PARAMETER TUNING OF AN ACTIVE FORCE CONTROL FOR CMG-BASED CONTROLLED SMALL SATELLITE

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DECLARATION

This thesis is the result of my own investigation, except where otherwise stated and has not previously been accepted in substance for any degree and is not being concurrently submitted in candidature for any other degree.

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PARAMETER TUNING OF AN ACTIVE FORCE CONTROL FOR CMG-BASED CONTROLLED SMALL SATELLITE

ABSTRACT

A control moment gyroscope (CMG) system is the most appropriate actuator to be actualized in attitude determination and control system (ADCS) of small satellite. These actuators can provide unique torque, angular momentum and slew rate capabilities to small satellites without any increase in power, mass or volume. This will help small satellites become more agile. A four single gimbal control moment gyroscope (4-SGCMG) is actualize by owing to its efficiency and plays a significant role in attitude control of agile small satellite. However, the efficiency of the CMG system is limited to in executing rapid attitude maneuver and high precision pointing due to external disturbance torques from space environment and internal disturbance of actuator. Hence, an Active Force Control (AFC) technique is enforced and integrated with proportional-derivative (PD) controller to enhance ADCS by rejecting the disturbance torques robustly. In this research, tuning the mass parameter of the dynamic system and the AFC gain, the robustness of AFC technique is triggered. So that, the PD-AFC attitude control scheme able to command the CMG system to give suitable control torque to fulfill the satellite mission. The singularity robust (SR) steering law is integrated with CMG system in order to neglect the singularity state of CMG system by setting the initial gimbal angle, that allowing space missions to be successfully executed. All mathematical models were made amenable for numerical simulations in Matlab®/Simulink® in order to study the effect of AFC parameters tuning on the attitude control performance of the satellite by considering the presence of external disturbance torques.

PARAMETER PENALAAN UNTUK KAWALAN TENTERA AKTIF BAGI BERASASKAN CMG DIKAWAL KECIL SATELIT

ABSTRAK

Sistem kawalan masa giroskop (CMG) adalah penggerak paling sesuai untuk actualized dalam sistem penentuan dan pengawalan sikap daripada satelit kecil. Elektrod ini boleh memberikan unik tork, pengabadian momentum sudut dan membunuh kadar keupayaan satelit kecil tanpa sebarang pertambahan kuasa, jisim atau isipadu. Ini akan membantu satelit kecil menjadi lebih tangkas. Yang empat gimbal tunggal kawalan giroskop ketika ini dicapai oleh kerana kecekapan dan memainkan peranan penting dalam mengawal sikap satelit kecil yang tangkas.Walau bagaimanapun, kecekapan sistem CMG adalah terhad dalam melaksanakan sikap yang pesat manuver dan berketepatan tinggi menunjuk disebabkan oleh gangguan luar torques dari ruang persekitaran dan gangguan dalaman penggerak.Oleh itu, teknik kawalan tentera aktif (AFC) yang dikuatkuasakan dan bersepadu dengan pengawal (PD) berkadar-derivatif untuk meningkatkan ADCS dengan menolak torques gangguan ditubuhkan di seluruh negara.Dalam kajian ini, penalaan parameter jisim sistem dinamik dan keuntungan AFC, keberkesanan teknik AFC dicetuskan. Supaya sikap PD-AFC mengawal skim dapat perintah sistem CMG memberi sesuai mengawal tork untuk memenuhi misi satelit.Itu singularity teguh (SR) stereng undang-undang bersepadu dengan sistem CMG untuk mengabaikan keadaan singularity CMG sistem dengan menetapkan sudut gimbal awal, yang membolehkan Ruang misi akan berjaya dilaksanakan.Semua model matematik telah dibuat dijumlahkan bagi simulasi berangka dalam Matlab®/Simulink® dalam usaha untuk mengkaji kesan parameter AFC Penalaan prestasi sikap kawalan satelit dengan mengambil kira kewujudan gangguan luar tork.

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LIST OF ABBREVIATION

ADCS	Attitude Determination and Control System
AFC	Active Force Control
AFCCA	Active Force Control Crude Approximation
AFCCAIL	Active Force Control Crude and Iterative Learning
ASCMG	Adaptive-Skew Control Moment Gyro
CEACS	Combined Energy and Attitude Control System
CMG	Control Moment Gyro
COTS	Commercially of The Self
DGCMG	Double Gimbal Control Moment Gyro
IGRF	International Geomagnetic Reference Field
IPACS	Integrated Power and Attitude Control System
ISS	International Space Station
LEO	Low Earth Orbit
LQ	Linear Quadratic
LVLH	Local Vertical Local Horizontal
MED	Momentum Exchange Device
MP	Moore-Penrose

MTGAC	Magnetic Torque Gimbal Angle Compensation
PD	Proportional-Derivative
PD+AFC	Proportional-Derivative+Active Force Control
PI	Proportional-Integral
PID	Proportional-Integral-Derivative
rpm	Rotation per minute
RW	Reaction Wheel
SGCMG	Single Gimbal Control Moment Gyro
SR	Singularity Robust
SSTL	Surrey Satellite Technology Ltd
SVD	Single Value Decomposition
TPM	Torque Product Minimization
VPDRF	Variable Periodic Disturbance Rejection Filter
VSCMG	Variable Speed Control Moment Gyro

LIST OF SYMBOLS

α	Orbit angle measure from the ascending node
β	Pyramid skew angle/Angular momentum precession angle
Δ	Differences between two values
δ	Gimbal angles vector of SGCMGs
$\dot{\delta}$	Gimbal angle rate
ζ	Damping ratio
θ	True anomaly
θ	Pitch angle
θ_s	Attitude angle of the satellite
$\dot{\theta}_s$	Angular rate of the satellite
$\ddot{\boldsymbol{\theta}}'$	Measured acceleration of the dynamic system
$\ddot{\theta}_s$	Angular acceleration of the satellite
λ	Singularity robust (SR) inverse steering law constant
$\mu_{\oplus}$	Earth gravitational constant (398600 km3/s2)
p	Atmospheric density
ϕ	Roll angle
ψ	Yaw angle
$\boldsymbol{\omega}$	Satellite angular velocity vector
$\omega_{I/B}$	Inertially referenced satellite's angular velocity relative to inertial coordinate system
ω	Argument of perigee

