ENHANCED ATTITUDE CONTROL STRUCTURE FOR SMALL SATELLITES WITH REACTION WHEELS

ZULIANA BINTI ISMAIL

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ENHANCED ATTITUDE CONTROL STRUCTURE FOR SMALL SATELLITES WITH REACTION WHEELS

By
ZULIANA BINTI ISMAIL

Thesis Submitted to the School of Graduate Studies, Universiti Putra Malaysia, in Fulfilment of the Requirements for the Degree of Doctor of Philosophy

August 2019
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Abstract of thesis presented to the Senate of Universiti Putra Malaysia in fulfilment of the requirement for the degree of Doctor of Philosophy

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August 2019

Chair : Y.Bhg. Prof. Dato’ Renuganth Varatharajoo, PhD, Ir
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Attitude accuracies of a three-axis satellite are highly influenced by space environment disturbances and uncertainties. Similar to actuators, an attitude controller also plays an important role and must be robust enough to cope with any disturbances and uncertainties. Various controllers have been used for satellite attitude controls either linear or nonlinear control theories. This thesis presents an enhanced attitude control structure for a small satellite with reaction wheels (RWs) and the wheel angular momentum unloading control using magnetic torquers (MTQs). In order to improve the attitude control performances, a proportional derivative-active force control (PD-AFC), and a Fuzzy PD-AFC are developed. For the momentum unloading control, a Fuzzy-proportional integral (Fuzzy-PI) is developed to remove the excess wheel momentum. Using the PD-AFC and Fuzzy PD-AFC, the actual disturbances torques are considered totally rejected by the system without having to have any direct prior knowledge on the actual disturbances itself. These days, however, satellites have become increasingly more complex, with many additional components, such as antennas, cameras, solar panels and mechanical manipulators. These components introduce flexible mode which results in a satellite dynamic system becoming highly nonlinear. Therefore, a robust nonlinear controller such as sliding mode control (SMC) is highly desirable. Besides, a number of studies have shown that, fractional order controller (FOC) could enhance the control system performance due to its extra degrees of freedom. In this thesis, a fractional order sliding mode control (FOSMC) is developed. In fact, this current work will be one of the maiden works on FOSMC for small satellites. All the proposed controllers were also tested for a satellite with only two functional RWs, in which the control allocation technique is proposed to solve the underactuated satellite attitude control problem. All the relevant attitude control architectures are developed together with their governing equations. Eventually, all control algorithms are numerically treated and analysed. The research results obtained proved that the PD-AFC, Fuzzy PD-AFC and FOSMC to be successful in achieving the overall stability attitude control system in the presence of external disturbances and
uncertainties, i.e., PD-AFC ($\pm 0.0040^\circ - 0.0055^\circ$); Fuzzy PD-AFC ($\pm 0.0010^\circ - 0.0015^\circ$); FOSMC ($\pm 0.00020$), and with the Fuzzy-PI for momentum unloading control whereby, the wheel momentum can be well maintained. Finally, the research for underactuated satellite attitude control performances using two RWs have been also successfully demonstrated and the research results proved that the control allocation technique provides a good performance in controlling the satellite attitude.
Abstrak tesis yang dikemukakan kepada Senat Universiti Putra Malaysia sebagai memenuhi keperluan untuk ijazah ijazah Doktor Falsafah

