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Assessment of the Orbital Elements of RazakSAT using the Keplerian Orbit Model and STK

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ABSTRACT

Attitude determination system (ADS) is one of the satellite sub-systems that functions to measure the vector of inertial reference and compute the orientation of the satellite relative to the Earth. However, any misalignments or disturbances on the ADS can affect the orientation of the satellite in terms of the orbital position. This paper focuses on determining the attitude of a satellite in order to determine the orbital position by using RazakSAT data in NEqO orbit. The Keplerian orbit model and Satellite Tools Kit (STK) software is implemented as an orbital model and compared with the NEqO of the RazakSAT data. The results show STK is more accurate. The results show that the STK software method outperforms the Keplerian orbit model in terms of determining the satellite's orbital position.

Keywords:
Satellite tools kit; keplerian orbit; Razak Sat

1. Introduction

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 consists of 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 as shown in Figure 1 [1,2].

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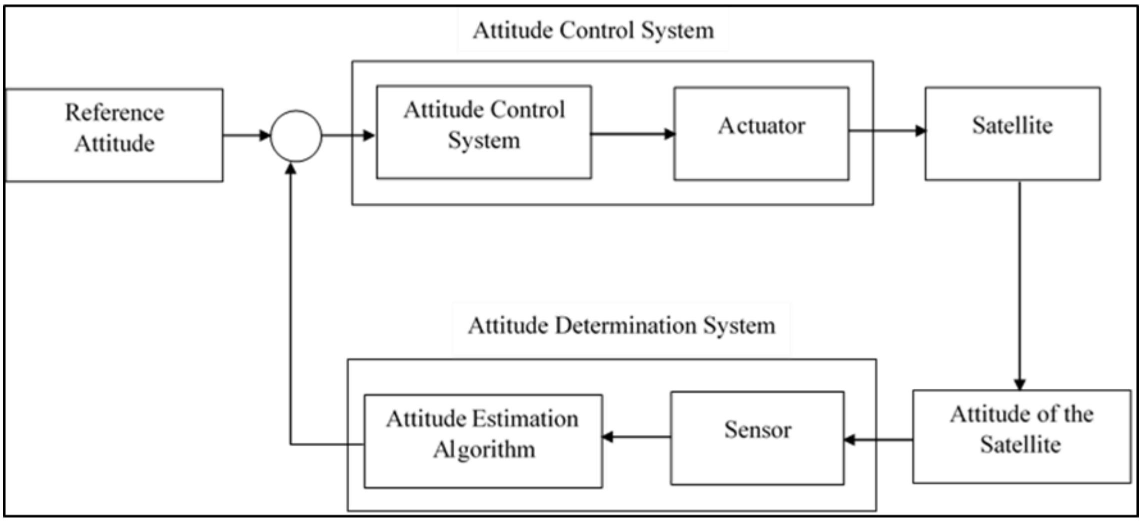


Fig. 1. Block diagram of an ADCS

The ADCS is significant in helping the satellite to sustain its orbit. The ADS consists of an attitude sensor and algorithm, with the task of ascertaining the satellite's orientation regarding the reference frame. ADS accuracy is restrained by noise, bias, and misalignment in measurement to provide the right attitude estimation [3]. Attitude determination commonly comprises of a certain volume of error as the ADS sensor is used for prediction [4]. The disturbances from the solar radiation pressure, aerodynamic forces, magnetic field, gravity gradient and force from the natural phenomena in space can foresee the misaligned measurement and real measurement of the ADS sensor as shown in Figure 2. The misalignments that come from the ADS disturbance can affect the satellite orientation regarding the orbital position in terms of Earth-Centered Inertial (ECI), Earth Centered Earth Fixed (ECEF) and Latitude Longitude Altitude (LLA). Therefore, this research focuses on determining the attitude of a satellite in order to determine the orbital position by using the RazakSAT data in NEqo orbit.

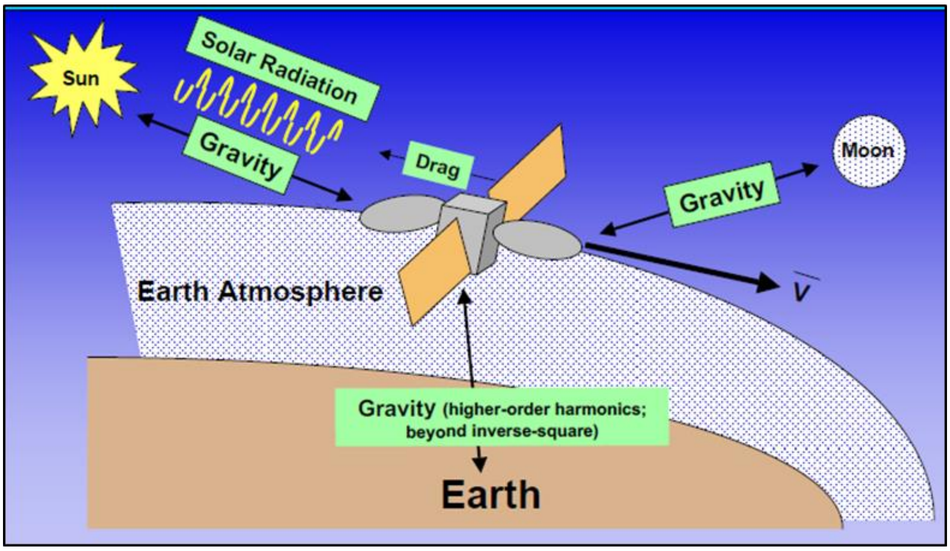


Fig. 2. Disturbances forces effecting orbit [11]

Unlike the polar orbit, near-polar or sun-synchronous orbits, 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 [5,6]. Thus, by having the real data of a satellite RazakSAT in NEqO orbit located at 9^0 inclination and 685 km at nominal altitude on NEqO from the orbit region, this offers an opportunity to researchers to perform an analysis of the satellite orientation in the NEqO orbit [7,8]. Nevertheless, the satellite orbital data NEqO orbited regions are scarce [9,10]. Based on these findings, research gap is identified for this work:

i. Satellite data needs is needs to investigate the orbital position satellite during it enters in the NEqO orbit by using orbit model.
ii. Verifications the satellite orbital data with satellite analysis approach and satellite modelling for ECI, ECEF and LLA.