$\omega_{\boldsymbol{n}}$	Natural frequency
$\omega_{\boldsymbol{o}}$	Orbital frequency
$\dot{\omega}$	Satellite angular acceleration vector
$\dot{\omega}_{I/B}$	Inertially referenced satellite's angular acceleration relative to inertial coordinate system
ω_{cmg}	Angular velocity of the CMG's flywheel
Ω	Right ascension of ascending node
$\Omega(\omega)$	4×4 Skew symmetric matrix
Φ	Euler rotation angle
a	Vernal equinox
$\mathrm{A}(\delta)$	Jacobian matrix
$\mathbf{A}^{+}$	Pseudoinverse of Jacobian matrix
$\mathbf{A}^{\mathbf{T}}$	Transposed of Jacobian matrix
$\mathbf{A}^{\#}$	Singularity robust inverse
$\mathbf{A_s}$	Satellite's exposed area
[A(q)]	Direction cosine matrix expressed in quaternion
a	Orbit semimajor axis
B	Geomagnetic field vector
B	Magnitude of the geomagnetic field vector
C_D	Drag coefficient
C_g	Centre of gravity
C_s	Solar radiation constant (1358 W/m2)
c	Speed of light (3.0×10^{8} m/s)
D	Residual magnetic dipole moment

$\boldsymbol{e}$	Eigenvector of rotation, $\mathbf{e}=[e1\ e2\ e3]T$
H	Total angular momentum of the system
$\dot{\mathrm{H}}_{\mathrm{B}}$	Rate of change of satellite's angular momentum in satellite's body frame
$\dot{\mathrm{H}}_{\mathrm{I}}$	Rate of change of satellite's angular momentum in inertial coordinate system
h	CMG system angular momentum vector
h	Orbital altitude/angular momentum of flywheel
h_o	Angular momentum magnitude of SGCMG
$h_{x,}\ h_{y,}\ h_z$	Angular momentum along satellite's body axes
$\dot{h}$	Generated CMG torque vector
$\dot{h}_x, \dot{h}_y,, \dot{h}_z$	Generated CMG torque along satellite's body axes
$\dot{h}_{cmg},$	Generated CMG torque
I	Satellite's moment of inertia tensor/Principle moment of inertia
$I_{x,}\ I_{y,}\ I_z$	Principle moment of inertia along satellite's body frame
I'	Estimated inertia vector of the dynamic system
i	Orbit inclination
i_s	Incidence angle
K_d	Derivative attitude control gain
K_p	Proportional attitude control gain
M	AFC gain
m	Singularity index
m_s	Satellite total mass
m_{cmg}	CMG system total mass

O_x, O_y, O_z	Axes of the reference frame
$\mathbf{q}$	Quaternion vector
$\mathbf{q_c}$	Commanded quaternion vector
$\mathbf{q_e}$	Quaternion error vector
$\mathbf{q_o}$	Output quaternion vector
$\mathbf{q_{i=1,2,3}}$	Vector elements of the quaternion
$\mathbf{q_4}$	Scalar which gives the magnitude of the rotation angle
$\mathbf{q^*}$	Complex conjugate quaternion
$\mathbf{q_{norm}}$	Normalized quaternion
$\|\|\mathbf{q}\|\|$	Quaternion norm
R	Distance from the centre of the Earth
R_e	Radius of the Earth Re=6370 km
R_o	Orbital radius of the satellite
$\mathbf{RPY_{init}}$	Initial satellite attitude (R – roll along X_B axis, P – pitch along Y_B axis and Y – yaw along Z_B axis)
$\mathbf{RPY_{ref}}$	Reference/desired satellite attitude (R – roll along X_B axis, P – pitch along Y_B axis and Y – yaw along ZB axis)
$S(\omega)$	3×3 Skew symmetric matrix
T	Orbital period
$\mathrm{T_{aero}}$	Aerodynamic torque
$\mathrm{T_c}$	Required magnetic torque vector
$\mathrm{T_d}$	External disturbance torque vector
$\mathrm{T_{dx}, T_{dy}, T_{dz}}$	External disturbance torques exerted on the satellite body
$\mathrm{T_{gg}}$	Gravity-gradient torque vector

T_{ext}	External torque vector
$T_{magnetic}$	Magnetic disturbance torque vector
T_M	Magnetic torque vector
T_{sp}	Solar radiation torque vector
t	Time
$\boldsymbol{u}$	Internal control torque generated by the CMG system
V	Satellite's velocity
W(s)	Weighting function
(X_I, Y_I, Z_I)	Inertia frame coordinate system
(X_O, Y_O, Z_O)	Orbit reference frame coordinate system
(X_B, Y_B, Z_B)	Satellite's body frame coordinate system
$\otimes$	Quaternion multiplication operator

CHAPTER 1: INTRODUCTION

1.1 General Overview

Smaller sizes are required for many of the space missions to execute completely different tasks autonomously on-orbit. Small satellites are catalyzing new applications and have become trendy, even as their terrestrial counterparts the laptop and the smartphone have done. These can embody high exactitude Earth observation and space observation, satellite examination, distributed platforms, constellations, satellite arrival and miniature celestial body probes. Many of those missions would require agility, it significantly will increase the potency and operational envelope of spacecraft and might considerably increase the return of mission information.

However, the space atmosphere is very complex, where there exist assorted disturbances like solar pressure torque, aerodynamic drag as well as magnetic disturbance torque, gravity gradient torque and space debris. Therefore, attitude determination and control system (ADCS) comes in as one of the vital subsystems of a spacecraft and responsible to determine and control the satellite angular orientation as well as maintaining the satellite's primary axes at a required direction throughout the mission despite the external disturbance torques acting on it. The attitude determination and control system help the spacecraft stay on mission objective with the best return in scientific data and to improve the control performance of a spacecraft.

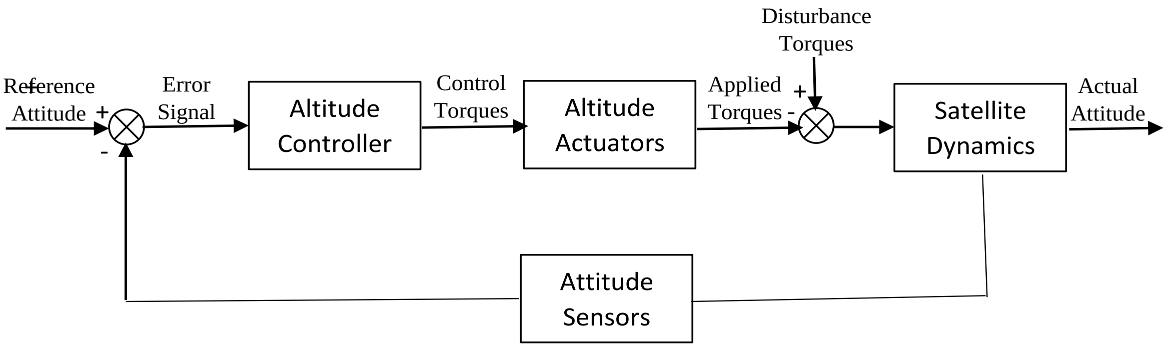

Figure 1.1 Block Diagram of a Closed-Loop Satellite Attitude Control System

All closed-loop control systems have the same basic elements. The purpose of this control loop is to ensure the actual attitude of the satellite approximately equal to the satellite's reference attitude. The desired state is one input to the controller and it compares this state to the actual state from the sensors. It decides on specific commands to send to the actuators, by comparing the difference between these two input signals. The actuator changes along with environmental inputs, affect the final output of the plant while the system sensors detect and measure this output as error signal and feedback into controller to give more precise command to actuator to increase the efficiency.

