AN ASSESSMENT OF THE ORBITAL ELEMENTS OF RAZAKSAT FOR ATTITUDE DETERMINATION USING EXTENDED KALMAN FILTER

by

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To my beloved parents
Hasan Kunji Ahmad & Meriam Ahmad

My lovely wife
Munira Hanim Awang@ Abd Rahman

My beloved Children
Nur Fatimah Zahirah
Nur Aishah Humairah
Muhammad Solahuddin Zulkarnain

And last but not least
To all my family members

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LIST OF ABBREVIATIONS

AC	Attitude Control
AD	Attitude Determination
ADS	Attitude Determination System
ACS	Attitude Control System
ADCS	Attitude Determination and Control System
ATSB	Astronautics Technology Sdn Bhd.
c.g	Center of Gravity
DCM	Direction Cosine Matrix
DKF	Discrete Kalman Filter
ESOQ1	Estimator of The Optimal Quaternion-1
ESOQ2	Estimator of The Optimal Quaternion-2
ECI	Earth Centered Inertial
ECEF	Earth Centered Earth Fixed
EKF	Extended Kalman Filter
ESOQ	Estimator of Optimal Quaternion
ExEMF	Experimental Earth Magnetic Field Probe
ExPSS	Experimental Pyramidal Sun Sensor
FOAM	Fast Optimal Attitude Matrix
GPS	Global Positioning System
IGRF	International Geomagnetic Reference Field
LEO	Low Earth Orbit
LLA	Latitude Longitude Altitude
MAE	Mean Absolute Error
MEMS	Micro-Electro Mechanical Systems
NeQO	Near Equatorial Orbit
NORAD	North American Aerospace Defenses Command
OBC	On-Board Computer
OBDH	On-Board Data Handling
FOAM	Fast Optimal Attitude Matrix
GPS	Quaternion Estimator
RMSE	Root Mean Square Error
SEADS	Space Experimental Attitude Determination System

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STK	Satellite Tool Kit
SVD	Singular Value Decomposition
TLE	Two Line Element
UTC	Universal Time Coordinate

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LIST OF SYMBOLS

A	Attitude matrix
a_s	Semi major axis
B	Magnetic Field vector
$B_{n,m}$	Contribution of spherical harmonics of n and m degrees
C_f	Distance from the ellipse center to the ellipse focal point (Earth)
$\Delta(t)$	Time difference between the epoch and the last perigee passage before epoch
ε	Obliquity of the ecliptic plane
t_0	Epoch time
E_a	Magnetic Field vector
R_a^b	Rotation matrix from a to b
ω_{ab}^a	Angular velocity vector of frame relative to frame a, given in b frame
e_s	Eccentricity
$P_n^m(\theta)$	Schmidt semi-normalize associated Legendre functions of degree m and n
ω_e	ECEF frame will rotate the ECI frame with angular velocity
T_r	Rotation period of Earth
N_r	Number of periods since epoch
N_{rx}	Number of periods after pass the Greenwich
β_0	Satellite's dropping angle
ω_0	Satellite angular velocity in orbit
μ_g	Earth's gravitational constant
α	Rotation about Z-axis with angle
λ	Longitude
M_{Sun}	Mean anomaly of the Sun

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$\lambda_{ecliptic}$	Sun position presented in inertial frame transformed to orbit frame
$P_n^m(\theta)$	Schmidt semi-normalize Legendre functions of degree n and m
R	Distance from satellite to Earth centre
$M_{SunEpoch}$	Anomaly of epoch
Ω	Ascending node
$\dot{\Omega}$	Ascending mode rate
i	Orbit inclination

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Penilaian Elemen Orbit RazakSAT untuk Penentuan Sikap menggunakan Penambahan Kalman Filter