In this paper, RazakSAT data are used as reference to give accurate information and computation time for the attitude estimation of ADS. Keplerian orbit model and Satellite Tools Kit (STK) software is introduced as an orbital model and compared with the NEqO of RazakSAT data. Besides that, the Keplerian orbit model and STK software were used to conduct and validate the reliability analysis for the orbital elements of RazakSAT such as the ECI, ECEF and LLA based on the Two-Line Element (TLE) provided by Astronautic Technology Sdn. Bhd. (ATSB).

2. Near Equatorial Orbit

The general characteristics of Low Earth Orbits (LEOs) have been meticulously studied by a variety of researchers, and extensive literature has been published by various organizations, regarding LEOs. At the onset of the first satellite, Sputnik-1, LEOs served for multiple missions. Quite surprisingly, LEOs and NEqOs offer specific mission-design characteristics that come with better coverage and contact pattern (with respect to the frequency and duration of coverage and contact regarding the regions at, or near, the equatorial belt) so that much better organized space borne systems can be applied (regarding highly inclined orbits) [12,13].

This is significant, based on a few considerations:

i. More frequent passes in each period translate into more communication time. More frequent passes during a given period of time translate into more communication time i.e. downlink/uplink of mission-specific data and Tracking, Telemetry and Command (TT&C) data. Following the short communication time during each pass for Low Earth Orbits (LEOs), the downlink of the missions-specific data from the satellite to the ground station(s) has always been an issue daunting enough for such missions.
ii. Next, more frequent passes translate into shorter revisit times. Revisiting time that is done adequately, in turn, is very important to the earth-orbiting missions in various ways and this has been studied extensively in mission-design studies. For humid equatorial regions, this condition is crucial, as these areas are often cloud- covered and the probability of obtaining cloud-free scenes is deemed to be low, from a mission-design standpoint. Thus, for a specific region at or near the equator, there should be more frequent passes to make sure that there are high enough probabilities for cloud-free scenes to be obtained in a specific time .
iii. Using remotely sensed data from the NEqO satellite system has many advantages especially in equatorial countries, all the equatorial countries located in tropical areas that are cloudy, and thus, have high humidity, enabling different kinds of risks to be performed [14].

RazakSAT mission is generally aimed to provide a greatly increased coverage of Malaysia, unlike other Earth observation satellites available. However, a standard normal sun- synchronous satellite suitable for weather forecast and spy due to its consistent lighting, is believed to only visit an area once every seven days, which renders it inefficient as the cloud covers may disrupt certain views of the area. To overcome this, RazakSAT set as a non-Sun synchronous satellite can revisit some parts of the Malaysian territory every 90 minutes, which potentially maximises its ability to exploit the cloud gaps [15].

Three distinct disadvantages of NEqO are as follows:

i. NEqO orbit discloses the satellite to the South Atlantic Anomaly (SAA) phenomenon on every orbit around the earth unlike the polar orbit, near-polar or sun-synchronous orbits, further heightening the risk of radiation damage to the satellite.
ii. LEO region is known to be easily exposed to trapped particles. A remarkable source of trapped particles is from a region called the SAA off the coast of Brazil, where the inner Van Allen Belt dip is closest to the Earth's surface. It is a region full of highly charged protons and electrons and is also responsible for the discharge effects in electronic devices that certainly affect a satellite's communication control and operation.
iii. NEqO image is different because it has nonlinear distortion. This distortion leads to the extraction of false features and incorrect image matching [16].

2.1 Keplerian Orbit Model

This orbit model is based on Kepler's equation to get an elliptical orbit. To understand the formula derivation in the orbit model, the important feature of the orbit is presented by Wertz *et al.,* [18]. These important features are described regarding the ellipse, the location of the satellite on the ellipse at a certain time and the orientation of the Earth. The prediction of the upcoming location can be relatively direct to the point because the features of the Keplerian are decided for a fixed time [17,18]. The mean anomaly M propagates with time uniformly, so

$$M = M_0 + n(t_0 - t) \tag{1}$$

where,

M_o: mean anomaly at epoch time.
t: current time.
t_o : epoch time.

Then the eccentric anomaly, E can be determined through the settlement of Kepler's formula. This equation is related to the mean anomaly M at the same time to the eccentric anomaly E, at the same time by,

$$M = E - EsinE \tag{2}$$

where,

$$E = 2.0\,X\,10^{-7}$$

Given that M obtaining E from Kepler's equation necessitates a numerical approximation, i.e. Newton's method. In this method, a Taylor sequences development of the function $f(E) = E - esinE - M$ is used. Successive estimates for E are given by,

$$E = E_0 \frac{M + esinE_0 - M}{1 - ecosE_0} \tag{3}$$

where, the starting value guess for E_0 is the mean anomaly M, a priori to the estimation. The next step is to include the expected perturbations of the satellite orbit, which can govern the Keplerian elements. The perturbations considered in this orbit model are the perturbation following the Earth and the third-body perturbations. Considering the Lagrange earthly equivalences [19], the flattening influence J^2 result with the upcoming time derivatives of the right ascension of the ascending node Ω and the argument of perigee ϖ,

$$\dot{\Omega}_{J_2} = \frac{3}{2} n J_2 \left(\frac{R_e}{a}\right) \frac{\cos i}{(1 - e^2)} \tag{4}$$

$$\dot{\omega}_{J_2} = \frac{3}{4} n J_2 (\frac{R_e}{a})^2 \frac{5cos^2 i - 1}{(1 - e^2)} \tag{5}$$

where,

$\boldsymbol{a}$: The Earth's radius
$\boldsymbol{J_2}$: $1.08284 \text{x} 10^{-3}$
i : 6378.14 km