Usually, the type of actuators and sensor are chosen based on the satellite's mission requirements and its applications. Control Moment Gyro as shown in Figure 1.2 is a kind of spacecraft attitude control actuator and it has the absolute advantage of powerful torque amplification capacity over Reaction Wheel (RW), thruster, and magnetic torquer.

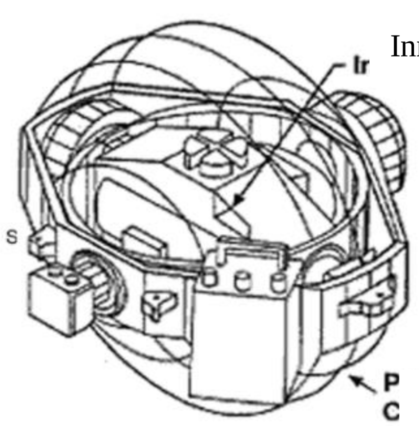

Figure 1.2 Control Moment Gyro

The design of Control Moment Gyroscope (CMG) is tested for BILSAT-1 an enhanced microsatellite and its proven to be a helpful test-bed towards designing a new generation of actuators for agile small satellites (Lappas et al., 2005a). This category of actuators will offer distinctive and uneven agility capabilities even to tiny and lower cost satellites, so that three- axis control with zero momentum actuators are the most suitable attitude control technique. The CMG can offer large output torque over required power for an agile small satellite for future missions such as Earth object tracking, multi-target pointing, stereographic and higher resolution imaging.

Agility, besides increasing the operational envelope of the satellite, will change such satellite to gather a lot of Earth and area science data than before whereas exploitation constant or perhaps fewer resources. This in practice means an immediate increase within the industrial and scientific value of these spacecraft. Small satellites are bound to face some challenging missions in the future that which will need a high degree of agility (high slew rates). The superior electrical power efficiency has been validated when utilizing a CMG cluster compared to that for a Reaction Wheel (RW) system and CMG is highly capable means to controlling agile small satellite (Lappas et al., 2005b).

Moreover, CMG is more applicable to small satellites by lowering its power consumption and offering better attitude control.

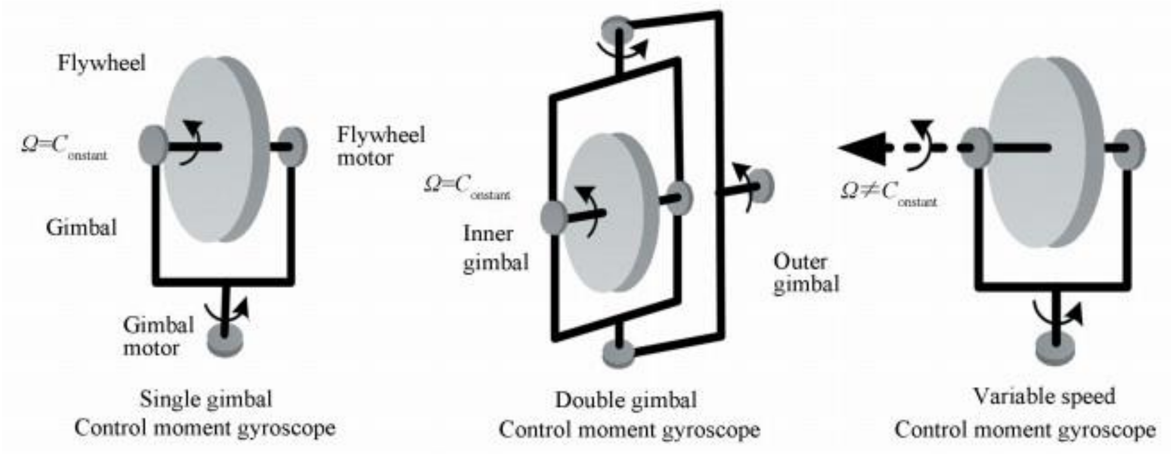

Figure 1.3 Different types of CMG

CMGs are classified into Single Gimbal CMG (SGCMG), Double Gimbal CMG (DGCMG) and Variable Speed CMG (VSCMG) as shown in Figure 1.2. The SGCMG is a one degree of freedom (DOF) exists for momentum direction and offer advantage in terms of torque, of mechanism compared to DGCMG. While, DGCMG consist of two gimbals mounted inner and outer, so that the output torque would be generated for 2 DOF. Variable Speed Control Moment Gyroscope, (VSCMG), both the magnitude and direction of flywheel momentum vector could be changed.

It is incredibly evident that DGCMG and VSCMG have 2 DOF with complex mechanical and dynamical system. SGCMG has been considered more due to the structural simplicity, light mass and in particular the powerful output capacity. Besides, the proposed CMG in cluster of four provide five times more toque output when and lower power consumption when it is compared to a cluster of four reaction wheels (Tissera et al., 2018). Thus, spacecraft equipped with four Single Gimbal CMGs of pyramid configuration arrangement has been used extensively because this configuration provides spherical momentum envelop and also having minimal redundancy (Courie et al., 2018).

1.2 Problem Statement

The major difficulty however in using CMG is the inherent geometric singularity where there is no available torque due to the coplanar state of the output torque of each CMG unit (Wie et al., 2001). Even the most advanced CMG steering logics cannot handle the singularity avoidance problem. A reliable control strategy need to be enforced as for small satellite application, low cost and simple yet robust technique must be considered. The SR inverse methods, which are based on quadratic optimization, were proposed to escape from any kinds of internal singular states while allowing the torque error (Bedrossian et al., 1990). Besides, previous study supports Active Force Control (AFC) is considered an advantageous technology due to its ability to provide robust position control and can easily be integrated into other controllers such as the traditional PD controller. The idea of AFC was first been initiated by (Johnson, 1971) and later (Davison, 1976) based on the principle of invariance and Newton's second law of motion. They showed that, by modelling the disturbances to some known linear differential equation, it is possible to maintain the system set point even with the present of disturbance torques.