ABSTRAK

Malaysia telah berjaya membangun dan melancarkan RazakSAT, satelit mini pertama ke orbit berhampiran Khatulistiwa (NeqO). Misi satelit adalah sebagai demonstrasi teknologi dan salah satu beban bayarnya adalah sebuah kamera resolusi tinggi untuk menangkap imej di rantau khatulistiwa. Sistem Penentuan Sikap (ADS) RazakSAT adalah untuk memastikan orientasi adalah relatif terhadap Bumi.. Jika terdapat kekurangan penyelarasan atau gangguan pada ADS, ia boleh mempengaruhi orientasi satelit dari segi kedudukan orbit. Oleh itu, penyelidikan ini berfokus kepada penentuan atitud satelit untuk menentukan kedudukan orbit untuk tujuan keperluan kawalan. Walaubagaimanapun,menjalankan pengkajian di persekitaran ruang angkasa bagi menentukan ketepatan dan waktu pengiraan adalah mustahil disebabkan oleh kekangan kos dan fasiliti yang sukar diperolehi. Memandangkan data RazakSAT adalah orbit berdasarkan NEqO, maka dicadangkan ia digunakan sebagai rujukan kepada data sebenar untuk menggantikan ujikaji di persekitaran ruang.Ini bertujuan memberikan maklumat dan masa pengiraan yang tepat bagi sikap anggaran ADS. Dalam tesis ini, data RazakSAT digunakan sebagai rujukan untuk memberikan maklumat yang tepat dan masa pengiraan untuk menganggarkan sikap ADS. Model orbit Keplerian diimplementasikan sebagai model orbit dan membandingkannya dengan data NEqO RazakSAT.Selain itu, Perisian Kit Alat Satelit (STK) digunakan untuk mengesahkan elemen orbit RazakSAT seperti Bumi Berpusat Inertial (ECI), Bumi Berpusat Bumi Tetap (ECEF) dan Latitud Longitud Altitud (LLA) berdasarkan Elemen Dua Jalur (TLE) yang diperolehi daripada Astronautik Technology Sdn Bhd (ATSB). Vektor matahari dan medan magnet di dalam kerangka orbit dianalisa menggunakan model matahari dan model Bidang Rujukan Geomagnet Antarabangsa (IGRF). Vektor matahari dalam kerangka jasad adalah analisis penderia matahari dan magnetometer. Hasilnya ECI dan ECEF juga dianalisa menggunakan STK. Model orbit Kepler didapati lebih tepat berbanding dengan STK untuk LLA, dimana ralatnya adalah kurang berbanding dengan model orbit Kepler. Hasil keseluruhan bagi ECI, ECEF, dan LLA adalah kurang daripada 5% yang memenuhi keperluan ATSB untuk orbit RazakSAT. Untuk analisis model matahari, kerana sukar mendapat maklumat tentang penderia matahari RazakSAT daripada ATSB, analisa model matahari data menggunakan model matahari dari STK dan model matematik yang dibangunkan dalam tesis ini. Medan magnet dalam bingkai jasad dan bingkai orbit menunjukkan bahawa STK adalah lebih tepat dan kurang ralat pengukuran daripada model IGRF.Untuk penghitungan sikap satelit, pendekatan rekursif seperti mengetahui Extended Kalman Filter (EKF) digunakan untuk menganggarkan atitud. Dalam tesis ini, model kinematik dan dinamik untuk kaedah EKF diturunkan dan dianalisis dari segi kawalan, pemerhatian, dan kestabilan. Hasilnya membuktikan bahawa model dapat dikendalikan, diperhatikan, dan keadaan stabil margin akan digunakan untuk perkiraan dengan menggunakan EKF.

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An Assessment of the Orbital Elements of RazakSAT for Attitude Determination using Extended Kalman Filter

