate the satellite's footprint, as follows:

$$Coverage(\%) = \frac{S_{Coverage}}{S_{Earth}} = \frac{2\pi Re^2(1-\cos\beta_0)}{4\pi Re^2} \tag{2.62}$$

$$Coverage(\%) = \frac{1}{2}(1-\cos\beta_0) \tag{2.63}$$

Chapter 3

Research Methodology

3.1 Orbit Design Process

The primary element of the design of the orbit and constellation is to document the mission criteria and goals used to identify the features of the orbit, trading-of with specific type of orbits, and the orbital properties of chosen orbit. These steps involve a lot of iterations and will be clarified further in this chapter.

3.1.1 Design Requirements and Objectives

The process of designing a constellation of satellites for any given space mission is quite complex and it requires multi-parametric optimization. The aim is to find a desired optimum constellation in order to reduce the complexity and cost of the system. Thus, the design objective of this research is to have an optimum constellation with the minimum number of satellites, and the best orbit configurations.

Besides the number of satellites, the best orbit configuration of the constellation can be obtained by specifying the trajectory of each satellite defined by six sets of Keplerian/classical orbital elements. Furthermore, the design requirements of this research are to provide navigation and surveillance continuous coverage service over the Indonesian region and to have the maximum age of the constellation under any orbital perturbations. However, the constellation deployment method and cost analysis will not be included in the orbit determination as it is outside the research scope of this research. Another limitation is that this research will estimate the age of the constellation without the presence of orbit maintenance.

According to [18] when designing an orbit design there is no absolute rules, table 3.1 will describe the basic method of giving a starting point and the steps for the orbit determination processes.

3.1.2 Establishing Orbit Types

The type of the orbit chosen will depend on the space mission segments based on the overall function of the satellite. This paper will consider the Earth-referenced orbit type as providing the necessary coverage of the Earth's surface. The main focus is to develop navigation-surveillance mission orbit by utilizing low Earth satellites.

TABLE 3.1: Summary of the orbit selection processes.

Number	Steps
1	Establishing the types of the orbit
2	Establishing basic requirements of the orbit-related mission
3	Assessing the Possibility of Specialized Orbits Utilization
4	Single Satellite and/or Constellation Evaluation Determination
5	Trades on Mission Orbit Design Parameters
6	Assessing Mission Launch and Disposal Options
7	Creating ΔV Budget

3.1.3 Establishing Orbit-Related Mission Requirements

The third step requires a trade-off for orbit-related requirements in order to meet the desired constraint on a mission. These requirements related to orbit may include several orbital limitations, such as the required altitude, the shape and orientation of the orbits for the constellation. All of the requirements will run the orbit design into different directions and characteristics.

The first parameter requirement that this research wants to address is the altitude of the satellite. The altitude affects the coverage swath width for surveillance mission and directly dependent on the strength of the antenna installed in the satellite. The second parameter is the inclination of the orbit. Since the area of interest of this research is the Indonesian region, in which it is located along the equator orbit, the inclination will start from a very small angle with not much variation.

The next parameter that needs to be considered is the perturbation. This research will consider perturbation under J_2 as the main perturbation, where it affects the lifetime of the orbit and the constellation as a whole. The expected lifetime of the orbit, as well as the options of the orbit design, will be further discussed in Chapter 4.

3.1.4 Assessing the Possibility of Specialized Orbits

When selecting the type of the orbit for the mission phase, this research needs to define first whether a specialized orbit applies or not. As explained in Chapter 2, specialized orbits are those with some of the unique characteristics. This research also will explore if certain specialized types orbits are suitable to establish the required constellation and are able to comply with the desired mission requirements.

The initial assumption is to utilize a combination between semi-synchronous orbit and an equatorial orbit to provide continuous coverage over the Indonesian region. The options will then need to be examined further considering the optimum number of satellites that need to be deployed.

3.1.5 Single Satellite and/or Constellation Decision

Obviously, utilizing only single satellite tends to give superiority in cost than the use of multiple satellites. However not all space mission can be implemented by simply using one satellite. A constellation, however as explained in Chapter2, may provide better coverage, higher reliability and survivability performance.

In order to comply with the mission requirements and objectives, budget limits, an orbit designer, we must consider the trade-of for a single satellite with large and complex instruments like the geosynchronous satellite against a constellation of smaller and simpler satellites like CubeSats in the LEO segmentation. Based on this fact, this research will also try to utilize small satellites, the microsatellites, as the main vehicle and has decided to construct possible constellation as the consequence of utilization of the small satellites.

3.1.6 Mission Orbit Design Parameters Trades

When developing an orbit constellation, common altitude and inclination for satellites deployed will generally be implemented since how the orbit decays will be strongly influenced by the two orbital elements; uniform decay among the satellites is more desirable in term of performance of the whole constellation. The variation of altitude in which affecting the coverage area will later be discussed in Chapter 4 in addition to the variation of the elevation angle of the antenna of each satellite. This method will create a broad variety of prospective attitudes and angles of elevation from which one or more options can be selected by this study.

3.1.7 Assessing Mission Launch and Disposal Options

While the launch stage contributes significantly to the mission expenses and will eventually restrict the quantity of mass that can be put in the orbit of any specified altitude, this study will suggest the launch vehicle alternatives and their price calculation as one of the study scope constraints.

Disposal of the spacecraft or the satellite has become important to any mission design. A good disposal plan that is going to the atmosphere must burn up or break up into harmless pieces. This because the potential debris and collision could harm other active spacecraft in the orbit. The benefit of Low Earth Orbit utilization is the easiness of throwing away any of out of service satellites in orbit directly into the atmosphere. By estimating the age of the constellation this research will also be able to have an evaluation of a disposal time plan that will be done in the future.

3.1.8 Create ΔV Budget

The ΔV budget is traditionally used by an orbit designer to account for the orbit change energy required. It will represent the total cost for each mission orbit scenario. Since a space mission is a sequence of distinct movements and transfers in orbits, such as, for instance, a satellite initially placed at the parking orbit before inserted into the target/nominal orbit or alternately it will go through a sequence of mission and then re-phases and lastly be transferred into a particular for mission extension or end-of-life mission. The total fuel budget is the summation of all ΔVs taken during the mission, started from orbit placing in transfer orbit, insertion into the nominal orbit and finally to the end-of-life of the mission. Lower total ΔVs is preferred, since it will translate into lower fuel budget, hence lower cost for the mission.

As stated previously, this research will not take into account the orbit deployment methods. Thus the overall ΔV budget will not be covered. Although the orbit will not be maintained, the age of constellation integrity will still be analysed by using a free orbit propagator.

Chapter 4

Results and Discussions

4.1 Required Coverage Area Calculations

As mentioned in the Chapter 1 the surveillance focus area of this research will cover the Indonesian airspace, thus the area of coverage needs to be calculated first. The Indonesian geographical location is located along the equatorial line with the coordinates at longitude in between 95.31644 and 140.71813 as well as at latitude in between -10.1718 and 5.88969.