The main challenge is no one have attempted to tune the parameters of an active force control, which able to robustly minimize the disturbance torques, since it has been used in various robotic and vibration control applications. However, the second challenge to trigger the robustness of AFC technique is to appropriately tune or estimate the mass parameter of the dynamic system and the AFC gain. So that, the PD-AFC attitude control scheme able increase the control performance to fulfill the satellite mission. In addition, implementing AFC technique into attitude control system, will require strong knowledge about attitude decision and control system of satellite with relevancy the planned controller schemes in order that the viability and credibility of those control techniques for fast perspective manoeuvre and precise attitude pointing missions are often outlined.

1.3 Research Objectives

The aims of this research are:

1. To evaluate the PD + AFC control technique on the attitude control performance for CMG-based controlled small satellite.
2. To study effect of AFC parameters tuning on the attitude control performance of CMG-based controlled small satellite.

1.4 Research Scope

This research only studies the effect of the proposed controllers (PD +AFC) schemes on attitude control performance of CMG based small satellite. The PD controller based algorithm and singularity robust avoidance law is used to study the effectiveness of the CMG based attitude control with respect to three-axis attitude control by Salleh and Suhadis (2014) but, the attitude pointing accuracy is limited (Salleh and Suhadis, 2014).

1.5 Research Approach

The effects of tuning the parameters of proposed controllers on CMG based small satellite are discussed based on the quantitative and qualitative approaches. The simulation software Matlab®-Simulink™ is used to perform the numerical studies of the attitude control performances with respect to planned controllers. The simulation results are presented by comparing the manoeuvring time, attitude manoeuvre performance, dynamics of CMG system as well as the different parameters.

1.6 Thesis Outline

In Chapter 1, new version of spacecraft requirements for future space exploration are discussed. The attitude control system that fits for the objective of space mission are chosen. Moreover, problem statements and research objectives, scope and approach are also presented.

The literature review is presented in Chapter 2, that consist of researches on CMG based small satellite. It covers the issues on using CMG system i.e singularity problem and steering law and the implementation of controllers.

Chapter 3 details the fundamental theories of the dynamics and kinematics of the satellite with the equations are formulated. The CMG system control technique also presented and modelled with singularity law and quaternion feedback controller. Then, the AFC scheme is modelled and implemented into the satellite attitude control system.

The numerical simulations are presented in Chapter 4 with proposed control strategy that have been modelled in Chapter 3. The control performance of the satellite respect to the proposed control scheme are analyzed and discussed.

Lastly in Chapter 5, gives the conclusions of the results obtained in the present study and few suggestions are proposed for future works.

CHAPTER 2: LITERATURE REVIEW

2.1 Configuration of CMG System-Based ADCS

As shown in Figure 2.1 the controller block is feeding commanded torque to steering algorithm block which computes and provide suitable gimbal angle rates to CMG dynamic block. This is to ensure CMG dynamic block can generate the required torque to fulfill the attitude control performance. On the other hand, the CMG dynamic block consist of cluster of CMGs which arranged in certain configuration that able to perform three- axis control of the satellite (Wie, 2005).

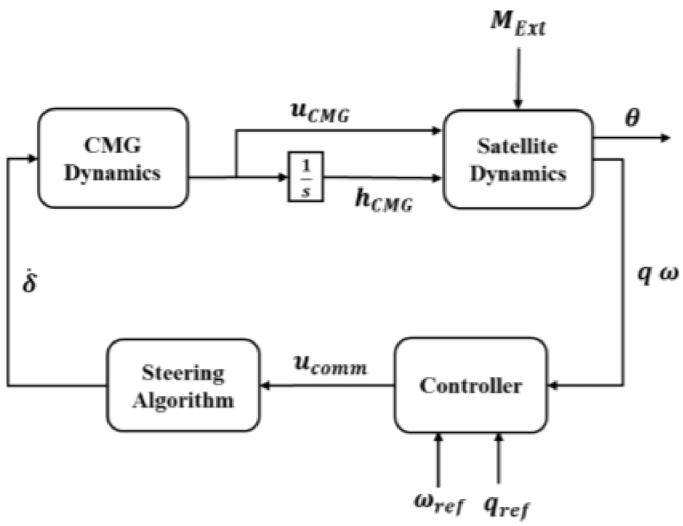

Figure 2.1: Basic closed loop attitude control system using CMG (Wie, 2005)

Moreover, 4-SGCMG cluster in pyramid configuration in Figure 2.2 is used in order to have full 3-axis control and it has capability of providing unique angular momentum, slew rate and torque to agile small satellites. (Lappas et al., 2019) showed that less electrical power and less mass volume are used with this configuration to enhance the capabilities. An optimum angular momentum gained by setting the correct skew angle of the pyramid configuration. So that, an equal amount of control torque is supplied along all three axes, but increasing the number of CMG wheels will lead to the possibility of encountering singular states which it is generally impossible to completely eliminate.

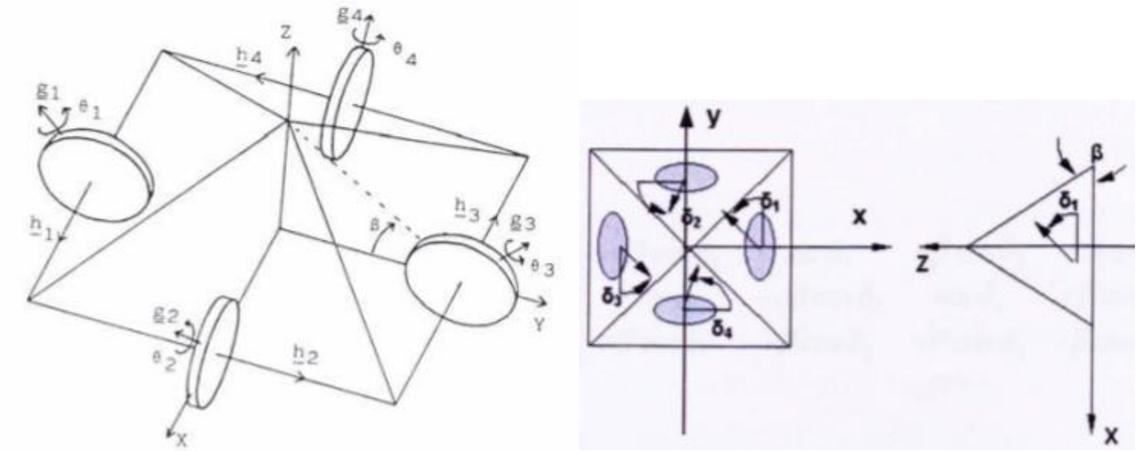

Figure 2.2: 4-SGCMG Pyramid Configuration (Lappas et al., 2005b)

In terms of the uniformity of the momentum envelope, the pyramid mounting arrangement with skew angle, (β =54.73º) of four CMGs are viewed as the most ideal arrangement. This enables us to effectively decide the singularity 'gaps', that is the gimbal angles in which singularities happens and for maximum momentum storage, the ideal skew angle would be, β = 90', which results in a box arrangement of four CMG's (Lappas et al., 2005a). This 4-SGCMG configuration with β =54.73º has been extensively studied and it presents huge challenge for developing singularity robust steering laws.