The Sun and the moon stimulate periodic changes in all Keplerian elements, but secular perturbations (the satellite in a low orbit experiences the largest orbital perturbations due to the atmospheric drag and non-spherical gravity field), only apply to the right ascension of the ascending node Ω and the perigee argument $1.08284 \text{ x } 10^{-3}$.

$$\dot{\Omega}_{sun} = -\dot{0}.00154 \frac{\cos \iota}{n} \tag{6}$$

$$\dot{\omega}_{sun} = 0.00077 \frac{5 \, cosi^2 - 1}{n} \tag{7}$$

$$\dot{\Omega}_{moon} = -0.00338 \frac{\cos i}{n} \tag{8}$$

$$\dot{\omega}_{moon} = -0.00169 \frac{5 \, cosi^2 - 1}{n} \tag{9}$$

From the results stated, the position vector of the satellite in the Earth-Centered Orbit (ECO) frame is established,

$$r_{ECO} = a \begin{bmatrix} \cos E - e \\ \sqrt{1 - e^2 sinE} \\ 0 \end{bmatrix} \tag{10}$$

For the computation of the position vector in the ECI and ECEF frames, the appropriate rotation matrixes Eq. (11) and Eq. (12) should be used in both the equations below,

$$r_{ECI} = R_z(-(\Omega + \dot{\Omega}_{J_2} + \dot{\Omega}_{moon} + \dot{\Omega}_{sun})t))R_x(-i)R_z(-(\omega + \dot{\omega}_{J_2} + \dot{\omega}_{moon} + \dot{\omega}_{sun})t)r_{ECO} \quad (11)$$

$$r_{ECEF} = R_z(-(\Omega + \dot{\Omega}_{J_2} + \dot{\Omega}_{moon} + \dot{\Omega}_{sun})t + \theta))R_x(-i)R_z(-(\omega + \dot{\omega}_{J_2} + \dot{\omega}_{moon} + \dot{\omega}_{sun})t)r_{ECO} \quad (12)$$

where,

R_Z : Rotation matrix on Z-axis
R_X : Rotation matrix on X-axis
a : Earth's radius
E : Eccentric anomaly
e : Eccentric of orbit
Ω : Ascending nod
$\dot{\Omega}$: Ascending mode rate
i : Orbit inclination
θ : Different angles between ECI and ECEF

2.2 Keplerian

Keplerian element parameters serve to compute the gravitational waves as the complex structures in the stellar systems. The gravitational waveform from the orbits of Keplerian parameters such as elliptical, circular, parabolic, and hyperbolic undergoes an evaluation using a computational approach [20]. Keplerian orbit can describe a technique of asteroid deflection with the aid of the spacecraft. There is proof that in general, the spacecraft can exert enormous force on the asteroid in comparison to the stationary gravity tractor, thus lowering the time taken to take effect on the asteroid's targeted deflection [21]. Four parameterized Kepler elements namely true anomy, eccentricity, the argument of perigee and semi-major axis are used to construct, and to materialise the transition and application of the resonant orbits for the polygonal-like periodic orbit (PLPO) [22]. Keplerian orbit has been helpful for the second satellites to estimate the volumetric estimation for the satellite encounter rates. The encounter volume is represented by an ellipsoid which has a consistent shape, size, and orientation in reference to the satellite's Radial-In Track- Cross Track frame. Through the means of computing, if or when the first satellite's orbit traverses it; it is estimated as proximate when it is inside the volume. The tool can help study the regions as graveyard orbits which are least likely to result in encounters. Moreover, the tool can be utilized to examine how frequent that the neighbouring satellite will trigger a volumetric warning when a candidate orbit is brought to comparison [23]. The Kepler elements are used for the orbit to check the observability estimation and autonomous navigation with two satellites due to their corresponding coordinate value. With the unperturbed Kepler orbit dynamics and basic requirement of k-order, local weak observability is computationally justified for the autonomous navigation system. Certain cases demonstrate that the dual satellite system can lead to a decreasing sense of observability and result in lower navigation accuracy. A new concept known as the k-order local weak observability is put forth and applied to examine the autonomous navigation system that employs the relative position measurements [24]. Predictive Bayesian statistics is the data from the Keplerian investigation for a new solar system, and the computation of the possible number of stars in the extent of the universe in making certain of the likelihood of life beyond earth. Predictive Bayesian statistical techniques are computed to apply for limited, unpredictable data, for results improvement. The result establishes a probability curve representing the probability of life in the universe consisting of both unpredictable and possible variability bounded by the result to define life probability in the galaxy and also life beyond the proximity to earth [25]. With the first-order time-explicit computational solution, the relative motions that range with various inter-satellite limits were examined in Keplerian orbit, and few typical trajectories of relative motions were given to adjust to the distributed space system design. When the orbital perturbations are reduced as opposed to the two-body gravitation, the trajectory shape of the relative motion range could be computed in Kepler orbit so that the starting orbital elements of chief and deputy can be discovered by Chao *et al.,* [26]. The impact of the improvements of Global Navigation Satellite System (GNSS) infrastructure on the Global terrestrial references frame (TRF) with the focus on the Earth rotation parameters is assessed by the simulation studied [26,27]. The results show that Kepler improved the pole coordinate estimate interferometry.

2.3 ECI Frame

ECI is a coordinate system with the coordinate centre found in the Earth's centre. The X-axis is on the vernal equinox (the cross point between the Earth orbit and the equator line) whereas the Z-axis is the position to the North Pole of the Y-axis. This frame is denoted as I and this is as shown in Figure 3.