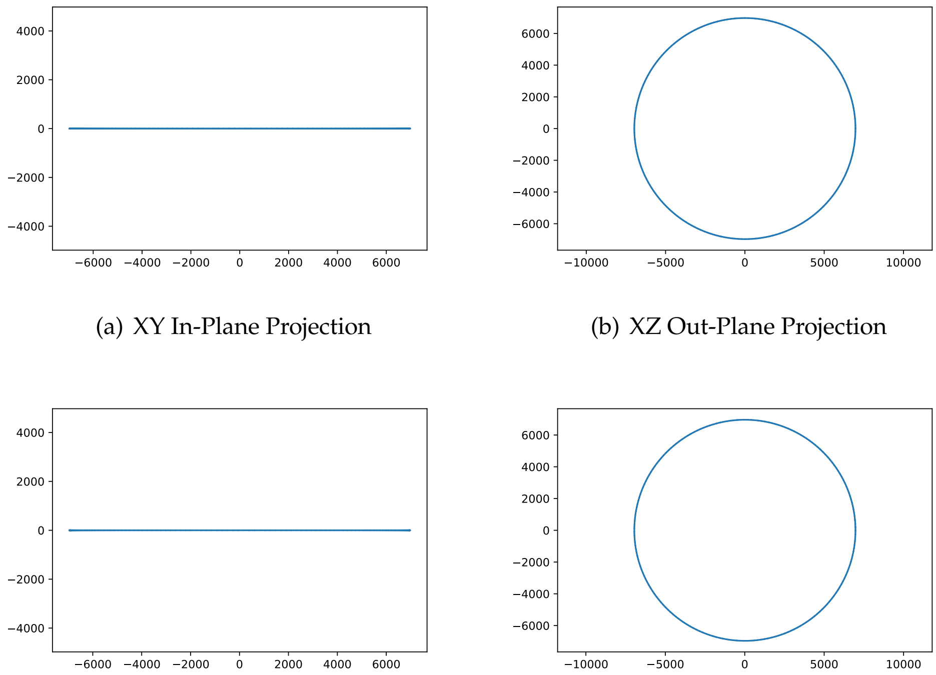
It is now possible to estimate the wideness and range of coverage that is needed to be covered as this research is aiming to have full and continuous coverage. The width of the longitude of the Indonesian region is around 45.4 degrees as for latitude wideness is around 16.1 degrees. At the equatorial region, for every longitude degree is equal to around 111.32 km while for every latitude degree is equal to around 110.57 km. Eventually, the wideness of the Indonesian region in kilometres is 5053.928 km in longitude and 1780.177 km in latitude.

4.2 Orbit Design Result

Reference orbit trajectory may be plotted after the Keplerian orbital elements are defined already. The initial assumption of the orbital elements are shown in the table 4.1. The trajectory then will be evaluated and tested under the effect of zonal perturbation from equation 2.6 The orbital elements will later be numerically integrated with fixed step size second order differential equations (ODE) method using the period of each orbit configurations. The number of the step samples of the orbit integration is set to be 1000 for every single orbital period. As can be seen from figure 4.1(a) until 4.4(d), after the J_2 secular perturbation term is introduced then the satellite orbit trajectory will get deviated from the orbit in the central gravitational field.

TABLE 4.1: Potential orbital elements variation.

Type of Orbit	$a(km)$	$i(^\circ)$	$\Omega(^\circ)$	e	$\omega(^\circ)$	$\theta(^\circ)$
Equatorial Orbit	6971	0	0	0	0	0
Near Polar Orbit	6971	85	118	0	90	0
Inclined Orbit	6971	20	118	0	90	0
Semi-synchronous Orbit	6971	98	118	0	90	0



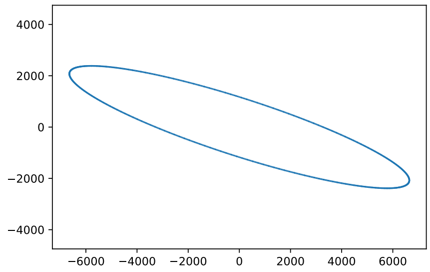
(a) XY In-Plane Projection

(b) XZ Out-Plane Projection

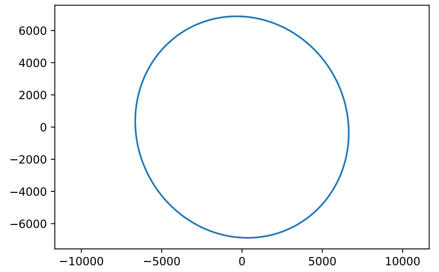
(c) XY In-Plane Projection under J_2

(d) XZ Out-Plane Projection under J_2

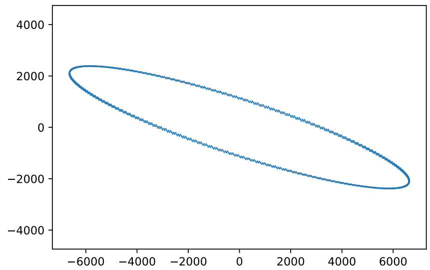
FIGURE 4.1: Equatorial Orbit Integration



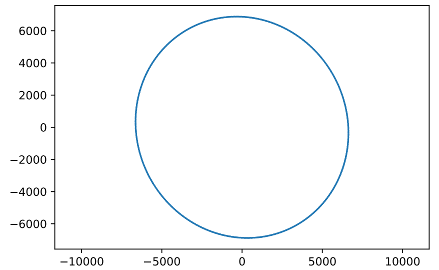
(a) XY In-Plane Projection



(b) XZ Out-Plane Projection

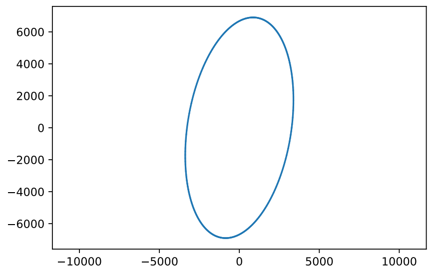


(c) XY In-Plane Projection under J_2

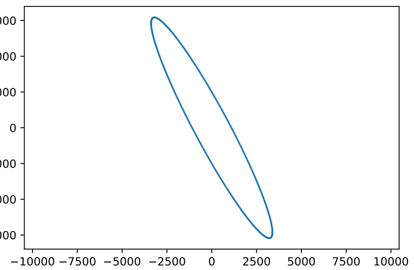


(d) XZ Out-Plane Projection under J_2

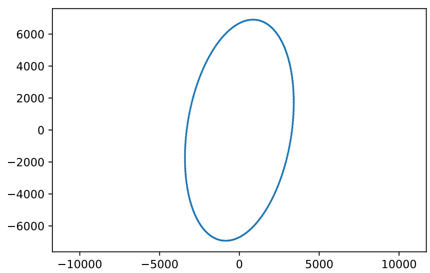
FIGURE 4.2: Inclined Orbit 20° Integration