2.2 Singularity Problem

The CMG cluster whose gimbals are not parallel to one another will trap into singularity if and only if all units output lies in a plane which the command vector is normal to, for some specific gimbal angel sets represented in Figure 2.3. Thusly, the vector which is opposite to the torque plane is known as singular vector and the corresponding gimbal angel sets are called singular gimbal angle also known as gimbal lock CMG system.

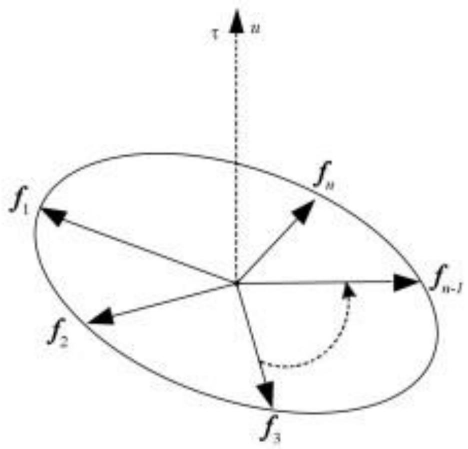

Figure 2.3: Singular torque

Designing of an effective steering algorithm in order to feed required control torque is challenging problem in spacecraft attitude control using SGCMG. The major objective for any successful approach has been the avoidance of singular states that preclude torque generation in a certain direction, the singular direction as shown in Figure 2.3. Only at some gimbal angles configuration the control torque can be generated along the direction if not that gimbal angle is not suitable and it undergoes singularity state. The singularity experienced by CMG system can be divided into external and internal singularity as in Figure 2.4. Total angular momentum which lies on the surface of the momentum envelop is known as external singularity and total angular momentum acts inside the momentum envelop is internal singularity. The external singularities can be effectively foreseen from the given CMG setup and mission profile, thusly they can be considered at the design step. An appropriately planned momentum management scheme can also calm the external singularity problem. On the other hand, the internal singularities are in general hard to envision (Yoon and Tsiotras, 2004).

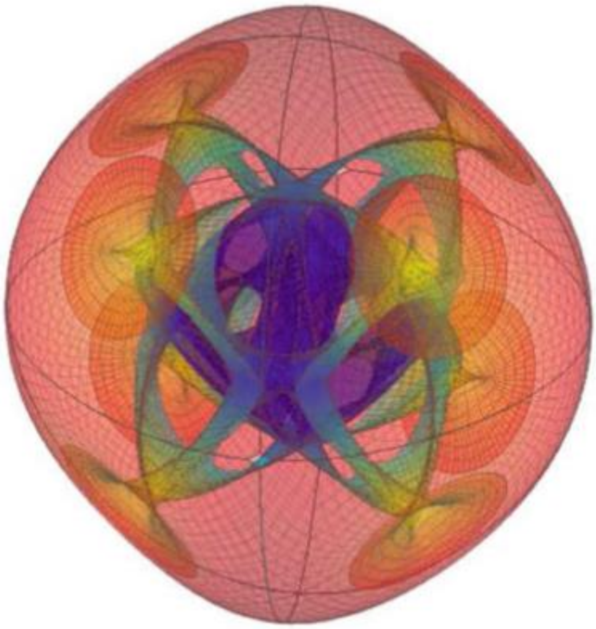

Figure 2.4: External and internal singular surfaces for 4-SGCMG pyramid array (external surface shown in red)

Therefore, the CMG inherent geometric singularity is the main problem which deserve more attention. An approach using the Moore-Penrose pseudoinverse with a nondirectional null-motion algorithm is one of the common steering law presented for single-gimbal control moment gyroscopes and it is considered as an exact method without any torque error (Bedrossian et al., 1990). However, it is still a tangent-based approach the singularity avoidance cannot be guaranteed. Moreover, Oh and Vadali, (1989) have observed that the initial gimbal angles have a profound influence on singularity avoidance. Then, Vadali et al. (1990) showed on how to determine a family of initial gimbal angles that avoid singularities by using a method based on back integration of the gyro torque equation from desired final conditions. Every member of this family is defined as a preferred initial gimbal angle set using the pseudoinverse steering law.

2.2.1 Singularity Avoidance Law

The singularity robust (SR) inverse is introduced by Nakamura and Hanafusa (1986) as an alternative to the pseudoinverse for robotic arm manipulators. This singularity avoidance law modified the MP pseudoinverse steering law by introducing weighting matrix and singularity avoidance parameter into the algorithm. Using the SR-inverse method, an approximate output torque close to the desired torque can be generated, even when the Jacobian is singular. Then later Bedrossian et.al (1990) coupled singularity robust inverse method with nondirectional null algorithm which provides better steering law for SGCMG by producing torque errors in the vicinity of the singularity.

Wei et.al (2001) presented a simple and effective way to escape from any internal singularities by generalising SR inverse. The Moore-Penrose pseudoinverse steering logic proposed in (Wie et al., 2000) is primarily for typical reorientation manoeuvres in which accuracy pointing is not required during reorientation manoeuvres and it completely uses the accessible CMG momentum space in the presence of any singularities. Despite the fact that there are unique missions in which prescribed attitude trajectories are to be exactly tracked in the presence of internal singularities, most useful cases will require an exchange off between robust singularity escape and the resulting, transient pointing errors. It also effectively generates deterministic dither signals when the system becomes near singular, so that any internal singularities can be escaped for any nonzero constant torque commands.

Furthermore, using single gimbal control moment gyros that derived using the Newton-Euler approach of the dynamic equations rotational manoeuvre of spacecraft is studied by Oh and Vadali, (1989) with implementing feedback control laws which are developed by using Liapunov stability theory. The implementation of feedback control law works well in mix with either velocity or acceleration steering laws. It is discovered that the velocity steering law is sufficient for recreations, yet the acceleration steering law gives progressively dependable and valuable information with respect to gimbal torques.