STRUKTUR KAWALAN ATITUD DIPERTINGKATKAN UNTUK SATELIT KECIL DENGAN EMPAT RODA REAKSI

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Ketepatan atitud satelit tiga paksi sangat dipengaruhi oleh gangguan persekitaran angkasa dan ketidakpastian. Sama seperti penggerak, pengawal atitud juga memainkan peranan penting dan mesti cukup mantap untuk mengatasi sebarang gangguan dan ketidakpastian. Pelbagai pengawal telah digunakan untuk mengawal atitud satelit sama ada teori kawalan linear atau tidak linear. Tesis ini membentangkan struktur kawalan atitud yang dipertingkatkan untuk satelit kecil dengan roda reaksi dan kawalan momentum roda reaksi menggunakan tork magnet. Untuk meningkatkan prestasi atitud, PD-AFC dan Fuzzy PD-AFC direka bentuk. Untuk kawalan momentum roda reaksi, Fuzzy-PI direka untuk mengurangkan momentum yang berlebihan. Menggunakan PD-AFC dan Fuzzy PD-AFC, tork gangguan boleh dibatalkan oleh sistem tanpa memerlukan pengetahuan terlebih dahulu mengenai gangguan sebenar tersebut. Kini, sistem satelit semakin kompleks, dengan banyak komponen tambahan, seperti antena, kamera, panel solar dan manipulator mekanik. Komponen ini memperkenalkan mod fleksible yang mengakibatkan sistem dinamik satelit menjadi sangat tidak linear. Oleh itu, pengawal bukan linear yang mantap seperti pengawal mod gelongsor SMC amat sesuai. Selain itu, beberapa kajian telah menunjukkan bahawa, fractional order control (FOC) dapat meningkatkan prestasi sistem kawalan. Dalam tesis ini, gabungan FOC dan SMC (FOSMC) direka. Malah, kajian ini akan menjadi salah satu kajian sulung untuk FOSMC diaplikasikan di satelit-satelit kecil. Semua pengawal yang dicadangkan juga telah diuji untuk satelit yang hanya ada dua roda reaksi yang berfungsi, di mana teknik peruntukan kawalan dicadangkan untuk menyelesaikan masalah pengendalian atitud satelit yang kegagalan roda reaksi. Kesemua arkitektur kawalan atitud yang relevan dibangunkan bersama dengan persamaan asas. Seterusnya, semua algoritma kawalan diuji secara berangka dan keputusan dianalisis. Keputusan kajian yang diperoleh membuktikan bahawa PD-AFC, Fuzzy PD-AFC dan FOSMC berjaya mencapai kestabilan keseluruhan walaupun adanya gangguan luar dan ketidakpastian sebagai contoh; PD-AFC (±0.0040°-0.0055°); Fuzzy PD-AFC (±0.0010°-0.0015°); FOSMC (±0.000200), dan dengan Fuzzy-PI untuk kawalan momentum, momentum roda reaksi dapat
dikawal dengan baik. Akhir sekali, kajian yang hanya ada dua roda reaksi yang berfungsi telah berjaya ditunjukkan dan keputusan kajian membuktikan bahawa teknik peruntukan kawalan memberi prestasi yang baik dalam mengawal attitude satelit.
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This thesis was submitted to the Senate of Universiti Putra Malaysia and has been accepted as fulfilment of the requirement for the degree of Doctor of Philosophy. The members of the Supervisory Committee were as follows:

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LIST OF ABBREVIATIONS

AFC  Active Force Control
ACS  Attitude Control System
ADCS Attitude Determination and Control Subsystem
CEACS Combined Energy and Attitude Control System
CMG  Control Moment Gyros
CLF  Control Lyapunov Function
DCM  Direct Cosine Matrix
DOBC Disturbance Observer Based Control
FIS  Fuzzy Inference System
FOC  Fractional Order Control
FOSMC Fractional Order Sliding Mode Control
GPS  Global Positioning System
LEO  Lower Earth Orbit
MF   Membership Function
MTQ  Magnetic Torquer
PD   Proportional-Derivative
PI   Proportional-Integral
PID  Proportional–Integral–Derivative
RW   Reaction Wheel
SMC  Sliding Mode Control
IAA  International Academy of Astronautics
LQR  Linear Quadratic Regulator
MIMO Multi Input, Multi Output
NASA National Aeronautics and Space Administration
# LIST OF SYMBOLS