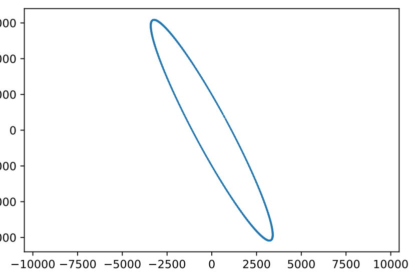
(a) XY In-Plane Projection



(b) XZ Out-Plane Projection

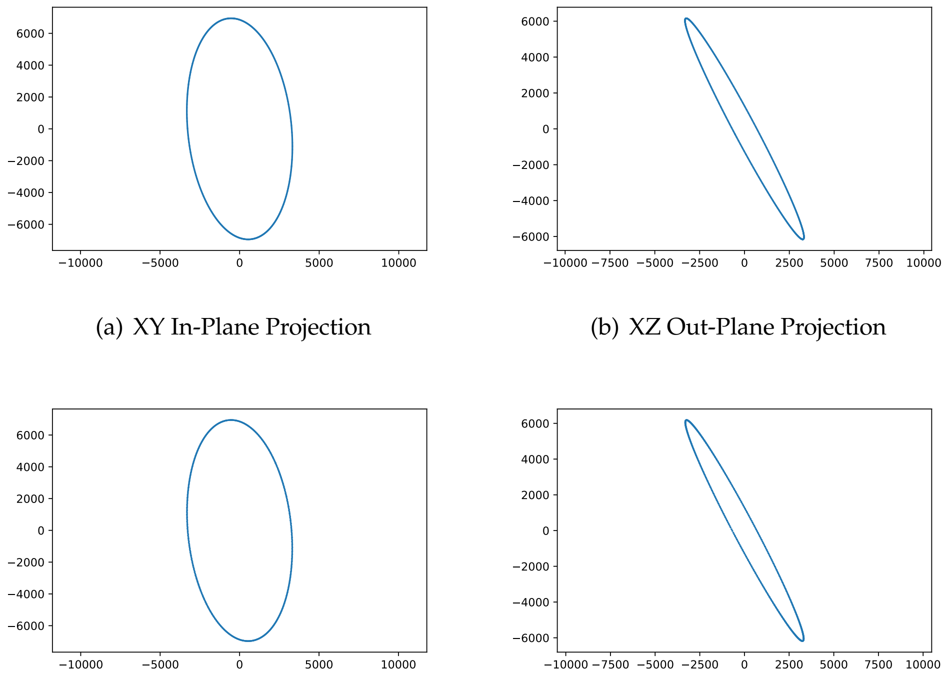


(c) XY In-Plane Projection under J_2



(d) XZ Out-Plane Projection under J_2

FIGURE 4.3: Sun-synchronous Orbit Integration



(a) XY In-Plane Projection (b) XZ Out-Plane Projection

(c) XY In-Plane Projection under J_2 (d) XZ Out-Plane Projection under J_2

FIGURE 4.4: Near polar Orbit Integration

In order to get the idea of the orbital parameter variations in detail, each individual orbital parameter will be plotted with respect of time in Keplerian orbit separately. The rate of change values of orbital elements evaluation of an orbit in different conditions can also be evaluated as can be seen in Table 4.2.

The equation 2.48 are then integrated to introduce the secular perturbation J_2 with the period of T = 5971 seconds for altitude 600 km with fixed step size of 1000 samples. After the secular perturbation J_2 is introduced and evaluated, the example variations of the orbital elements under the influence of J_2 perturbation can be seen in figure 4.5. This figure will show the regression of the ascending node and argument of perigee under the J_2 perturbation as the main interest.

TABLE 4.2: Rate of change in orbital elements under J_2 influence.

COE	**Equatorial**	**Inclined**	**Near Polar**	**Sun Sync**
$\dot{a}(km)$	0	0	0	0
$\dot{i}(^\circ)$	0	0	0	0
$\dot{\Omega}(^\circ)$	-1.475×10^{-6}	-1.386×10^{-6}	-1.285×10^{-7}	2.052×10^{-7}
$\dot{e}$	0	0	0	0
$\dot{\omega}(^\circ)$	2.949×10^{-6}	2.518×10^{-6}	-7.093×10^{-7}	-6.659×10^{-7}
$(\dot{M})$	1.086×10^{-3}	1.085×10^{-3}	1.084×10^{-3}	1.084×10^{-3}

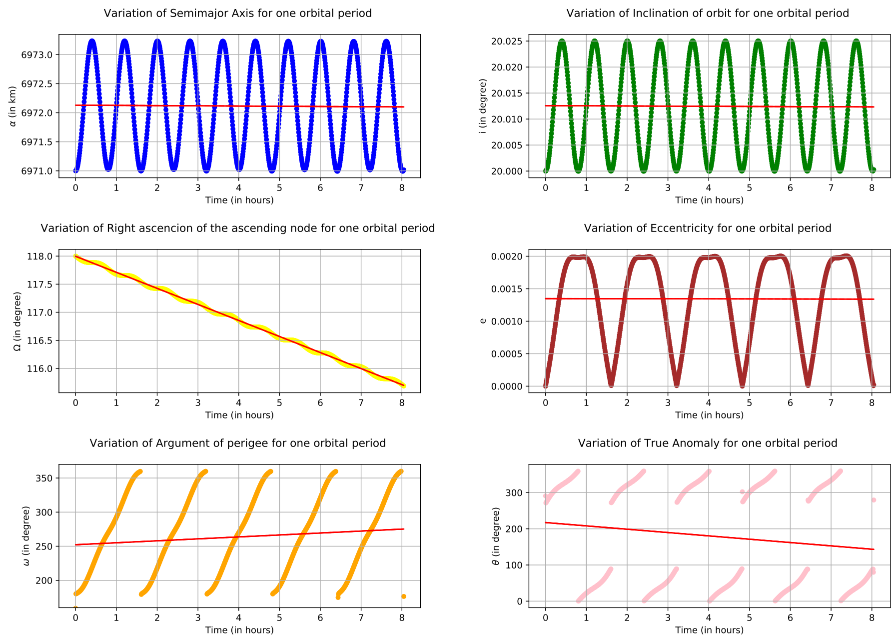


FIGURE 4.5: Orbital parameter variation example due to main harmonic J_2.

4.3 LEO Satellites Coverage Area Results

According to eq 2.51 and 2.55 , it is obvious that the coverage belt of the satellite depends strongly on elevation angle. The largest possible coverage belt can be obtained when the elevation angle is equal to $0°$. Since this research is utilizing Low Earth Orbit configuration, the coverage belt variations simulation will range the altitudes from 600 km up to 1000 km. These altitude parameters will then be simulated and calculated with variations of elevations angle between 0 and 8 degrees with steps of $2°$ following the possible antenna configuration of the satellites. This relationship between altitude and elevation angle will define the variations of coverage belt of the satellite.

TABLE 4.3: The Coverage belt wideness variations.

Altitude (km)	**600**	**700**	**800**	**900**	**1000**
Elevation (ε)	**D (km)**	**D (km)**	**D (km)**	**D (km)**	**D (km)**
0°	5660	6138	6586	7010	7416
2°	5234	5709	6156	6581	6986
4°	4841	5313	5756	6178	6581
6°	4483	4948	5387	5804	6203
8°	4158	4614	5046	5458	5852

TABLE 4.4: The Coverage belt wideness on longitude angle variations.

Altitude (km)	**600**	**700**	**800**	**900**	**1000**
Elevation (ε)	**Lon (°)**	**Lon (°)**	**Lon (°)**	**Lon (°)**	**Lon (°)**
0°	47.87	51.39	54.62	57.59	60.36
2°	44.02	47.54	50.75	53.72	56.48
4°	40.49	43.97	47.16	50.10	52.84
6°	37.26	40.68	43.82	46.72	49.43
8°	34.32	37.66	40.74	43.59	46.25

By knowing the range of satellite altitude and elevation, it is then possible to evaluate the range value of the nadir angle α. The elevation and nadir angle values will be used to evaluate the central angle β. This central angle is needed in order to find the coverage area of the satellites and by using the equation 2.63, it is possible to know the percentage coverage area of the satellite as a fraction of the area of the Earth.

TABLE 4.5: The Coverage belt area percentage variations.

Altitude (km)	**600**	**700**	**800**	**900**	**1000**
Elevation (ε)	**(%)**	**(%)**	**(%)**	**(%)**	**(%)**
0°	4.30	4.94	5.57	6.18	6.78
2°	3.64	4.24	4.82	5.39	5.95
4°	3.09	3.64	4.17	4.70	5.22
6°	2.62	3.12	3.61	4.10	4.58
8°	2.23	2.68	3.13	3.57	4.02

4.4 Ground Track Visualization

The projected path of a satellite to the Earth's surface along the vector position of inertial geodetic frame is known as ground track. This track is basically consists of traces of the satellite along the surface of the Earth, an imaginary line between the satellite and the center of the Earth. The line consists of a set of satellite location points at which the satellite will fly through or cross the zenith, in the frame of reference of a ground observer.