2.3 Attitude Control Technique

The attitude control system is a backbone for a satellite, in order to accomplish its mission for a period of time, so it has to be designed with adequate 3-axis attitude control technique. The design of CMG-based small satellite should include its satellite dynamics and kinematics, disturbance torques, CMG system and its arrangement, singularity avoidance law and most importantly attitude controller which should be designed prior to mission requirements and constraints. There are many studies on CMG-based satellite attitude control system have been carried out using different types of attitude controller and its effect on the attitude control performance will be discussed in following section.

2.3.1 Attitude Controller

Linear parameter-varying (LPV) control strategy for a 4DOF CMG is designed by (Abbas et al., 2013) and compared with linear time-invariant (LTI) controller. Indeed, to achieve a wide range stabilizing operation and improved tracking capability linear parameter-varying (LPV) gain-scheduling techniques are developed. Highly complex dynamic model of a mechanical plant can be controlled with this technique. Contrary to LTI controllers, this LPV controller can stabilize the plant in a wide range of high-performance operations. Yet, after the low-pass filtering by numerical differentiation, the angular velocities were estimated, since the angular velocities were not measured. On the other hand, the gain-scheduled LPV state-feedback controller is not tested on CMG-based satellite to evaluate attitude control performance. However, the dynamics of spacecraft modelled in LPV control theory and Gain-Scheduled (GS) controller is applied using Linear Matrix Inequalities (LMIs) by (Sasaki and Shimomura, 2017) to study the attitude control of a satellite with pyramid-array VSCMGs about all three axes. Through several numerical simulations, they proved that GS controller can attain overall stability and control performance of satellite's attitude control at the same time via LPV control theory.

Fllowed by, a proportional- derivative (PD) based attitude control approach on small satellite with 4-SGCMG cluster presented by Salleh and Suhadis (2014) and singularity robust (SR) steering law implemented with it. This study proved that PD controller is very effective in stabilizing the three axiz attitude manoeuvre and also minimizing the disturbance torque. Besides, tuning the gains of contoller would increase the setteling time of manoeuvre. Nevertheless, preferable intial gimbal angle is choosen for a better efficiency of actuator.

Moreover, Babak Baghi (2016) make use of a non-linear variable gains PD (NVG-PD) controller implementing in flexible satellite model which equipped with a CMG actuator. This non-linear variable gains PD (NVG-PD) controller just uses angular velocity of the rigid body and the attitude parameters as the input data. On the other hand, the performance of the control system is better with this controller in some important features such as diminishing most extreme control torque, decreasing greatest peak of deflection of the appendages and improving robustness of the controller against the orbital disturbances. It is demonstrated that a PD-like controller can universally asymptotically stabilize this satellite by using Lyapunov's direct strategy.

Meanwhile, a fuzzy logic controller is designed for a nanosatellite (QBITO) in a nearby mission (QB50) by Calvo et.al (2016) and its attitude performance compared with a traditional proportional integrative derivative (PID) controller. The comparison made based on attitude performance and dynamic actuations, controller is significantly more efficient than the custom PID for similar accuracy. The fuzzy controller is capable of keeping the error under the 0.01 radian consuming as much as 65% less energy than that needed by the PID controller for the same manoeuvre. However, the fuzzy control takes longer time than the PID to reach a stabile state, even though both converge at the same settling time. Chessab Mahdi, (2016) proposed fuzzy PID controller in attitude determination and control system of nano satellite and compared with fuzzy PD controller. This study proves fuzzy PD requires less overshoot but longer settling time compared to fuzzy PID, but fuzzy PID is more stable than fuzzy PD, since it requires shorter time to reach stable manoeuvre.

Then, Mackunis et.al (2016) developed a neural network (NN) attitude control system for CMG-actuated small satellite and its controller performance is proven via Lyapunov stability analysis. In this study, the satellite is subjected to parameteric uncertainty, uncertain actuator dynamics, and nonlinear disturbance torques. Based on the analysis, for time, this uniform NN attitude tracking controller adapts the varying satellite inertia properties, parametric inertia matrix uncertainty, unknown dynamic friction in the CMG gimbals, and variations in input torque due to electromechanical disturbances in the gimbal loops. Moreover, the NN controller compensates for unmodeled external disturbances and uncertainties caused by unknown static CMG gimbal friction in the input torque. However, simulation study has not been conducted till now by implementing NN controller on CMG-based satellite's attitude control system. Therefore, the controller's performances are not able to evaluate and compared with other type of controllers.

Furthermore, a SGCMG based agile microsatellite is tested with linear quadratic regulator (LQR) and linear quadratic Gaussin (LQG) control technique (Tayebi et al., 2017). This method provides better attitude stabilization with large angles and with more accuracy compared with feedback quaternion and PID controller. Besides, separate steering law of CMGs and different initial conditions is not needed for these controllers, yet these control techniques is very complex to implement.

2.3.2 Active Force Control Technique

The performance of the controllers mentioned in previous study, is not satisfying when it comes to eliminating disturbance torques and also accuracy level to achieve desired attitude. In order to improve manoeuvre of the satellite, the control technique should able to do well estimation of disturbance torque and removing it effectively is needed. Not only that, the control tecnique should able to supports CMG-based small satellite applications and simple hardware used for onborad computation.

Therefore, Mustafa Mohebbia (2016) studied the vibration of a three degree-of-freedom (DOF) model using short length drive shaft has been robustly vanquish through the implementation AFC along with a conventional PID controller. This study proved that when a PID controller was used, it took longer execution time to reduce the vibration and slight inbalance during frequency oscillation. However, with help of AFC technique the vibration can be eliminated more effeciently by feeding estimated disturbance force through inverse transfer function of the actuator and the signal summed up with PID control. The combination of PID control and AFC strategy provide outstanding performnce in reducing vibration with the amplitude hovering a zero datum without any offset and yielding an extremely low frequency trending (Mustafa Mohebbia, 2016). Moreover, real time vibration control of active suspension system is studied with and without AFC using Iterative Learning Algorithm (ILA) along with PID controller, indeed the AFC-ILA scheme gives an excellent performance in compensating the disturbances (Kalaivani et al., 2013). The settling time for control scheme PID without AFC-ILA is 11.5 second, while for control scheme PID with AFC-ILA is 2.9 second which is significant difference in performance. Therefore, the AFC technique is compatible with conventional controllers and has potential applying to any system dynamics. Nevertheless, the control algorithm is mathematically simple and computationally not intensive.