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>$a$</td>
<td>Semi-major axis [$m$]</td>
</tr>
<tr>
<td>$\zeta$</td>
<td>Damping Ratio for the attitude control loop</td>
</tr>
<tr>
<td>$\mu_\oplus$</td>
<td>Earth gravitational constant, $\mu_\oplus = 3.986 \times 10^{14} m^3/s^2$</td>
</tr>
<tr>
<td>$\mu_f$</td>
<td>Magnetic field’s dipole strength, $\mu_f = 7.9 \times 10^{15} Tesla \cdot m^3$</td>
</tr>
<tr>
<td>$\omega$</td>
<td>Argument of perigee</td>
</tr>
<tr>
<td>$\omega_o$</td>
<td>Orbital frequency [rad/s]</td>
</tr>
<tr>
<td>$\omega_{Earth}$</td>
<td>Earth’s rotation frequency, $\omega_{Earth} = 7.29211515 \times 10^{-5} rad/s$</td>
</tr>
<tr>
<td>$\omega_n$</td>
<td>Natural Frequency for the attitude control loop [rad/s]</td>
</tr>
<tr>
<td>$\omega_x, \omega_y, \omega_z$</td>
<td>Satellite’s body angular rate [rad/s]</td>
</tr>
<tr>
<td>$\dot{\omega}_x, \dot{\omega}_y, \dot{\omega}_z$</td>
<td>Satellite’s body angular acceleration [rad$^2$/s$^2$]</td>
</tr>
<tr>
<td>$C(q)$</td>
<td>Transformation Matrix</td>
</tr>
<tr>
<td>$\phi$</td>
<td>Roll attitude [rad or degree]</td>
</tr>
<tr>
<td>$\theta$</td>
<td>Pitch attitude [rad or degree]</td>
</tr>
<tr>
<td>$\psi$</td>
<td>Yaw attitude [rad or degree]</td>
</tr>
<tr>
<td>$\Phi$</td>
<td>Euler angle of rotation</td>
</tr>
<tr>
<td>$\Omega$</td>
<td>Right ascension of ascending node</td>
</tr>
<tr>
<td>$\nu$</td>
<td>True anomaly</td>
</tr>
<tr>
<td>$[R_{rw}]$</td>
<td>Reaction wheel allocation matrix</td>
</tr>
<tr>
<td>$B^2$</td>
<td>Magnitude of the Earth’s magnetic field vector [Tesla]</td>
</tr>
<tr>
<td>$B_0$</td>
<td>Earth’s magnetic field vector in the Body coordinate system [Tesla]</td>
</tr>
<tr>
<td>$C_s$</td>
<td>Average magnetic field intensity in low earth orbit, $B_0 = 2.5 \times 10^{-5} Tesla$</td>
</tr>
<tr>
<td>$c$</td>
<td>Speed of light, $c = 3 \times 10^8$ m/s</td>
</tr>
<tr>
<td>$e$</td>
<td>Euler axis of rotation</td>
</tr>
<tr>
<td>$h$</td>
<td>Satellite’s Altitude</td>
</tr>
<tr>
<td>$h_w$</td>
<td>RW momentum of a satellite [kgm$^2$]</td>
</tr>
<tr>
<td>$i$</td>
<td>Satellite orbit inclination with respect to the equatorial plane</td>
</tr>
<tr>
<td>$J$</td>
<td>Satellite moments of inertia [kgm$^2$]</td>
</tr>
<tr>
<td>$\Delta J$</td>
<td>Uncertainty of moment of inertia tensor</td>
</tr>
<tr>
<td>$K_p$</td>
<td>Proportional attitude control gain</td>
</tr>
<tr>
<td>$K_i$</td>
<td>Integral attitude control gain</td>
</tr>
<tr>
<td>$K_d$</td>
<td>Derivative attitude control gain</td>
</tr>
<tr>
<td>$m$</td>
<td>Magnetic dipole moments of the magnetic torquers [Am$^2$]</td>
</tr>
<tr>
<td>$M_E$</td>
<td>Geomagnetic strength of dipole, $7.9 \times 10^{15} Wb \cdot m^{-1}$</td>
</tr>
<tr>
<td>$q_e$</td>
<td>Error of quaternion</td>
</tr>
<tr>
<td>$q_r$</td>
<td>Solar reflectance factor</td>
</tr>
<tr>
<td>$R_E$</td>
<td>Radius of the Earth, $R_E = 6378 km$</td>
</tr>
<tr>
<td>$s$</td>
<td>Sliding mode surface</td>
</tr>
<tr>
<td>$T_o$</td>
<td>Period of orbit</td>
</tr>
<tr>
<td>$T_{tw}$</td>
<td>Applied torque vector by reaction wheels [Nm]</td>
</tr>
<tr>
<td>$T_m$</td>
<td>Magnetic torque vector induced by magnetic torquers [Nm]</td>
</tr>
<tr>
<td>$T_d$</td>
<td>External disturbance torque vector [Nm]</td>
</tr>
<tr>
<td>$T_{pd}$</td>
<td>PD control torque [Nm]</td>
</tr>
<tr>
<td>$T_{pda AFC}$</td>
<td>PD-AFC control torque [Nm]</td>
</tr>
</tbody>
</table>
\[ T_f \quad \text{Fuzzy control torque [Nm]} \]
\[ T_{fp_{d}a_{f}c} \quad \text{Fuzzy PD-AFC control torque [Nm]} \]
\[ T_{smc} \quad \text{Sliding mode control torque [Nm]} \]
\[ T_{eq} \quad \text{Equivalent control torque [Nm]} \]
\[ T_{sw} \quad \text{Switching control torque [Nm]} \]
\[ X_B, Y_B, Z_B \quad \text{Satellite’s body coordinate system} \]
\[ X_D, Y_D, Z_D \quad \text{Desired coordinate system} \]
\[ X_{IE}, Y_{IE}, Z_{IE} \quad \text{Inertial Earth (IE) coordinate system} \]
\[ \times \quad \text{Cross-product operator} \]
\[ \eta \quad \text{Positive constant for sliding condition} \]
\[ \lambda \quad \text{Minus slope of the sliding surface} \]
\[ \phi \quad \text{Euler angle of rotation} \]
\[ b \quad \text{Boundary layer} \]
CHAPTER 1