The form of the ground track of any satellites will vary depending on the Keplerian elements, that describe the size and orientation of the orbit, of the satellite. Figure 4.6 will show the example of the equatorial ground track orbit. This figure will present the movement prediction of the satellite along the equator line. The movement for the inclined, sun-synchronous and near polar orbits will be shown in figure 4.7, 4.8, 4.9 respectively.

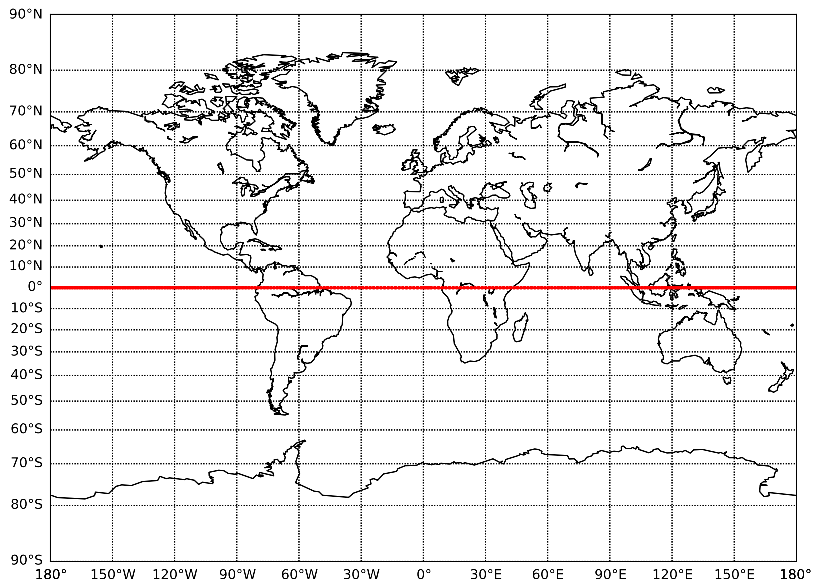


FIGURE 4.6: Equatorial orbit ground track.

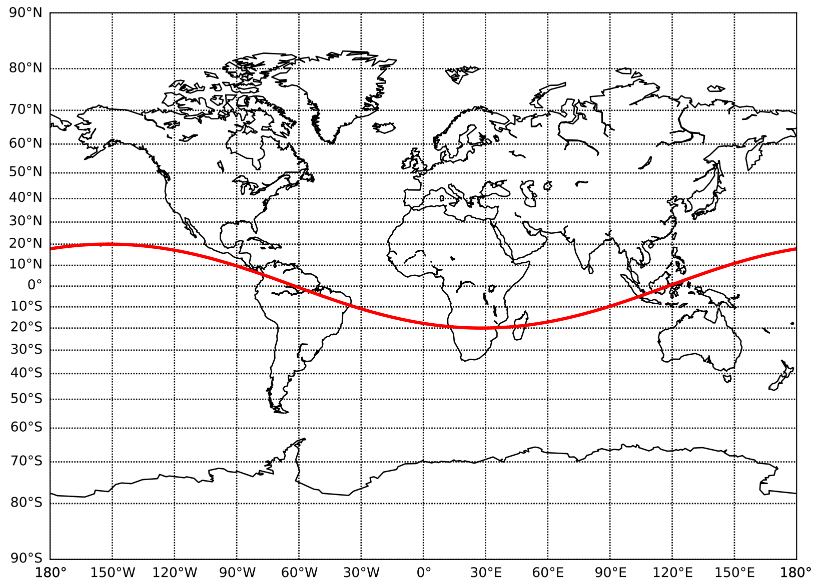


FIGURE 4.7: Inclined orbit ground track.

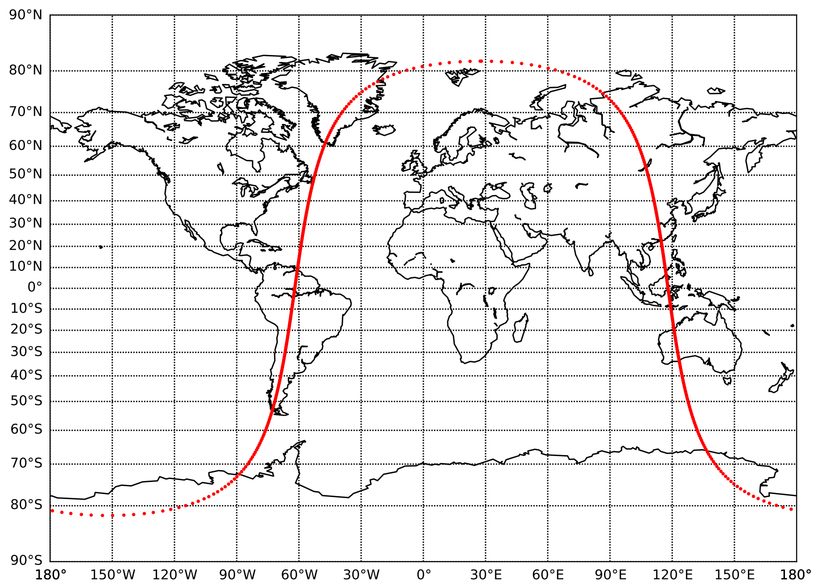


FIGURE 4.8: Sun-synchronous orbit ground track.

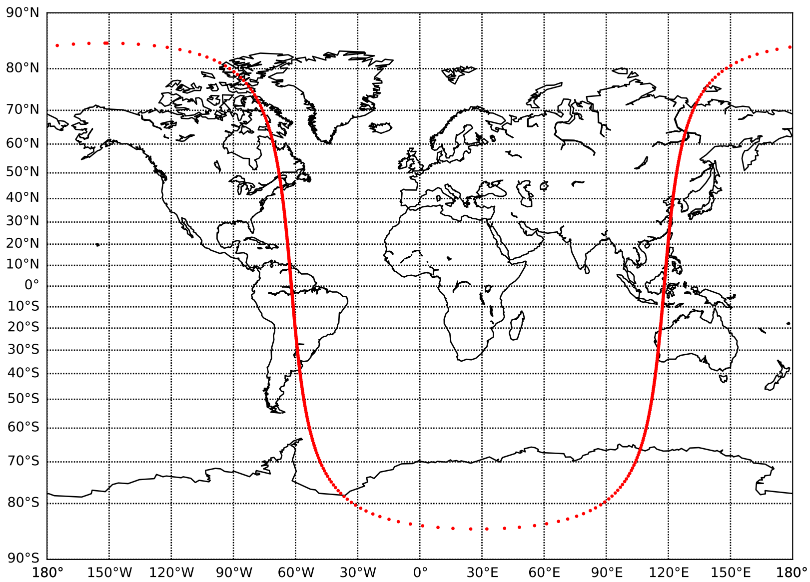


FIGURE 4.9: Near polar orbit ground track.

4.5 Constellation Variations Evaluation

After defining the possible orbits that can be implemented in order to have full and continuous coverage in Indonesia, the variation of the constellation design can now be analysed. This research will consider the possibility of utilizing single and double plane constellation configuration. Table 4.6 will show the possible constellation configuration for single plane configuration with different altitude that provides a different number of satellites needed to have the full and continuous coverage over the Indonesian region as planned in this research.

TABLE 4.6: Single plane constellation configuration with 5° elevation angle.