Not only that, a fuzzy PD controller with AFC scheme is employed in the satellite attitude control system equipped with reaction wheels arrangement in pyramid configuration and studied by Ismail and Varatharajoo (2016). In this study, fuzzy PD controller act as attitude stabilizer, while, AFC act as disturbance estimator and eliminate the external disturbance torques. Three axis satellite attitude control is performed using reaction wheels and under momentum dumping mode. Based on previous study, the concept of AFC used in satellite attitude control application relies on the measurement of angular acceleration, which can be translated to the estimated torques via crude approximation of the satellite inertia. In addition this study proves that the three axis attitudes accuracies can be improved up to ±0.001 degree through the fuzzy PD-AFC control technique for attitude control performance and the three-axis wheel angular momentums were sustained during the attitude control tasks (Ismail and Varatharajoo, 2016).

On the other hand, previous study by Varatharajoo et al (2011) on two degree of freedom spacecraft attitude controller is modelled with PD-AFC controller for 50kg small satellite and also implementing combined energy and attitude control system (CEACS). The results were compared CEACS with and without AFC. The output with PD-AFC attitude control performances are superiorly better than solely PD type controller (Varatharajoo et al., 2011a). Besides, the tuning of AFC system gains provides even better control performnace when the combination of estimated inertia and AFC constant are tuned correctly (Varatharajoo et al., 2011b).

Hence the AFC technique has ability to eliminate external disturbance torques more effeciently by feeding estimated disturbance force through inverse transfer function of the actuator and the signal can be summed up with conventional controllers. Besides, it also act as robust controller scheme when the AFC parameters are chosen correctly by comparing several values. In fact, the simplicty of this control scheme reduces the hardware complexity, easy for onborad computation, and also easy undergo tuning process. Eventually, no study had been conducted implementing AFC technique with conventional controller on CMG-based satellite to evaluate its attitude control performance. Thus, this research is conducted to study the attitude manoeuvre operation in all three axes by intergrating AFC scheme along with PD controller for CMG-based controlled small satellite. Based on the studies, this control technique should able to improve the performance of attitude control, stabilize the satellite by rejecting external disturbance torque and provides rapid manoeuvring capability.

CHAPTER 3: METHODOLOGY

In this chapter the fundamental theories to build a CMG-based controlled small satellite are described. At the beginning, type of satellite and its mission configuration is defined, followed by the external disturbance torques that causes the satellite to be unstable and orbital parameters. Then, the satellite's equation of motion and CMG dynamics are explained. Finally, the implementation of AFC technique with PD controller and its working principle is explained.

3.1 Satellite Mission and Configuration

In this thesis, a microsatellite that operates in circular low Earth orbit (LEO) for earth observation mission has been considered. As the requirement of the mission the satellite should able to perform rapid attitude manoeuvre along with high precision pointing, so SSTL microsatellite as shown in Figure 3.1 that employs a flexible modular design with capable

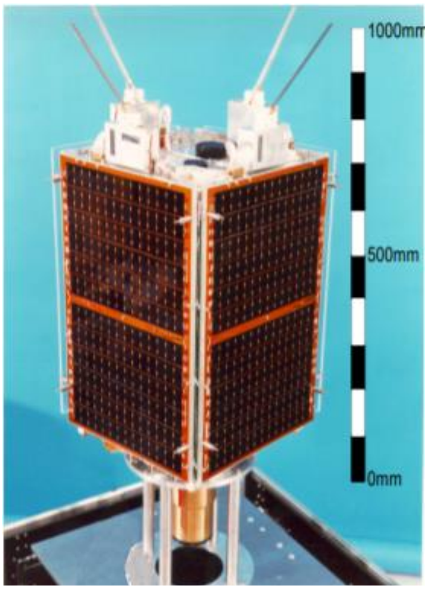

Figure 2.1: SSTL Microsatellite

of supporting wide range of mission in LEO has been considered. This spacecraft total mass is 50-70 kg and designed for 3 year or more, but its proven to over 10 years operation. 4-SGCMGs system in pyramid arrangement is fitted in this satellite as an actuator for satellite attitude control. The configuration of the satellite with 4- SGCMGs system is illustrated in Figure 3.2.

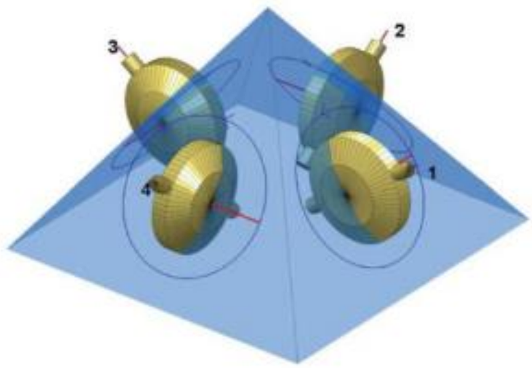

Figure 3.2: Pyramid Configuration of 4-SGCMG (Courie et al., 2018)

3.2 External Disturbance Torques

The usual external disturbance torques that affects the LEO satellites are the aerodynamic torque, solar radiation torque, gravity gradient torque and magnetic field torque. Each external disturbance torques varies with satellite attitude as shown in Figure 3.3. For each related axis these external disturbances can be divided into the constant and harmonic components. Harmonic torques are varying in a sinusoidal mode in an orbit, while constant torques are accumulating with time and not averaging out over an orbit (Larson and Wertz, 1992).

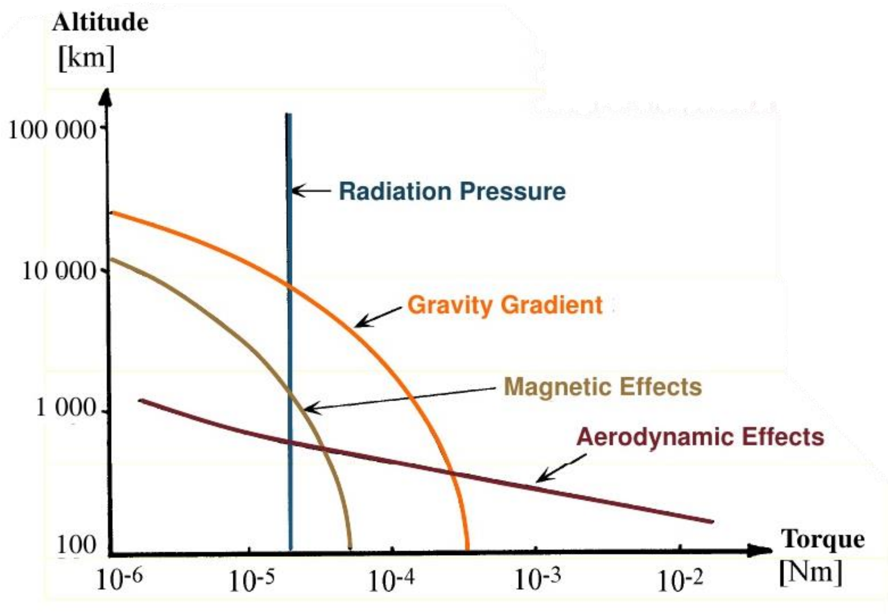

Figure 3.3: External Disturbance Torques versus Satellite's Altitude (Larson and Wertz, 1992)