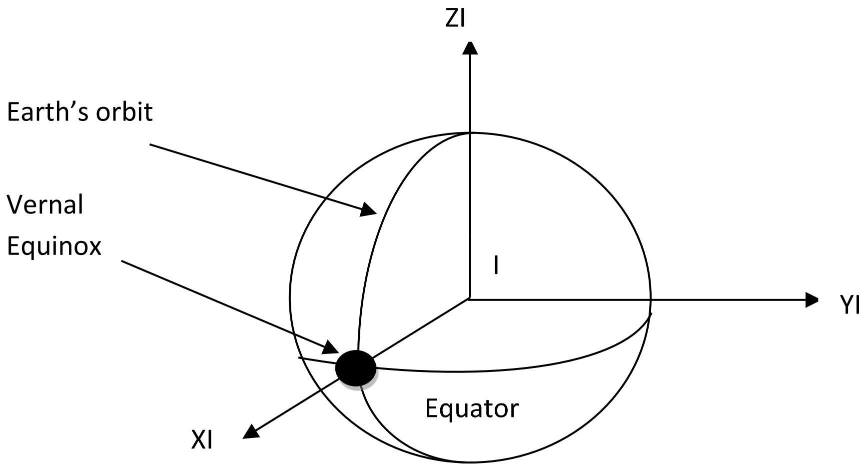


Fig. 3. ECI Frame [28]

2.4 ECEF Frame

Relative to ECI, ECEF is a coordinate frame that rotates together with Earth rotation. The X- axis is on the cross point between Greenwich meridian and equator line. The point is assigned as 0^0 longitude and latitude. The Z-axis is the position to the North Pole, and Y-axis is complementary to the right hand. The ECEF frame rotates the ECI frame with an angular velocity $\omega_e = 7.2921$ x 10^{-5} rad/s. The frame is denoted as E in Figure 4.

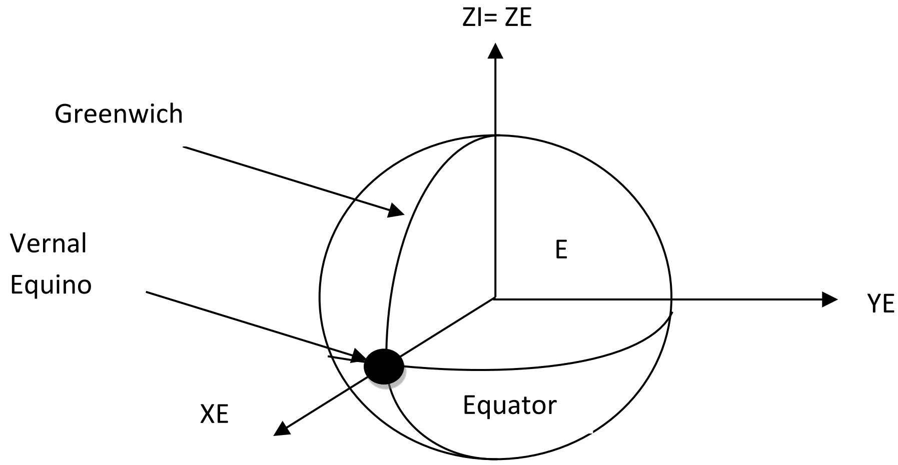


Fig. 4. ECEF Frame [28]

2.5 LLA

The spherical and rectangular celestial elliptic systems are shown in Figure 5. The fundamental circle and plane are those defined by the ecliptic, and the origin O is usually cantered in the Sun. The position of a point satellite in space is defined by its radial distance *A*, ecliptic longitude l, and ecliptic latitude *L*. The angle is measured eastward from ¡ around the ecliptic from $0^0$to 360^0. The angle *L* is measured from the ecliptic plane to $+90^0$ at the north ecliptic pole (NEP) or south ecliptic pole (SEP).

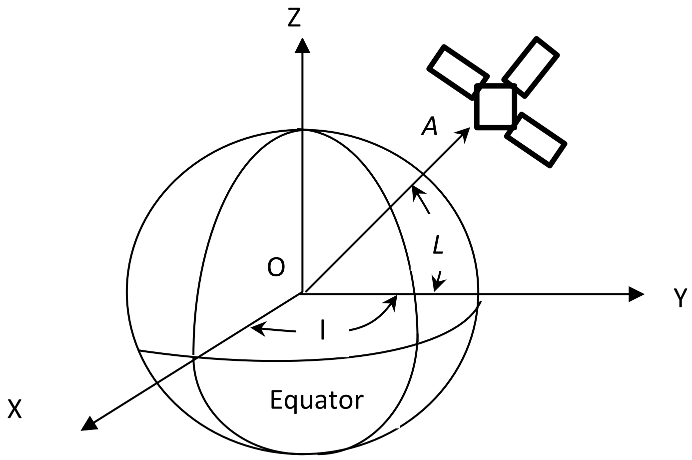


Fig. 5. Celestial ecliptic coordinate system [29]

2.6 Satellite Tools Kit (STK)

Satellite Tools Kit (STK) is software that simulates vehicle navigation and communication. This is great news to satellite developers, since rigorous simulation can be done before spending millions on sending a non-tested satellite in space. STK can well design and develop complex and dynamic simulations for real problems that have to do with the ground vehicles, aircraft, spacecraft, and satellite. Free access 2D and 3D modelling as shown in Figure 6 has made it easy and convenient for professionals or designers to create a complex model system and assess the targeted system performance [30-32]. In this research, STK 10.1 is used to simulate the attitude position of ECI and velocity of RazakSAT using the TLE data provided by ATSB.