INTRODUCTION

1.1 General Overview

A satellite is a space vehicle launched by a rocket and placed in an orbit around the Earth. It is designed for various applications such as Earth observation, weather forecasting, communications, scientific exploration, defence purposes, and also for university experiments. In recent years, the development of small satellites has gained more attention than that of the larger satellite because it can be developed to actualise advanced space missions at low cost in short amount of time, with less complexity and the ability to provide valuable scientific returns (Inamori, 2012).

According to International Academy of Astronautics (IAA), small satellites can be categorised into four groups based on their weight which are minisatellite (less than 1000 kg), microsatellite (less than 100kg), nanosatellite (less than 10 kg), and picosatellite (less than 1 kg) (Ram S & Joseph N, 2014). The progressive development of Micro-Electro-Mechanical Sensors (MEMS) and Commercial Off-the-Shelf Components (COTS) contributed to the emerging of small satellites space mission in the recent years.

The Attitude Determination and Control System (ADCS) is one of the satellite subsystems which is the primary field area covered in this research. Most of the satellite components can be miniaturised to reduce the cost and at the same time able to retain their high performance (Nicolai et al., 2014). Nevertheless, all must-have seven subsystems in any satellites are summarised in Table 1.1.

<table>
<thead>
<tr>
<th>Satellite Subsystems</th>
<th>Functions</th>
</tr>
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<tbody>
<tr>
<td>Attitude Determination and Control System (ADCS)</td>
<td>Determines and controls the satellite angular orientation throughout the mission.</td>
</tr>
<tr>
<td>Telemetry, Tracking and Command (TTC)</td>
<td>Provides satellite housekeeping data to the ground station.</td>
</tr>
<tr>
<td>Command and Data Handling (CDH)</td>
<td>Distributes commands received from the ground station to the satellite.</td>
</tr>
<tr>
<td>Power</td>
<td>Provides and manages power to the satellite.</td>
</tr>
<tr>
<td>Structures and Mechanism</td>
<td>Provides an interface for all subsystems</td>
</tr>
<tr>
<td>Guidance and Navigation:</td>
<td>Determines the satellite's state vector and its orbital elements.</td>
</tr>
<tr>
<td>Thermal</td>
<td>Provides acceptable temperature ranges for all the satellite’s component.</td>
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</table>