ABSTRACT

Malaysia has successfully developed and launched RazakSAT, the first mini satellite at the Near Equatorial Orbit (NeqO). The mission was focus on a technology demonstrator and one of the payloads is a high-resolution camera to capture images at the equatorial region. The purpose of RazakSAT's Attitude Determination System (ADS) is to ensure that the orientation is relative to Earth. Any misalignments or disturbances on the ADS can affect the orientation of the satellite in terms of the orbital position. Therefore, this research focuses on determining the attitude of a satellite in order to determine the orbital position for control requirement purposes. However, conducting experiments in the space environment to determine the accuracy and computation time is impossible due to cost constraints and the fact that facilities are scarce resources. As RazakSAT data is NEqO based orbits, so it proposed to use as reference to the actual data to substitute for the experiments in the space environment. The main aim is to give accurate information and computation time for the attitude estimation of ADS. In this thesis, RazakSAT data are used as a reference to give accurate information and computation time for the attitude estimation of ADS. Keplerian orbit model is implementing as an orbital model and compares with the NEqO of RazakSAT data. Besides that, Satellite Tools Kit (STK) software were used to conduct and validate the reliability analysis for orbital elements of RazakSAT such as the Earth-Centered Inertial (ECI), Earth Centered Earth Fixed (ECEF) and Latitude Longitude Altitude (LLA) based on the Two-Line Element (TLE) provided by Astronautic Technology Sdn Bhd (ATSB). The sun vector and magnetic field vector in the field in the orbit frame were analysed using the sun model, and International Geomagnetic Reference Field (IGRF). The sun vector in the body frame is the measurement from the sun sensor and magnetometer. The result on ECI and ECEF was analysed as well using the STK. The Keplerian orbit model was found to be more accurate than STK for LLA where the error yield is less compared to the Keplerian orbit model. The overall result for ECI, ECEF, and LLA is less than 5%, which also meets the error requirement of ATSB for RazakSAT orbit. For the sun model analysis, due to the difficulties in obtaining information about the RazakSAT sun sensor from ATSB, the analysis for the sun model utilized the results from STK and mathematical model developed in this thesis. The result shows that the percentage error for the sun in the orbit frame and body frame is acceptable, less than 5%. The magnetic field in the body frame and orbit frame from STK is more accurate compared to IGRF model. From the overall orbital results, RazakSAT was found are able to fulfill the requirement of the orbital specification while entering its orbit. For the attitude estimation of the satellite, a recursive approach known as an Extended Kalman Filter (EKF) is used to estimate the attitude. In this thesis, the kinematic and dynamics models for the EKF method are derived and analyzed in terms of controllability, observability, and stability. The result proved that the model could be controllable, observable, and marginally stable condition where it can be used for estimation by using EKF.

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CHAPTER 1 : INTRODUCTION

1.1 Project Background

By definition, a satellite is any object knows as a spacecraft which orbits a planet. Satellites can be natural ones, such as the moon orbiting the Earth or the Earth orbiting the Sun. They can also be artificial, like the International Space Station (ISS) orbiting the Earth. For this thesis, we shall be making reference to the artificial type. Celestial objects such as the moon orbiting a planet in the solar system or human-made ones typically launched into outer space from Earth to collect data and images are also types of satellites. For artificial satellites, they can be categorized on the basis of their mass as shown in Table 1.1.

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Table 1.1 : Categorization of satellite respect with to mass (Narayanan,2008;Sharun,2012)

Class	Mass
Large satellite	> 1000 kg
Small satellite	500-1000 kg
Mini-satellite	100-500 kg
Micro-satellite	50-100 kg
Nano-satellite	< 10 kg
Pico-satellite	< 2 kg

Today, there is heavy dependence on satellites to assist man’s daily tasks. The classification of a human-made satellite depends on its weight and height as it goes around the earth (based on the earth). A Low Earth Orbit (LEO) orbits the Earth between the heights of 400 to 1000 km. High Earth Orbit (HEO) on the other hand is located 10,000 to as far as 35,791.81 km from Earth and it is ideal for navigation while Medium Earth Orbit (MEO) represents the distances in between high and low Earth orbit(Martinelli & Sánchez Peña, 2005;Sharun,2012;Nakasuka et al., 2018).

A satellite consists of various subsystems that are brought together so that the satellite can be operated according to its mission. There are four important satellite subsystems, i.e. the attitude determination and control system (ADCS), telemetry, tracking and command (TT&C), command and data handling (C&DH) and electrical power subsystem (EPS). There is a special task for every subsystem to ensure that the satellite is maintained while orbiting, i.e.:

i. The attitude determination and control system (ADCS) serves to determine the satellite's current position and attitude, and control by having it stabilized or oriented into the desired direction during the mission, while taking into accounts the action of the external disturbance torque.

ii. The telemetry, tracking, and command (TT&C) provides the communication line between the spacecraft and ground station systems that receive, process, and transmit data signals' downlink and uplink to meet the requirements of the mission.

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iii. The command and data handling(C&DH) functions as the ‘brain’ of the satellite which oversees the system’s overall operating system using an on-board computer (OBC).