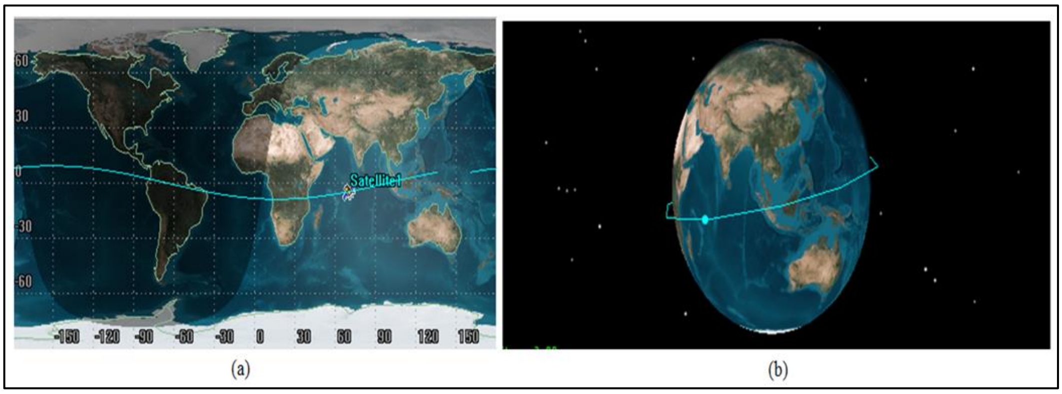


Fig. 6. STK 10.1 software (a) 2D graphic view (b) 3D graphic view

3. Methodology

3.1 Process of Orbit Model Kepler

Orbit data is the information of orbit that has the inclination, orbit form, altitude, and TLE. From orbit data, the satellite position vector can be calculated using the orbit model. With the application of the Kepler model, the position vector is calculated for this thesis. The process flow of the orbit model Kepler model is shown in Figure 7, and the position vector is presented in ECI and ECEF. The vector should come in handy for other environmental models to calculate the objects vector which concerns the satellite. From Figure 7, the position vector calculation is started by way of extracting TLE to get the satellite epoch time (the initial time while the satellite enters the orbit in Julian date), along the time the following location can be calculated directly through the Kepler model. The orbit model's output is the satellite position vector in ECI. From here, the ECI vector can be transformed into ECEF and LLA.

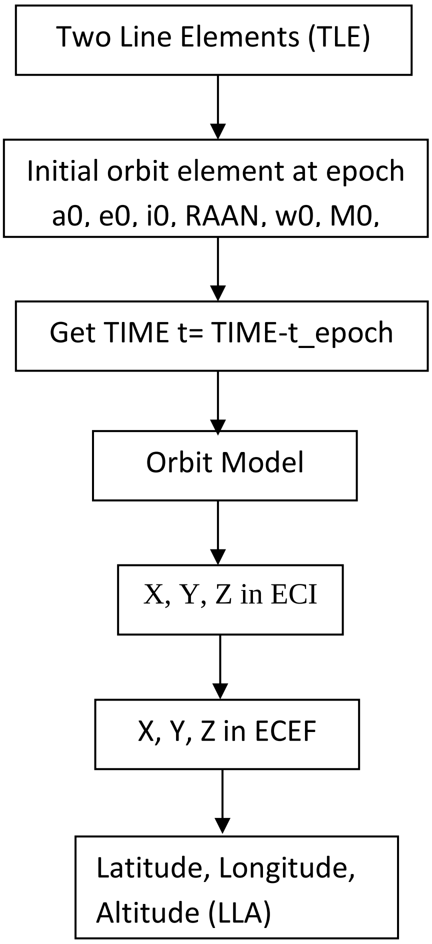


Fig. 7. Process of orbit model Kepler

3.2 Transformation from ECI to ECEF Frames

The rotation of the ECEF frame relative to the ECI frame is a rotation around the Z-axis where ZI and ZE are both a coincidence. This rotation can be decided by a simple rotation surrounding the Z-axis with angle $\alpha = \omega_e t$, where ω_e the Earth's rotation velocity is and t is the time passed since ECEF, and ECI frame is aligned [33]. Due to the fact that the rotation α is the negative right-handed, the rotation matrix from ECEF to ECI is given below,

$$R_e^i = R_e^i(-\alpha) = \begin{bmatrix} \cos(-\alpha) & \sin(-\alpha) & 0 \\ -\sin(\alpha) & \cos(-\alpha) & 0 \\ 0 & 0 & 1 \end{bmatrix} = \begin{bmatrix} \cos(\alpha) & -\sin(\alpha) & 0 \\ \sin(\alpha) & \cos(\alpha) & 0 \\ 0 & 0 & 1 \end{bmatrix} \quad (13)$$

$$R_e^i = (R_e^i)^T = \begin{bmatrix} \cos(\alpha) & \sin(\alpha) & 0 \\ \sin(\alpha) & \cos(\alpha) & 0 \\ 0 & 0 & 1 \end{bmatrix} \quad (14)$$

The term *e* is the exact coincidence time between the ECEF and ECI frame according to Astrodynamical Almanac, occurred on December 31st, 1996, 17h 18m 21.82565s. This time is translated to a Julian date to $f_s = 2453096.2018$ [34]. It yields a formula to calculate the rotation matrix between ECI and ECEF which has a rotation angle between the vernal equinox and Greenwich which is α ,

$$T_r = \frac{2\pi}{\omega_e} \tag{15}$$

Where T_r is a rotation period of Earth.

$$N_r = \frac{(J_{Dnow} - J_{Dfs}).86400}{T_r} \tag{16}$$

$$N_{rx} = N_r - floor(N_r) \tag{17}$$

$$\alpha = \omega_e(N_{rx}.T_r) \tag{18}$$

Here, N_r is a number of periods since epoch and N_{rx} is a number of the period after passing the Greenwich.

3.3 Transformation from ECEF Frames to LLA

According from Figure 5, the longitude of the satellite can be directly determined by

$$l = a\tan\left(\frac{y}{x}\right) = a\tan 2\,(y, x) \tag{19}$$

where atan2(y,x) is the four-quadrant inverse tangent, which returns the arctangent of y/x in the range $-\pi$ to π rad.