The ADCS can be categorised into two separate subsystems, namely the Attitude Control System (ACS) (i.e. the control actuators) and the Attitude Determination
System (ADS) (i.e., the attitude and angular velocity sensors). The commonly used attitude sensors include Sun sensors, Earth sensors, magnetometers, star trackers, Global positioning systems (GPS), and gyroscopes, where the details are described in (Larson & Wertz, 2005). However, the primary focus of this thesis is only on ACS studies. ACS is a substantial subsystem of a satellite in controlling and maintaining the high accuracy autonomous attitude pointing and rapid slewing capabilities in the presence of environmental and systematic errors.

Upon separating from the launch vehicle, the satellite will tumble to an undefined angular rate. As shown in Figure 1.1, this first phase is called as the detumbling mode where the satellite’s angular rate is reduced to a lower speed through the attitude acquisition process. Then, the satellite will be put into the safe mode (sun pointing) where the satellite slews such that its solar panels face the sun to allow the battery to be charged and this mode is also used in backup emergency in case normal operation mode fails. Next, in the idle mode, the satellite’s batteries will be charged efficiently and is standby for inertial pointing mode, nadir pointing mode and target pointing mode (Pong et al., 2010).

![Figure 1.1: Satellite’s phase (Klinkner, 2012)](image)

In the inertial pointing mode, the cameras can be pointed to a celestial object like stars, the sun, or the moon. In the nadir pointing mode, the payload cameras are pointed
"directly down" towards the Earth and while in target pointing mode, the satellites are pointed towards a specific target. Besides the attitude pointing, the attitude tracking is also required to be performed for specific missions especially for the satellite with a limited supply of electrical power where it has to track a Sun-optimal trajectory. Other than that, the communication satellites have to track other satellites passing by either for transmitting or receiving data. Furthermore, some satellites need the attitude tracking if the satellites have the mission to collect measurement and to take pictures of objects from far away (Mohammad & Ehsan, 2008). Thus, to enable the attitude control for the entire duration of the mission with agile manoeuvring capabilities and high pointing accuracies, the robust attitude control system is highly desired.

Besides, the satellite’s lifespan is in the range of 1 to 10 years, and throughout the mission, the satellite design, development, and operation can be affected by the natural space environment. In addition, the satellites in their orbits are also influenced by unwanted motions such as the libration, nutation, and precession if no countermeasures are performed by the satellite. The attitude stabilisation methods can be categorised into two main methods, which are the active control method and the passive control method. The brief description of both methods is summarised as illustrated in Figure 1.2.

**Figure 1.2: Satellite attitude stabilization methods** (Larson & Wertz, 2005)

In the early days of the space era, passive controls were the favoured option for a space mission. A passive technique using gravity gradient stabilisation is sufficient enough for early satellite missions such as Explorer 1 and Intelsat 1. Less hardware used, fuel-free, simple, and low cost are the factors contributing to the adoption of passive techniques (Sidi, 1997). In the middle of the space era, passive techniques were no longer relevant due to the transition from a small space mission to the more massive satellite with the sizes ranging in hundreds of kilograms.