3.2.1 Aerodynamic Torque

The aerodynamic torque will change according to the atmospheric density as the satellite attitude varies as shown in Figure 3.3. The aerodynamic torque defined as

$$T_{aero} = \frac{1}{2}\left(\rho C_D A_S V^2\right)\left(C_{pa} - C_g\right) \tag{3.1}$$

where $F = \frac{1}{2}\left(\rho C_D A_S V^2\right)$, the aerodynamic force, ρ is the atmospheric density which depends on the atmospheric condition, C_D is the is the drag coefficient usually between 1 and 2 for free flow molecular, A_S is the satellite’s exposed area, V is the velocity of the satellite. C_{pa} is center of aerodynamic pressure and C_g is the center of gravity of the satellite. The aerodynamic torque is the sum of constant and harmonic quantities and it is modelled as follows

$$T_{aero} = T_{aero,constant} + T_{aero,harmonic} \tag{3.2}$$

where, $$T_{aero,constant} = \frac{T_{aero,max}}{2} = \frac{T_{day,max}}{2} + \frac{T_{night,max}}{2} \tag{3.3}$$

and $$T_{aero,harmonic} = (T_{aero,max} + T_{aero,constant})\ \sin(\omega_0 t) \tag{3.4}$$

3.2.2 Solar Radiation Torque

Based on Figure 3.3 the solar radiation does not depend on satellite's attitude and its order of magnitude is near to 10^{-5} Nm. Solar radiation torque happens on satellite's body and defined as

$$T_{sp} = F\left(C_{sp} - C_g\right) \tag{3.5}$$

where C_{sp} is the center of solar pressure, C_g is the center of gravity while the solar force, F as follows

$$F = \frac{C_s}{C} A_s (1 + q_r)\cos(i_s) \tag{3.6}$$

where C_s is the solar radiation constant, c is speed of light (3×10^8 m/s), A_s is the satellite's exposed surface area to sun, q_r is the reflectivity factor ranging from 0 to 1 and i_s is the incidence angle of the sun. The harmonic component of the solar radiation is considered as

$$T_{solar,harmonic} = (T_{solar,max} + T_{solar,constant})\ \sin \omega_0 t \tag{3.7}$$

Where the satellite magnitude changes periodically along with the orbit.

3.2.3 Gravity Gradient Torque

Unequal body design of the satellite causes the unbalance moment of inertia on body and lead to gravity gradient torque. The gravity gradient torque is considered as constant torque, when the satellite points the Earth. The general equation of gravity gradient torque for circular orbit is

$$T_{gg} = 3\frac{\mu_{\oplus}}{R_0}\begin{bmatrix}(I_Z - I_Y)\phi \\ (I_Z - I_X)\theta \\ 0\end{bmatrix} \tag{3.8}$$

where I is the satellite moments of inertia, $\mu_{\oplus}$ is the Earth's gravitational constant ($\mu_{\oplus}$ = 398600 km^3/s^2), R_0 is the orbit radius, ϕ and θ are the euler angles.

3.2.4 Magnetic Field Torque

When the satellite moves through the Earth's magnetic field, residual magnetic dipole moment is generated, which causes the satellite to be affected by geomagnetic disturbance torques. The magnetic field torques given by

$$T_{magnetic} = D\,\boldsymbol{B_{LVLH}} \tag{3.9}$$

where B_{LVLH} is the Earth's magnetic field (Tesla) and D is the residual dipole (Am^2). Orbital altitude, orbital inclination and residual magnetic dipole moment of a satellite are the prime causes of magnetic disturbance torque (Larson and Wertz, 1992).

3.3 Orbital Parameters

The motion of the satellite in space is modeled using the orbital parameters as shown in Figure 3.4 and these orbital parameters will be the input parameters in the simulation analysis.

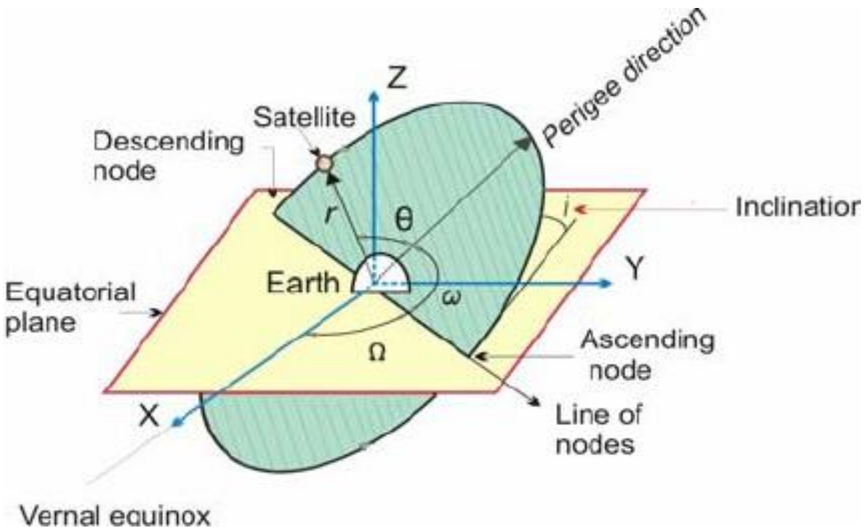

Figure 3.4: Orbital Elements

The angle from the Vernal Equinox to the ascending node is the point where the right ascension of ascending node (RAAN), **Ω** of satellite that passes through the equatorial plane. The inclination angle of an orbit, ***i*** is the angle between the satellite orbit plane and the Earth's equatorial plane. Usually, when the inclination angle is equal to 0˚ or 180˚, the orbit is called as an equatorial orbits and orbits with inclination angle of 90˚ is called as a polar orbit. The argument of perigee, **ω** is an angular measurement within the orbital plane from ascending node to perigee in the direction of the satellite motion. Thus, these three elements **(Ω, *i*, ω)** explain the orientation of the orbit. Moreover, the true anomaly, **θ** describes the satellite's location with respect to the time and is the only orbital element that changes with time (Larson and Wertz, 1992).

The circular orbit is considered in this study for analysis. The sum of Earth's radius, $\boldsymbol{R_e}$ and satellite's attitude, **h** is equal to the semi-major axis, **a,** since the orbit is circular. Therefore, with any given value of the altitude the orbital period, T_0 can be calculated as follows