Orbit Selection	Altitude	Number of Satellites
Equatorial, Circular	600	9.27
	700	8.51
	800	7.92
	900	7.44
	1000	7.05

The number of satellites given in the table 4.6 is coming from the calculation of the possible LEO satellites coverage area given in the table 4.3. The number of satellites results will then assume that the elevation angle between the antenna of the satellite and the ground station is 5°. As the widest Indonesian area that needs to be covered is around 5053.93 km, using a 2° maximum elevation angle is enough for a satellite to cover the whole Indonesian region with one single visiting time using the lowest altitude configuration.

The reason behind the equatorial orbit selection is because of the unique location of Indonesia. As stated previously in the early section of this chapter, the geographical location of Indonesia is located along the equator line of the Earth. This unique characteristic makes equatorial orbit as the best option to have the possibility of covering the Indonesian region at any time. Circular orbit configuration selection is also based on the fact that there will be very slight surface condition difference along the equator. Having the circular orbit is then beneficial to have the constant coverage area of each satellite in the orbit plane.

TABLE 4.7: Common ADS-B payload satellite configuration.

Name	Altitude (km)	Elevation Angle (°)
SkyFox CubeSat	600	5
GOMX-4 NanoCom AX100	700	5
HyperScout	500	5
Iridium NEXT	780	12
Proba-V	820	5

For multiple planes constellation configuration, it is possible to evaluate the coverage of the constellation of the Indonesian region as time progresses. It means that the total of covered region of the Indonesian airspace can be modelled as the sum of coverage percentage with respect to time of all the constellation's member. The way to evaluate the total coverage percentage of the satellites in the Indonesian region will start from computing the position of the satellites for each plane using non-Keplerian orbit propagation. Assuming the shape of the access area is circular and knowing all basic requirements to evaluate the access area, for example, the altitude and elevation angle, it is now possible to evaluate the access area for each satellite along its plane at any specific time.

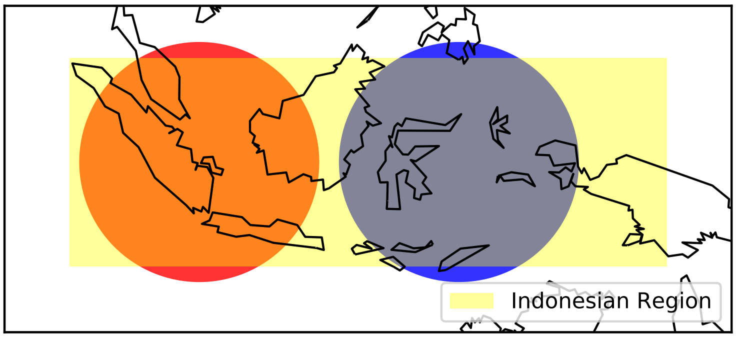
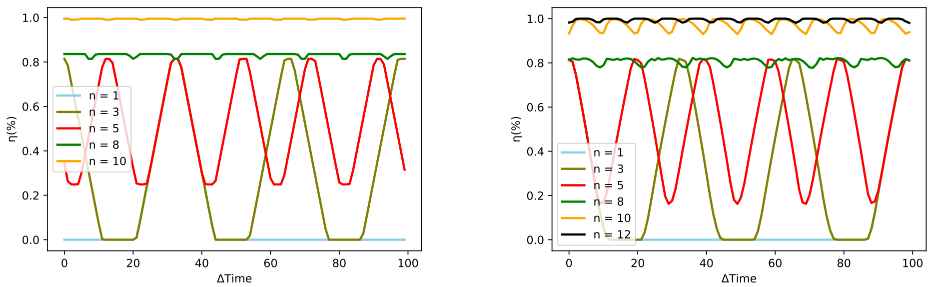


FIGURE 4.10: Access area analysis of the Indonesian region provided by the satellite constellation coverage (red and blue circle).

This access area of each satellite will then be projected into a planar area which represents the Indonesian region. The intersection between the access area polygon and the Indonesian region will then be evaluated as a percentage of possible satellite coverage area. The coverage percentage (η) for any constellation configuration is the ratio of the constellation coverage for one orbital period and the total area of the Indonesian region. The illustration is shown in figure 4.10. This method will also be useful in order to define the age of constellation for any planes.

Propagating the satellite position for any period of time in order to have the detail position of the satellite will be useful to define the street coverage of the constellation. The figures 4.11(a), 4.11(b) and 4.13 will give the best representation on how the coverage percentage performance information can be plotted with the variation of the number of satellites needed for each constellation configuration. Over one orbital period, the variation of the number of the satellites for each constellation will give a different value of coverage percentage.



(a) Equatorial orbit constellation

(b) Inclined orbit constellation

FIGURE 4.11: Single plane constellation coverage percentage for n satellites configuration.

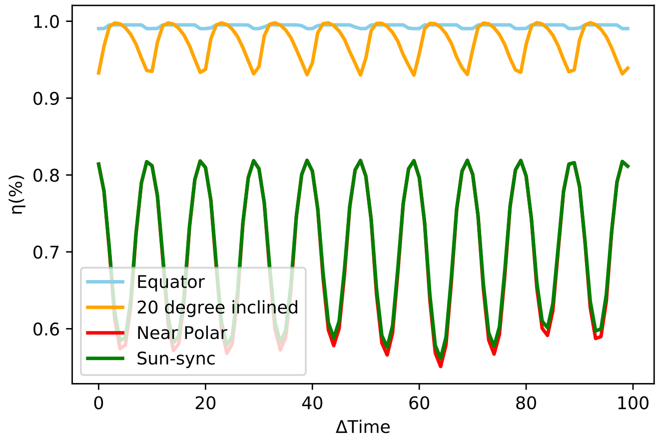


FIGURE 4.12: Coverage percentage for different types of orbit provided by 10 satellites configuration.

For a single plane configuration, this research tried to evaluate each orbit configuration to see its coverage percentage performance starting with the equatorial orbit as shown in figure 4.11(a), while another evaluation is for the inclined orbit as shown in the figure 4.11(b). Both results show a unique characteristic of the coverage percentage performance for each orbital type. This research assumption of having more than 90% coverage at any time is achieved by both configurations.

In total, 10 satellites are needed to have the coverage percentage performance that is more than 90% for both configurations. But, it is clear to see that the equatorial constellation configuration provides the highest coverage performance that gives the lowest value of 99% coverage of the Indonesian region. This is also in agreement with the numerical analysis result of total satellite needed for equatorial orbit with an altitude of 600 km and 5° of elevation as shown in table 4.6.It is stated that the total satellite needed is more than 9 satellites.

Since this research already has a benchmark number of satellites needed, this research is now able to compare the coverage performance provided by 10 satellites in total for a single constellation for different orbit configuration as shown in figure 4.12. From the figure, it is now safe to say that equatorial constellation configuration still provide the best coverage percentage performance among to the others.

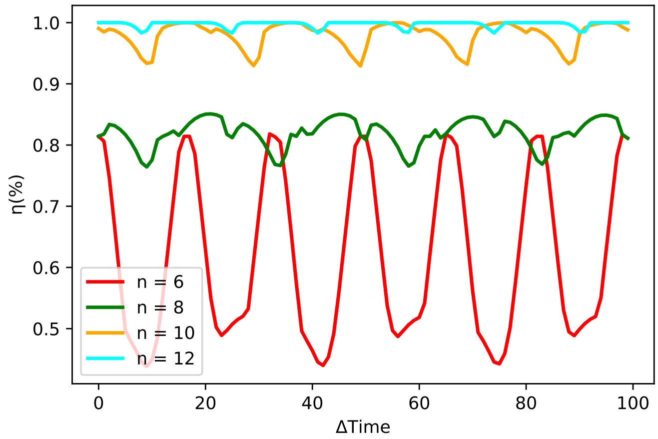


FIGURE 4.13: Multiple plane constellation configuration coverage percentage for n satellites using equatorial and 20° inclined orbit.