Meanwhile, three-axis stabilisation technique is preferable compared to spin stabilisation because the three-axis stabilisation could provide greater pointing accuracy in the order of very accurate milli-radians and could allow solar arrays to be
continuously oriented towards the sun (Ram S & Joseph N, 2014). Nevertheless, in an effective three-axis stabilisation technique, the satellite must have the actuators to produce angular torque. The selection of actuator is based on the mission of the satellite. For a short mission duration, a thruster is a good option for spacecraft like Shuttle and Soyuz which does not require high fuel consumption for attitude control. Furthermore, these spacecraft are not built for high pointing accuracy requirements. Besides the thrusters, the momentum exchange devices like reaction wheels (RWs) and control moment gyroscopes (CMGs) are also the common actuators adopted. Both RWs and CMGs are capable of spinning freely and are functioning based on the conservation of angular momentum (zero-momentum biased). When their angular momentum is changed, the angular momentum of the satellite must also change to conserve its net angular momentum.

Either RWs or CMGs are the ultimate options that can provide higher attitude accuracy for satellite attitude control (Larson & Wertz, 2005). RWs spin along a fixed axis at a variable speed and the angular momentum is varied by increasing or lowering the speed. Meanwhile, CMGs spin along a rotating axis at a constant speed and the angular momentum is varied by rotating their spin axis. The reaction torque of RWs acts on the satellite as the wheel speed varied, while the wheel speed is fixed for CMGs resulting in the change of spin axis’s direction relative to the satellite. CMGs are suitable for the three-axis control but are often not considered to be used on small satellites due to the complexity of the mechanical and control system needed to implement an effective CMG, see Figure 1.3.

![Diagram of a) RW torque and b) CMG torque](Votel & Sinclair, 2012)

The magnetic control via the use of magnetic torquers (MTQs) is another favoured option for small satellites either for attitude control or momentum unloading tasks. MTQ are advantageous regarding their low cost and lightweight as they contain no moving parts, making them less vulnerable to failure. The major obstacle in the magnetic actuation is that a magnetic torque will only produce a maximum torque when aligned with the local magnetic field vector. The available torque, therefore, depends on the current local magnetic field vector, and independent torques on all three axis of a control system using three orthogonal magnetic torques are hard to achieve. Accordingly, the yaw axis of the satellite is not controllable over the magnetic poles of the Earth, and the roll axis will lose its torque over the equator. Since the magnetic field is continuously changing, magnetic control has become nonlinear and time-varying. In this study, the RWs are used to control the satellite’s attitude while the MTQs are used to unload the wheel momentum.
1.2 Problem statements

Technically, the satellite systems are complicated due to several issues such as time variant, delays, and nonlinearities. Moreover, other factors that contribute to the complexity of the dynamic satellite systems are the presence of uncertainty, actuator saturation, and the actuator faults itself. In general, the significant uncertainties are due to the changes of satellite’s inertia matrix as well as the external environmental disturbances. Frequently, the satellite’s inertia uncertainties are due to the measurement errors during the pre-launch testing, changes in the overall satellite system configuration, and fuel consumption during the mission (Tiwari et al., 2016). In most cases, the uncertainties and disturbances degrade the satellite attitude control performance and could cause mission failure. Besides, the attitude control system (ACS) needs to provide a high accuracy pointing and manoeuvring capabilities by the selected earth observation instruments and the space missions. Therefore, the attitude controller must be robust enough to cope either the uncertainties or external disturbances and the satellite’s attitude is supposed to be in controlled and have good attitude pointing performances for the entire mission.

The RW is chosen as an actuator in this study since it can provide high precision torque and high accuracy. The effectiveness of reaction wheels as satellite actuators is already well known. The famous Hubble Space Telescope (HST) and Midcourse Space Experiment (MSX) spacecraft have proven the capability of reaction wheels in controlling the spacecraft attitude. Despite their known advantages, these momentum exchange may suffers from the drawback of wheel momentum saturation (Yang, 2017). Thus, attitude control system is impractical without the momentum unloading control. Commonly, the MTQs are chosen as the secondary actuators to unload the momentum, and the challenge is to ensure continuous controllability for all three axis. Therefore, the optimum momentum unloading control technique is significant and needs to be designed according to the mission requirements.