The physical radius of the point of satellite and the radius in the x-y plane are computed and used in an initial estimate of the altitude.

$$r = \sqrt{x^2 + y^2 + z^2} \tag{20}$$

$$p = \sqrt{x^2 + y^2} \tag{21}$$

The geocentric latitude and altitude are computed exactly and used as the initial value for the geodetic latitude in the iteration loop.

$$L = a\tan\left(\frac{p}{z}\right) = a\tan 2(p, z) \tag{22}$$

$$A = \frac{p}{\cos(L)} - \frac{R_N}{\sqrt{1 - e^2 \sin(L)}} \tag{23}$$

3.4 Implementation and Simulation

Both the implementation and simulation are performed to determine and enhance the parameters of the Keplerian orbit model as an orbital model with the actual environment characteristics. The attitude determination methods are applied with the Razak SAT’s actual environment and attitude data. The environment model and the attitude of the satellite are required by the controller to give control action to ensure the stability of the satellite. It begins with the collection of the physical characteristics of RazakSAT and TLE data. The process begins by calculating the satellite position vector using the Keplerian model, which, in turn, will produce a satellite position regarding ECI, ECEF, and LLA. The process is continued for the environment data produced from the satellite simulation software Satellite Tool Kit (STK) by AGITM. The STK will produce data containing the satellite position, sun and magnetic field vector, and the satellite attitude. The position vector from the model and STK is verified with the help of the RazakSAT position vector data. It is then used to calculate the sun and magnetic field vector using the sun and magnetic field model. The model and STK are verified using the vector from the RazakSAT data. If it is verified, the process continues to calculate attitude determination (AD) using the EKF method. Then, the attitude from STK and AD is examined and confirmed. If the attitude from AD methods is accepted, then the stimulation of the attitude determination output is in order. The orbit model will produce a satellite position vector in orbit (in ECI and ECEF frames). TLE is the initial information of the satellite orbit and a basic data used for calculating the position vector in orbit [35]. The data is given by NORAD containing the initial position, number orbit rotation and other important pieces of information about the satellite while going round the Earth. The raw data of RazakSAT TLE is supplied by ATSB, consisting of various pieces of orbit information listed in Table 1. When RazakSAT enters the orbit on July 14th 2009, the orbit model calculates the position vector by loading some values (as seen in Table 1 and Table 2) into the orbit equation. To get the vector in ECI frame and ECEF frame, the Equation calculates the position vector start by using Eq. (10) to Eq. (11) from the Keplerian orbit model. The values in Table 1, Table 2, date, and time of RazakSAT are important items in running the STK simulation. Normally, STK will produce the vector of the satellite position, sun, and magnetic field as well as the attitude of RazakSAT in quaternion and Euler. RazakSAT will orbit the Earth a few times in one day based on the TLE data provided in Table 1. To simplify the models' verification and simulation, the simulation data of STK is run only for two orbits. The transformation from ECI to ECEF frame and ECEF frame to LLA uses Equation part 3.2 and 3.3.

Table 1
Explanation of TLE data

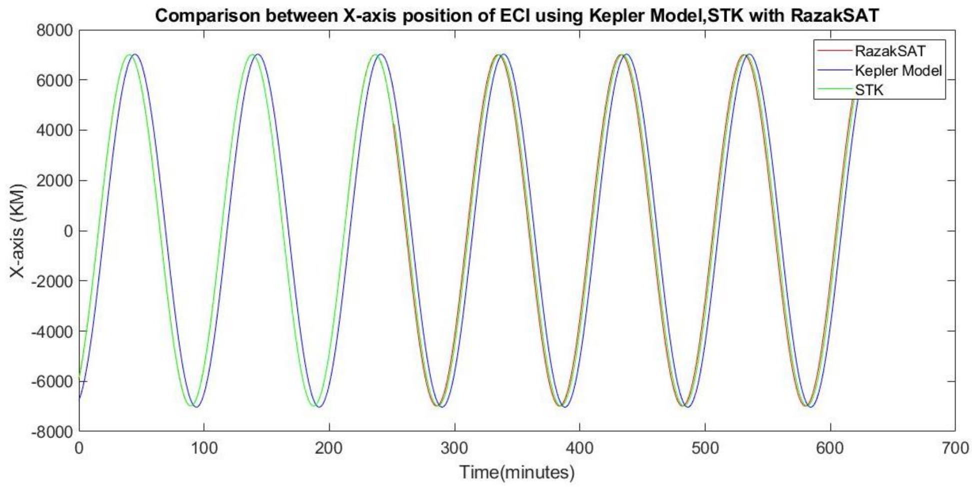
Elements	Parameters
Satellite Number	35578
Epoch year	09
First derivative of Mean Motion	00000748
Second derivative of Mean Motion	00000-0
BSTAR drag term	34706-4
Ephemeris Type	0
Element Number	316
Check sum 1	7
Inclination	008.9895
Right ascension of ascending node	213.5248
Eccentricity	0017552
Mean anomaly	323.5560
Mean motion	14.66093039
Revolution number of epoch	26305
Check sum 2	0

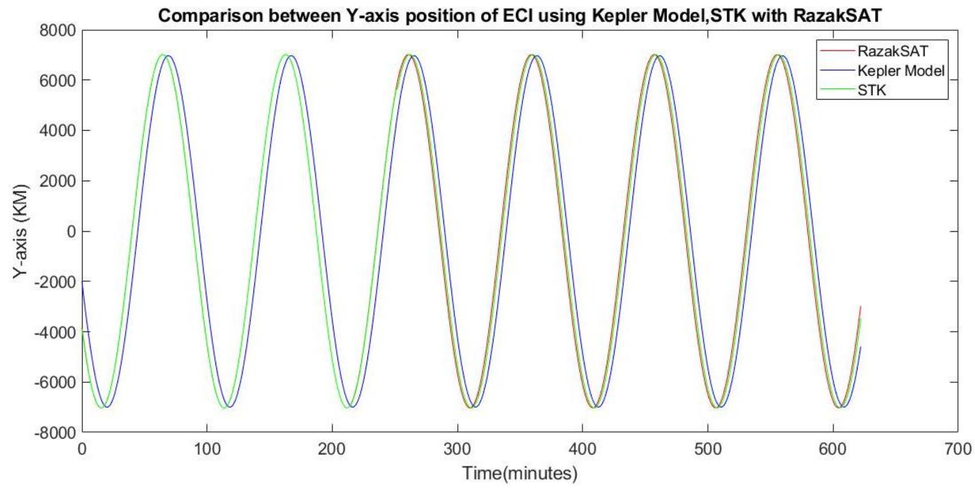
Table 2
Constant definition [36]