On the other hand, for the multiple planes constellation configuration, in this case the combination between the equatorial and inclined orbit configurations, at least 10 satellites is also needed to have more than 90% coverage at any time. But in order to have the same coverage percentage performance as the single plane equatorial constellation configuration, more or less 12 satellites are needed for this configuration.

4.6 Age of Constellation Estimation

The purpose of evaluating the age of constellation for any design options is to have the correct estimation on how long the constellation will last without any help from orbit maintenance mechanism. The approach will start with determining the rate of change of the altitude, the right ascension of ascending nodes ($\dot{\Omega}$) and the argument of perigee ($\dot{\omega}$) caused by J_2 perturbation for the orbit type chosen with the variation of orbit altitude and period.

The altitude will vary from 600 km to 1000 km with step variation of 100 km. For a complete calculation, the perturbation under atmospheric drag will also be included in the calculation. This perturbation will affect the altitude of the orbit with the variation of orbit altitude and period as well. The dimension of the satellite that will be considered in order to calculate the atmospheric drag is as follows:

TABLE 4.8: Satellite basic configuration

Type of Satellite	Microsatellite
Mass (kg)	50-100
Dimension (cm)	$50 \times 50 \times 50$
Antenna Elevation (°)	0-5

The results for single constellation with circular equatorial orbit option are as follows:

TABLE 4.9: Rate of altitude, right ascension of ascending node, and argument of perigee caused by the combination of J_2 and atmospheric drag perturbations.

Altitude	$\dot{H}(km/day)$	$\dot{\Omega}(°/sec)$	$\dot{\omega}(°/sec)$
600	-2.170×10^{-3}	-1.47×10^{-6}	2.95×10^{-6}
700	-1.928×10^{-3}	-1.40×10^{-6}	2.81×10^{-6}
800	-1.731×10^{-3}	-1.34×10^{-6}	2.67×10^{-6}
900	-1.580×10^{-3}	-1.27×10^{-6}	2.54×10^{-6}
1000	-1.307×10^{-3}	-1.21×10^{-6}	2.43×10^{-6}

After evaluating various configuration for the feasible constellation options under all of the research assumptions, this research is now able to conclude that the single plane equatorial constellation as the most feasible option. It requires 10 satellites in total for 600 km altitude to have the full-time coverage of the Indonesian region. By defining the specific constellation configuration, estimating the age of constellation will be the next important step be done.

TABLE 4.10: Altitude decrease on 600 km altitude configuration orbit.

Time (year)	**ΔAltitude (km)**	**Possible coverage radius (km)**
0.5	39.5	2232.12
1.0	79	2132.31
1.5	119	2027.92
2.0	158	1922.54
2.5	198	1810.24
3.0	237	1696.06

The first step is to propagate the satellite position to several time periods under the influence of the designated perturbations. The numerical analysis result in the table 4.10, for 600 km altitude configuration, the life expectancy of the constellation will be shown up to 3 years. The figure 4.14 will have a good visualization of what will happen with the coverage percentage of the constellation. The equatorial configuration utilizing 10 satellites in the constellation will be used as the main focus of the evaluation. The time variation will start from 6 months up to 3 years where the satellite is expected to have less than desired coverage percentage. The result shows conformity with the numerical analysis as the coverage percentage decreased significantly following the altitude reduction.

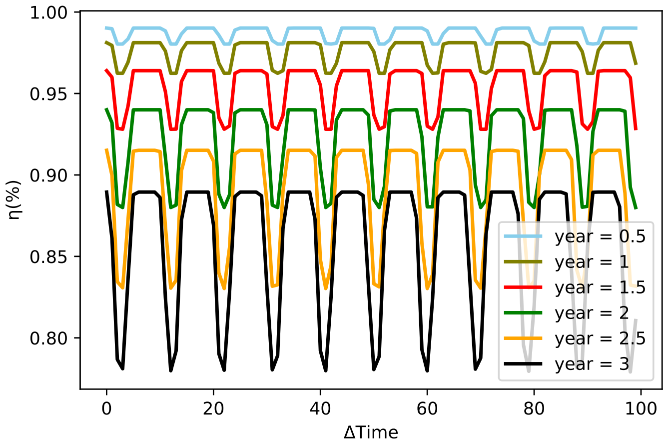


FIGURE 4.14: Age of constellation service visualization for 10 satellites equatorial configuration.

Following the result that is shown in figure 4.14, the age of constellation service can now be determined. It is when the constellation can no more cover the Indonesian region with 90% of coverage percentage minima. By the visualization on the line plot, the constellation will be on service for up to 2 years, exactly when the constellation will start to have less than 90% of coverage percentage.

Chapter 5

Conclusions and Recommendations

5.1 Conclusions

In conclusion, considering the unique location of the Indonesian region which located along the equator line, the equatorial orbit type offers the best coverage performance with the less number of satellites needed compared to other orbital types. Circular orbit configuration for the equatorial orbit is used as it provides the best constant coverage area utilizing the slight Earth surface bulge differences along the equator.

Combining the best orbital type and shape as found in this research, the current best constellation that is plausible to be implemented is the single plane equatorial constellation configuration. This configuration is able to provide more than 90% of the coverage percentage as required in this research and utilizing the lowest number of satellites needed with 10 satellites in total. The lowest coverage number that can be provided by this configuration is more or less 99% of the coverage percentage.

Under the influence of zonal gravitational J_2 and atmospheric drag perturbations, the constellation configuration chosen in this research will have more or less 2 years of service age. This 2 years service period is the limit of the constellation to always provide more than 90% of the coverage percentage without having any method of orbit maintenance.

5.2 Remarks for the Future Work

For future work and development, more variations of the inclination angle for the single plane constellation configuration could be done to see the performance differences that can be provided by each inclination configuration. By this evaluation, more multiple plane configurations can be examined further in order to see the possibility of having less number of satellites with at least the same coverage performance as single plane equatorial configuration constellation.

Cost analysis, as well as the deployment method, could be included in the future research to have a comprehensive analysis and a good comparison with the cost of the ground-based system. By the complete cost analysis and calculation, the best utilization choice between the ground and satellite-based system for air traffic monitoring and monitoring system can be determined better.