Another issue deals with underactuated satellite attitude control which refers to satellite with less than three attitude control actuators. A set of four RWs in a pyramid configuration have been used in this study due to its controllability and redundancy reasons. If one of the RWs fails, the attitude control system can still generate any direction of torque by the remaining wheels. However, if two out of four RWs fail, external disturbances will cause the satellite to lose its ability to correct the attitude error. If the failure is irrecoverable, the satellite’s mission could be loss, as experienced by Hayabusa (Choi, 2005) and NASA Kepler spacecraft (Cowen, 2013). However, the three-axis attitude stabilisation is still can be achieved by using the two RWs left and with the assistance of specialised technique by which only a few have been investigated. Hence, this study proposed a control allocation technique to ensure controllability of the underactuated satellite specifically for the satellite with four RWs in a pyramid configuration.
1.3 Research Objectives

The objectives of this research are:

a. To develop robust satellite attitude controllers, namely the active force control (AFC) and the fractional order sliding mode control (FOSMC); and to design a Fuzzy-proportional integral (Fuzzy-PI) controller for the momentum unloading control of satellite reaction wheels.

b. To design a novel attitude control scheme for an underactuated satellite with two reaction wheels through the control allocation technique.

c. To validate the numerical testings and attitude control performances of all the developed satellite attitude architectures together with their governing equations both for nominal and underactuated satellite attitude controls.

1.4 Scope and Limitation of Studies

Several assumptions are set for this research. The satellite is assumed to be a rigid body actuated by four RWs in a pyramid configuration and employs three MTQs for wheel momentum unloading. The satellite is a small satellite that the mass is less than 100kg (microsatellite). The principal axes are aligned with the body axes. The satellite is built for Low Earth Orbit (LEO) and also for the attitude pointing mission. The controllers provided for the typical space missions during the Euler angles are relatively small. The attitude parameterisation via quaternion is employed in this work. The satellite’s moment of inertia matrix is known, and its value is constant for the entire duration of the mission. However, the inertia matrix and the external disturbance torques are considered to be uncertain, for instance, 10% of variation is set for the satellite’s inertia matrix (Tiwari et al., 2016). The satellite is assumed to be positioned at high inclination orbit because the Earth’s magnetic field strength at low inclinations is relatively weak. Thus, the use of MTQs in low orbit inclination is ineffective as only small magnetic control torques can be produced for wheel momentum unloading tasks. Since this study is considered for small satellites, it is essential to define the reasonable magnetic dipole moment saturation limits, especially in the simulation model. For small satellites, the range of the dipole saturation is from 1 Am² to 35 Am² (J. Lee et al., 2002).

1.5 Thesis Outline

In the first chapter, a brief description regarding small satellites and attitude control methods are introduced. Apart from that, the problem statement and the objectives of the research are also presented. Chapter 2 presents a summary of the literature that has been reviewed which includes the previous and current researches on the satellite
attitude control, in both the actuator and controller studies. It covers the implementation of different attitude control laws, the issues on the RW’s momentum unloading, the RWs configuration, and an underactuated satellite.

Chapter 3 details all the fundamental satellite theories used in this study such as coordinate systems, attitude representations, and angular velocity. The satellite attitude dynamics and kinematics equation are formulated. The RW’s control strategies and the wheel momentum unloading scheme are also presented in this chapter. Chapter 4 describes in details the enhanced control structure for the satellite attitude control with RWs.

The numerical simulations based on the proposed control strategy are presented in Chapter 5. The satellite attitude control and wheel momentum unloading performances for all the cases tested are presented and discussed as well. The conclusion is drawn in Chapter 6, and some suggestions are given for future researches.
REFERENCES


wheels with continuous momentum dumping. *AAS/AIAA Spaceflight Mechanics Meeting.*


BIODATA OF STUDENT

Zuliana Binti Ismail was born on 20 Oct. 1983 in Kuantan, Pahang, Malaysia. She received her Bachelor in Aerospace Engineering from Universiti Sains Malaysia, Penang in August 2006. Since then she worked as a research assistant at the Satellite Laboratory, Department of Aerospace Engineering Universiti Putra Malaysia, Selangor.

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LIST OF PUBLICATIONS


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