Elements	Parameters
G (Gravitational constant)	6.6720e^{-11}m^3/(Kgs^2)
M_e(Mass of the earth)	5.9742e^{24}Kg
R_e(Radiu of the earth)	6.378137e^{6}m
u_g(Earth gravitational constant)	3.9860e^{14}m^3/s^2
J_{2000}(Julian date at 1^{st} Jan 2000-12:00:00 UTC)	2451545
J_{fs}(Julian date at 31^{st} Dec 1996-17:18:21.8256 UTC)	2450449.22108594
θ_0(Ascension on greenwich at 1^{st} Jan 2000:00:00:00 UTC)	99.96779469
ω_e(Earth angular velocity)	0.004178 deg/s
ω_0(Satellite angular velocity)	0.001064 deg/s
B_0(Initial magnetic field vector)	$[2.592\mathrm{e}^{-5}, -5.973\mathrm{e}^{-5}, 0.00019]^T$

4. Results

The output of the orbit model is the satellite position in Earth Centered Inertial (ECI) and transformed to Earth Centered Earth Fixed (ECEF) frame and to Latitude Longitude Altitude (LLA). The result of the satellite position in ECI frame using the Keplerian orbit model, Satellite Tools Kit (STK), and comparison with RazakSAT is shown in Figure 8. The figure shows the comparison of the satellite position vector on X-, Y-, and Z-axis in the ECI frame. From the figure, the Keplerian orbit model and STK produced a position which is almost the same with the true value from RazakSAT.





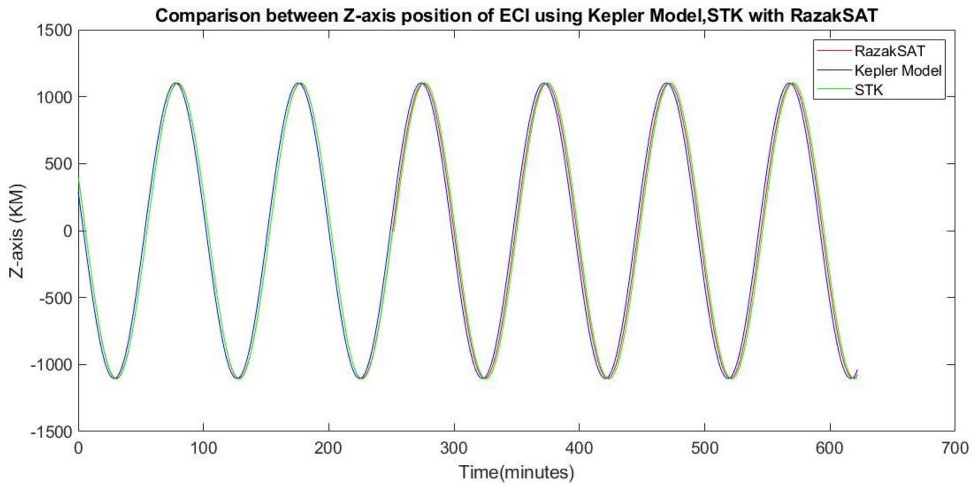


Fig. 8. Comparison of ECI around X, Y, Z-axes using Keplerian orbit model, STK, and RazakSAT

The different positions between the Keplerian orbit model, STK and RazakSAT orbit are defined as error (presented in km). The analysis of the ECI position between the Keplerian orbit model compared to RazakSAT is presented in Table 3. Firstly, the analysis method in terms of the maximum error, percentage error, RMSE, MAE and standard deviation is used to validate the ECI position vector of the Keplerian orbit model by bringing it to comparison with the RazakSAT. From Table 3, the maximum position errors for ECI on X, Y and Z-axis are 15.240 km, 40.741 km, and 1.480 km respectively. For all ECI axes, the percentage error is less than 5% error, where the maximum errors are 15.240 km, 40.741 km, and 1.480 km respectively for x, y, z –axes.

Table 3
Residual analysis on ECI position vector between Keplerian orbit model compared to RazakSAT

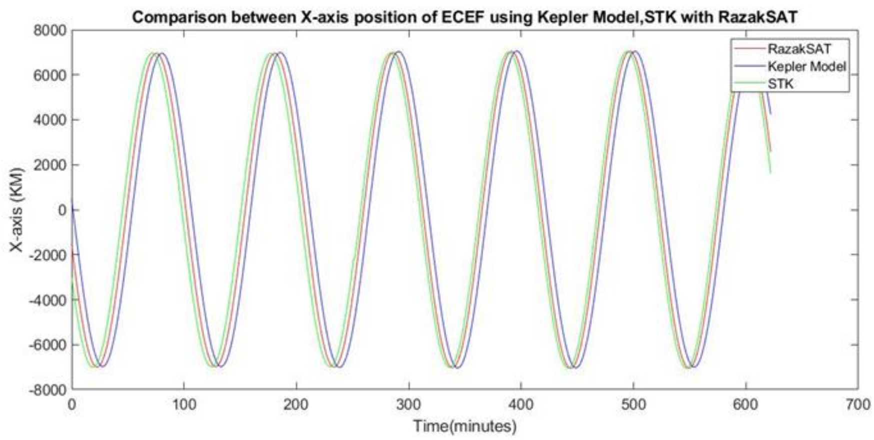
ECI Frame	Maximum Error (KM)	Error (%)	RMSE (KM)	MAE(KM)	Standard Deviation
X- axis	15.24	0.217	1415.86	1280.74	543.73
Y- axis	40.74	0.584	1390.47	1249.32	555.49
Z- axis	1.48	0.134	90.78	81.56	62.98