Appendix A

Function library

```
from scipy.integrate import odeint
import numpy as np
import math

# G_const for gravitational constant, ME for mass of earth,
# RE for radius of earth
G_const = 6.67408 * 10**(-11)
ME = 5.972364730419773 * 10**24
RE = 6378100 * 10**(-3)
# Declaring value of miu
miu = (G_const * ME) * 10**(-9)
# Declaring teh value of perturbation J2
J2 = 1.082629 * 10**(-3)

# declaring unit vector
i_unit = np.array([1, 0, 0])
j_unit = np.array([0, 1, 0])
k_unit = np.array([0, 0, 1])

# defining function sv2coe(r_vec, v_vec) for transforming state vector
# to classical orbital elements
# in the following definition of function sv2coe, sv is a state vector
# and coe is classical orbital elements
# the input element of the function sv2coe that need to be defined
# consists of:
#   r_vec is position vector of the satellite
#   v_vec is velocity vector of the satellite
#   all the input vector should be in 3x1 matrix
#
# the possible output of this function are as follows:
#   a = the magnitude of semimajor axis (in km)
#   i = the magnitude of inclination (in degree)
```

```
#   omega_capt = the magnitude of right ascencion of the ascending
#  node (in degree)
#   e = the magnitude of eccentricity (no unit)
#   omega_case = the magnitude of argument of perigee (in degree)
#   theta = the magnitude of true anomaly (in degree)
#   h = the magnitude of specific angular momentum (in km^2/s)
#   T = the magnitude of period the orbit (in hour)
# all the output will be in form of output array
# The output of the function needs to be recalled as an array function of
# [lists of desired output] = sv2coe(position vetor, velocity vector)


def sv2coe(r_vec, v_vec):
    # Calculating distance from r_vec
    r = np.sqrt(np.dot(r_vec, r_vec))
    # Calculating speed from v_vec
    v = np.sqrt(np.dot(v_vec, v_vec))
    # Calculating radial velocity
    v_rad = (np.dot(r_vec, v_vec)) / r
    # if v_rad > 0:
    #    print("Satellite is flying away from perigee")
    # else:
    #    print("Satellite is flying towards perigee")
    # Calculating specific angular momentum
    h_vec = np.cross(r_vec, v_vec)
    # Calculating magnitude of specific angular momentum
    h = np.sqrt(np.dot(h_vec, h_vec))  # first orbital element
    # Calculating inclination
    i = np.arccos(h_vec[2] / h)
    i = i * 180 / np.pi  # second orbital element
    # if i <= 90:
    #    print("The orbit is retrograde")
    # elif 90 < i < 180:
    #    print("The orbit is posigrade")
    # else:
    #    print("The orbit is greater than 180 and in quadrant ambiguity")
    # Calculating vector node line
    N_vec = np.cross(k_unit, h_vec)
    # Calculating magnitude of vector node line
    N = np.sqrt(np.dot(N_vec, N_vec))
    # Calculating right ascencion of the ascending node
    if N_vec[1] >= 0:
```

```
    omega_capt = np.arccos(N_vec[0] / N)
else:
    omega_capt = 2 * np.pi - np.arccos(N_vec[0] / N)
omega_capt = omega_capt * 180 / np.pi  # third orbital element
# Calculating eccentricity vector
e_vec = (1 / miu) * (np.dot((v**2 - (miu / r)),
                            r_vec) - (np.dot((r * v_rad), v_vec)))
# Calculating magnitude of eccentricity
e = np.sqrt(np.dot(e_vec, e_vec))  # fourth orbital element
# Calculating argument of perigee
if e_vec[2] >= 0:
    omega_case = np.arccos((np.dot(N_vec, e_vec) / (N * e)))
else:
    omega_case = 2 * np.pi - np.arccos((np.dot(N_vec, e_vec) / (N * e)))
omega_case = omega_case * 180 / np.pi  # fifth orbital element
# if np.dot(N_vec, e_vec) > 0:
#    print("Argument of perigee is in first or fourth quadrant")
# else:
#    print("Argument of perigee is in second or third quadrant")
# Calculating eccentric anomaly
# EA = np.arctan2(np.sqrt((a * (1 - e**2)) / miu) * np.dot(r_vec, v_vec),
# (a * (1 - e**2)) - r)
# Calculating true anomaly
if v_rad >= 0:
    theta = np.arccos((np.dot(e_vec, r_vec) / (e * r)))
else:
    theta = 2 * np.pi - np.arccos((np.dot(e_vec, r_vec) / (e * r)))
theta = theta * 180 / np.pi  # sixth orbital element
# if np.dot(e_vec, r_vec) > 0:
#    print("True anomaly is in first or fourth quadrant")
# else:
#    print("True anomaly of perigee is in second or third quadrant")
# Calculating mean anomaly
#MA = EA - e * np.sin(EA)
#MA = MA  *180 / np.pi
# Calculationg radius of perigee and radius of apogee
rp = (h**2 / miu) * (1 / (1 + e * np.cos(0)))
ra = (h**2 / miu) * (1 / (1 + e * np.cos(np.pi)))
# Calculating semi major axis
a = 0.5 * (rp + ra)
# Calculating mean motion
n = np.sqrt(miu / (a**3))
```

```
    # Calculating period of an orbit
    T = (2 * np.pi) / n
    T = T / 3600
    # Calculating semi-latus rectum
    p = a * (1 - e**2)
    return [a, i, omega_capt, e, omega_case, theta, h, T, n, p, r]

# defining function coe2sv(h, i, omega_capt, e, omega_case, theta)
# for transforming state vector to classical orbital elements
# in the following definition of function sv2coe, sv is a state vector
# and coe is classical orbital elements
# the input element of the function sv2coe that need to be defined
# consists of:
#   h = the magnitude of specific angular momentum (in km^2/s)
#   i = the magnitude of inclination (in degree)
#   omega_capt = the magnitude of right ascencion of the ascending node
# (in degree)
#   e = the magnitude of eccentricity (no unit)
#   omega_case = the magnitude of argument of perigee (in degree)
#   theta = the magnitude of true anomaly (in degree)
#   all the inputs should be in each correct unit
#
# the possible output of this function are as follows:
#   r_peri = the position vector in perifocal coordinates (in km)
#   v_peri = the velocity vector in perifocal coordinates (in km/s)
#   Q_geo = the matrix Q_geo transformation from perifocal to geocentric
# equatorial coordinates (no unit)
#   r_geo = the position vector in geocentric frame (in km)
#   v_geo = the velocity vector in geocentric frame (in km/s)
# all the output will be in form of output array
# The output of the function needs to be recalled as an array function of
# [lists of desired output] = coe2sv(h, i, omega_capt, e, omega_case, theta)


def coe2sv(a, i, omega_capt, e, omega_case, theta):
    h = np.sqrt(a * miu * (1 - e**2))
    # Calculating position vector in perifocal coordinates
    r_peri = (h**2 / miu) * (1 / (1 + e * np.cos(theta))) * \
        np.array([np.cos(theta), np.sin(theta), 0])
    # Calculating velocity vector in perifocal coordinates
    v_peri = (miu / h) * np.array([-np.sin(theta), e + np.cos(theta), 0])
    # Calculating matrix Q_geo transformation from perifocal to geocentric
```

```
    # equatorial coordinates
    A = (np.cos(omega_capt) * np.cos(omega_case) -
         np.sin(omega_capt) * np.sin(omega_case) * np.cos(i))
    B = (-np.cos(omega_capt) * np.sin(omega_case) -
         np.sin(omega_capt) * np.cos(i) * np.cos(omega_case))
    C = (np.sin(omega_capt) * np.sin(i))
    D = (np.sin(omega_capt) * np.cos(omega_case) +
         np.cos(omega_capt) * np.cos(i) * np.sin(omega_case))
    E = (-np.sin(omega_capt) * np.sin(omega_case) +
         np.cos(omega_capt) * np.cos(i) * np.cos(omega_case))
    F = (np.cos(omega_capt) * np.sin(i))
    G = (np.sin(i) * np.sin(omega_case))
    H = (np.sin(i) * np.cos(omega_case))
    I = (np.cos(i))
    Q_geo = np.array([[A, B, C], [D, E, F], [G, H, I]])
    # Transforming perifocal position and velocity to geocentric frame
    r_geo = np.dot(Q_geo, r_peri)
    v_geo = np.dot(Q_geo, v_peri)
    return [r_peri, v_peri, Q_geo, r_geo, v_geo]