During the mission, the electrical power subsystem (EPS) provides stores, distributes, and controls the satellite electrical power needed to operate all electronics devices of a satellite.

In satellite systems, as part of a satellite sub-system, the Attitude Determination Control System (ADCS) carries a significant function that is to stabilize the satellite systems in the mission from start to finish. The ADCS is divided into the Attitude Control System (ACS) which acts to stabilize the attitude, and the Attitude Determination System (ADS) that measures the vector of inertial reference and computes the satellite’s orientation and position relative to Earth. To be references to correct during on orbit, the measurement vectors of the inertial reference and computation orientation of the ACS are as shown in Figure 1.1.

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To meet its purpose, the ADCS is significant in helping the satellite to sustain in its orbit. The ADS consists of an attitude sensor and algorithm, with the task to ascertain the satellite’s orientation with regard to the reference frame. To produce fast and appropriate control for the ACS, it would be essential to have the accurate output for the correction measurement from the ADS. The attitude of satellites would be very important to be measured using the ADS sensor to get an accurate estimation attitude. The components of the ADS sensor are important so that the orientation of near-equatorial orbit, relative to other satellites can be acknowledged. The satellite attitudes are expressed by the attitude position and angular velocity in the satellite body frame relative to the local coordinate orbit frame. Attitude estimation can be determined using either the deterministic method or by recursive method. The deterministic method applies when attitude is found based on two or more vector observations from a single point at a time. The recursive method adopts the kinematics and/or dynamics model, and the prediction of the attitude of satellite in time series of calculation from observation of one or more vectors can be done (K.F.Jensen, 2010;Hasan et al., 2016).

There are two sides to the ADS sensor- a reference's sensor or an inertial references sensor. References sensor comprises of sun sensor, magnetometer, star tracker, and horizon scanner provided via vector observation. A reference sensor usually consists of noisy vector measurement at low frequency, and they usually need two or more sensors to overcome the issue of unobserved ability. An inertial sensor such as gyroscope can improve the measurement as it gives an angular rate relative to the inertial frame. That said, the sensor is limited following the noise and bias error in measurement during disturbances and that it cannot work alone during unbounded error in the attitude estimation over time. Therefore, in terms of the noise present in a measurement, the attitude estimation algorithm is needed to make ready for the best estimate of the attitude. The attitude estimation from ADS can become the feedback into the ACS for the control algorithm which will send the desired command to the actuators according to the satellite attitude desired (Wertz, 1978; Sidi, 1997; Sellers,2003; Shahrun, 2012).

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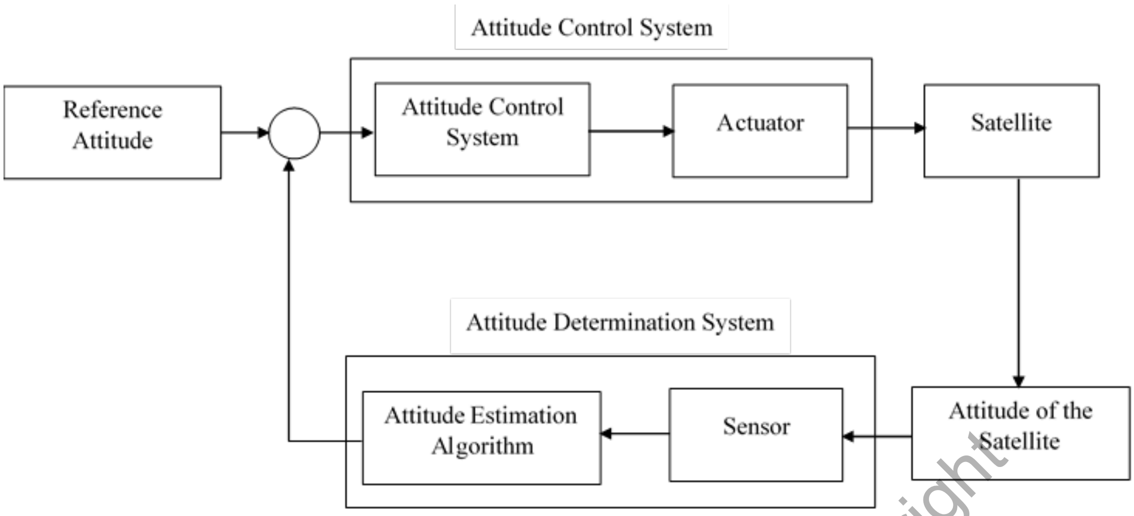