# defining function sphericalCoor(r_vec) for finding the longitude
# and latitude
# coordinate positions of the satellite
# the input element of the function sphericalCoor that need to be
# defined consists of:
#   r_vec = vector position of the satellite (in km)
#
# the possible output of this function are as follows:
#   r = size of the radius of the satellite posisiton (in km)
#   longitude = the longitude coordinate of the satellite (in degree)
#   latitude = the latitude coordinate of the satellite (in degree)
# all the output will be in form of output array
# The output of the function needs to be recalled as an array function of
# [lists of desired output] = sphericalCoor(r_vec)


def sphericalCoor(r_vec):
    r = np.sqrt(np.dot(r_vec, r_vec))
    latitude = np.arcsin(r_vec[2] / r)
    latitude *= 180 / np.pi
    longitude = np.arctan2(r_vec[1], r_vec[0])
    longitude *= 180 / np.pi
```

```
    return [r, longitude, latitude]


# In the following definition of orbit propagator
# derivative function F,
# s is a state vector whose components represent the following:
#   s[0] = Horizontal or x position
#   s[1] = Vertical or y position
#   s[2] = Lateral or z position
#   s[3] = Horizontal velocity
#   s[4] = Vertical velocity
#   s[5] = Lateral velocity
# while t is the period of the orbit (in second) that needs
# to be defined more
# further; t = np.linspace(start,end,interval)
#
# The input for the defined derivative function consists of:
#   x0 = Initial position in x axis
#   y0 = Initial position in y axis
#   z0 = Initial position in z axis
#   vx0 = Initial velocity in x axis
#   vy0 = Initial velocity in y axis
#   vz0 = Initial velocity in z axis
# The solution of the derivative function will be using odeint() function,
# where will be declared as:
# solution = odeint(F,[x0,y0,z0,vx0,vy0,vz0],t)

def F(s, t):
    a = -miu * s[0] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2)
    b = -miu * s[1] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2)
    c = -miu * s[2] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2)
    return [s[3], s[4], s[5], a, b, c]

# In the following definition of orbit propagator derivative
 # function F1 or
# Perturbation Accelerations due to J2,
# s is a state vector whose components represent the following:
#   s[0] = Horizontal or x position
#   s[1] = Vertical or y position
#   s[2] = Lateral or z position
#   s[3] = Horizontal velocity
#   s[4] = Vertical velocity
```

```
#   s[5] = Lateral velocity
# while t is the period of the orbit (in second) that needs
 # to be defined more
# further; t = np.linspace(start,end,interval)
#
# The input for the defined derivative function consists of:
#   x0 = Initial position in x axis
#   y0 = Initial position in y axis
#   z0 = Initial position in z axis
#   vx0 = Initial velocity in x axis
#   vy0 = Initial velocity in y axis
#   vz0 = Initial velocity in z axis
# The solution of the derivative function will be using
# odeint() function,
# where will be declared as:
# solution = odeint(F,[x0,y0,z0,vx0,vy0,vz0],t)
#
# Plotting the orbit from orbit propagator
# 1. Define the period (t) of the orbit in second
# 2. Declare the solution function using odeint() function
# 3. Declare a void array, e.g. results = []
# 4. Example of getting the array of integral results:
#    for data in range(0,1000):
#      r_test = np.squeeze(solution[data:data+1,0:3])
#      v_test = np.squeeze(solution[data:data+1,3:6])
#      t1 = np.linspace(0,8200,data+1)
#      results.append(sv2coe(r_test, v_test))
#    results = np.array(results)
# 5. Example of command plotting:
#    import matplotlib.pyplot as plt
#    x1 = results[:,0]
#    y1 = t1
#    plt.plot(y1,x1, '.')
#    plt.axis('equal')
#    plt.show()


def F1(s, t):
    Pt = (3 / 2) * (miu * J2 * (RE**2))
    r = (s[0]**2 + s[1]**2 + s[2]**2)**(1 / 2)
    a = -miu * s[0] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2) - \
        Pt * ((1 - 5 * (s[2]**2 / r**2)) * (s[0] / r**5))
```

```
    b = -miu * s[1] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2) - \
        Pt * ((1 - 5 * (s[2]**2 / r**2)) * (s[1] / r**5))
    c = -miu * s[2] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2) - \
        Pt * ((3 - 5 * (s[2]**2 / r**2)) * (s[2] / r**5))
    return [s[3], s[4], s[5], a, b, c]

# In the following definition of orbit propagator derivative function F2
# including the Perturbation Accelerations due to J2 and Atmoshperic Drag,
# The complete explanation will be the same as the function F1


def F2(s, t):
    Pt = (3 / 2) * (miu * J2 * (RE**2))
    v = (s[3]**2 + s[4]**2 + s[5]**2)
    Fd = -0.5 * 1.454 * 2.7 * 0.15 * v * 10**(-13)
    r = (s[0]**2 + s[1]**2 + s[2]**2)**(1 / 2)
    a = -miu * s[0] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2) - \
        Pt * ((1 - 5 * (s[2]**2 / r**2)) * (s[0] / r**5)) + Fd
    b = -miu * s[1] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2) - \
        Pt * ((1 - 5 * (s[2]**2 / r**2)) * (s[1] / r**5)) + Fd
    c = -miu * s[2] / (s[0]**2 + s[1]**2 + s[2]**2)**(3 / 2) - \
        Pt * ((3 - 5 * (s[2]**2 / r**2)) * (s[2] / r**5))
    return [s[3], s[4], s[5], a, b, c]

# The rate() function is defining the value of the mean classical orbital
# elements rate of change,
# the input components of the function is described as following:
# i = inclination of the orbit (in degree)
# e = eccentricity of the orbit (no unit)
# n = mean motion of the orbit (no unit)
# p = semi-latus rectum/orbit parameter (no unit)
#
# The output components of the function comprises of:
# semiMajRate = rate of semi major axis (in km/second)
# eccenRate = rate of eccentricity (no unit)
# incliRate = rate of inclination (in degree/second)
# omgRate = rate of right ascencion of the ascending node (in degree/second)
# omgCaseRate = rate of argument of perigee (in degree/second)
# meanAnomaly = rate of mean anomaly (in degree/second)


def rate(i, e, n, p):
```

```
    rateFactor = n * J2 * ((RE / p)**2)
    semiMajRate = 0
    eccenRate = 0
    incliRate = 0
    omgRate = -(3 / 2) * (rateFactor * np.cos(i * np.pi / 180))
    omgCaseRate = (3 / 4) * rateFactor * (5 * (np.cos(i * np.pi / 180)**2) - 1)
    meanAnomaly = (n + (3 / 4) * np.sqrt(1 - e**2) *
                   rateFactor * (3 * (np.cos(i * np.pi / 180)**2) - 1))
    return semiMajRate, eccenRate, incliRate, omgRate, omgCaseRate, meanAnomaly

# defining function coverageBelt(altitude, elevAngle) to find the radius of
# satellite's coverage belt wideness
# the input element of the following function consists of:
#    altitude = the altitude of the satellite above sea level (in km)
#    elevAngle = it is the elevation angle relative from the ground station
# horizon plane to the satellite (in degree)
#    all the inputs should be in each correct unit
#
# the possible output of this function is as follows:
# slantRange = the distance from ground station to the satellite, it also
# defines the radius of satellite coverage belt wideness (in km)


def coverageBelt(altitude, elevAngle):
    slantRange = RE * (np.sqrt((((altitude + RE) / RE)**2) -
                               (np.cos(elevAngle * np.pi / 180)**2))
                       - np.sin(elevAngle * np.pi / 180))
    nadirAngle = np.arcsin((RE / (altitude + RE)) *
                           np.cos(elevAngle * np.pi / 180))
    nadirAngle = nadirAngle * 180 / np.pi
    centralAngle = 90 - elevAngle - nadirAngle
    coverageArea = 2 * np.pi * \
        (RE**2) * (1 - np.cos(centralAngle * np.pi / 180))
    earthArea = 4 * np.pi * RE**2
    coverPercent = (coverageArea / earthArea) * 100
    return slantRange, nadirAngle, centralAngle, coverageArea, coverPercent

# In[ ]:
```

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