Figure 1.1 : Block Diagram of an ADCS

1.2 Problem Statements

ADS accuracy is restrained by noise, bias, and misalignment in measurement to provide the right attitude estimation (Adam, 2013). Attitude determination commonly comprises of a certain volume of error as the ADS sensor is used for prediction(Appel, 2005). The disturbances from the solar radiation pressure, aerodynamic forces, magnetic field, gravity gradient and force from natural phenomena in space can foresee the misaligned measurement and real measurement of ADS sensor as shown in Figure 1.2.

The misalignments that come from ADS disturbance can affect the satellite orientation with regard to the orbital position. Therefore, the aim of this work is to focus on the attitude of a satellite to find out more about the orbital position for control requirement purposes. To determine the attitude of a satellite, satellite orbit data is required to look closely into the orbital position satellite as it enters the orbit using the Keplerian orbit model. Unlike the polar orbit, near-polar or sun-synchronous orbits,

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NEqO orbit exposes the satellite to the South Atlantic Anomaly (SAA) phenomenon on every orbit it takes around the earth, automatically increasing the risk of radiation damage to the satellite (Adam et al., 2009; Suparta and Zulkeple, 2014). Thus, by having the real data of a satellite RazakSAT, it is in NEqO orbit such as the satellite orbital position. To compare, the Satellite Tools Kit (STK) suite of software can be applied to carry out the orbit simulation. Nevertheless, satellite orbital data NEqO orbited regions are scarce (Hasan et al., 2015).

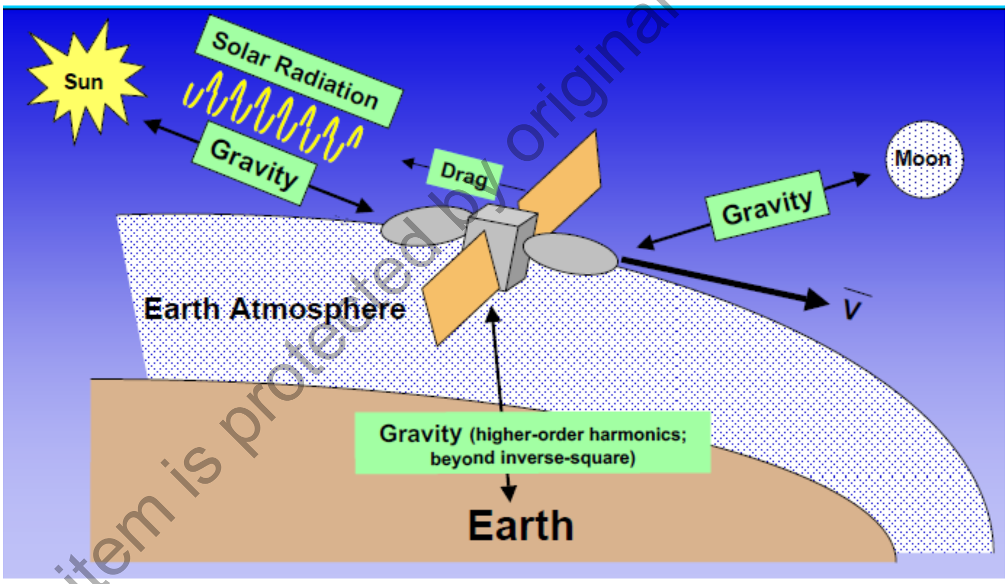

Figure 1.2 : Disturbances Forces Effecting Orbit (Howley, 2018)

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A lot of researchers have dedicated years of study proposing to improve the attitude estimation algorithm (Fadly, 2010; Fadly, 2016; Cilden et al., 2017; Soken, 2018). The attitude estimation using the recursive method is superior to the deterministic method. The deterministic method does not have any of the two measurement vectors that yield inconsistent data. Past literature has shown that the commonly used estimation algorithm in the recursive method is the Extended Kalman Filter (EKF) which gives the fastest algorithm, and the most accurate one too, despite