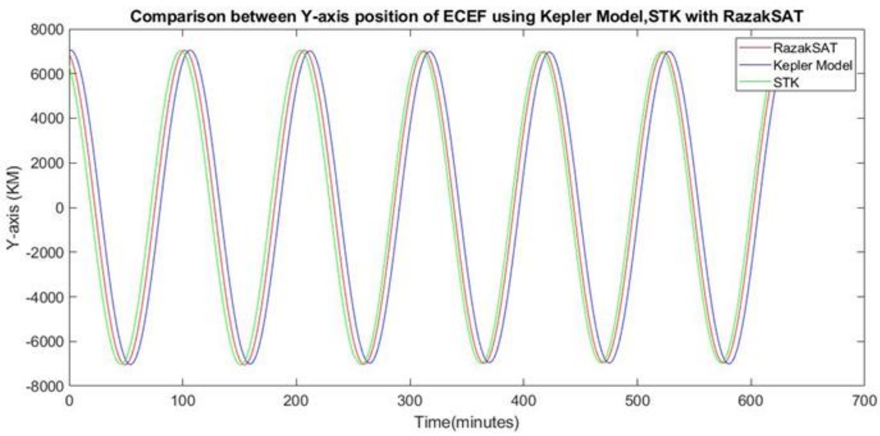
Root Mean Square Error (RMSE), Mean Absolute Error (MAE), and Standard deviation for all axes are significant respectively. Secondly, it is used to validate the ECI position vector of STK by comparing it to RazakSAT which is as shown in Table 4. From Table 4, the maximum position errors for ECI are 0.758 km, 0.410 km and 5.176 km respectively for X, Y and Z- axes. The same with Table 3, the percentage error for the Keplerian orbit model is less than 5%, where the RMSE and MAE for STK, and standard deviation are shown in Table 4 respectively. Based on the results on Table 3 and Table 4, it is discovered that STK is more accurate compared with the Keplerian orbit model and RazakSAT data functions as the reference. However, the analysis result around Z-axis shows that the Keplerian orbit model provides better accuracy than STK.

Table 4
Residual analysis on ECI position vector between Satellite Tools Kit (STK) compared to RazakSAT

ECEF Frame	Maximum Error (KM)	Error (%)	RMSE (KM)	MAE(KM)	Standard Deviation
X- axis	0.758	0.0108	287.71	204.90	201.98
Y- axis	0.410	0.0059	278.74	196.74	197.47
Z- axis	5.176	0.471	45.58	33.00	31.43

The result satellite positions in ECEF by using the Keplerian orbit model, STK, and comparison with RazakSAT are shown in Figure 8. As with the ECI frame, the results from Figure 9 show that the ECEF frame for the Keplerian orbit model and STK is almost similar to the pattern for RazakSAT. The residual analysis on the ECEF position between the Keplerian orbit model and STK are compared with RazakSAT as presented in Table 5 and Table 6.





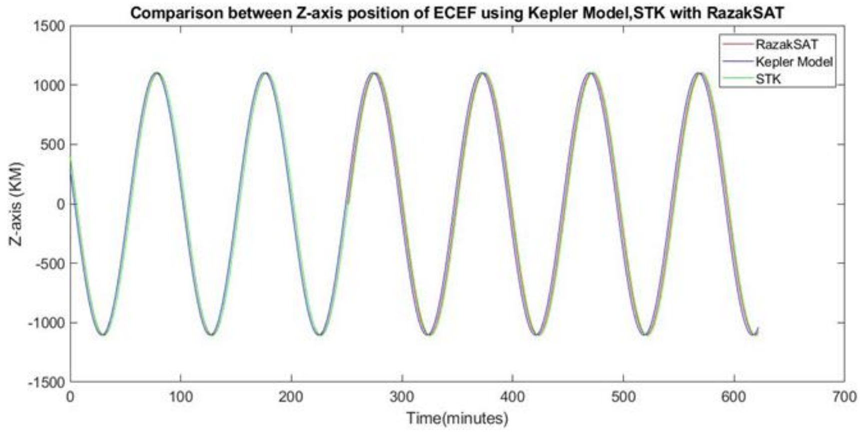


Fig. 9. Comparison of ECEF around X,Y,Z-axes using Keplerian orbit model, STK, and RazakSAT

Based on the results from Table 5 and Table 6, the different results on the maximum position error and percentage error between STK and Keplerian orbit model are less than 5%. However, the result error for RMSE and MAE gave too far marginal distance, especially for the Keplerian orbit model. According to this analysis, although the Keplerian orbit model illustrates that the maximum position error and percentage error are better than STK, in terms of RMSE, MAE and standard deviation, STK produces smaller error compared to the Keplerian orbit model. Therefore, for the ECEF frame, STK is still accurate compared to the Keplerian orbit model. The result points to a significant result with the percentage error less than 5% which follows ATSB specification. According to the result as well, RazakSAT is still in the right position on the orbit determined.

Table 5
Residual analysis on ECEF position vector between Keplerian orbit model compared to RazakSAT

ECEF Frame	Maximum Error (KM)	Error (%)	RMSE (KM)	MAE(KM)	Standard Deviation
X- axis	7.902	0.112	1392.478	1252.04	609.42
Y- axis	5.127	0.073	1413.88	1277.02	606.86
Z- axis	3.696	0.336	90.781	81.563	39.86

Table 6
Residual analysis on the ECEF position vector between Satellite Tools Kit (STK) and RazakSAT

ECEF Frame	Maximum Error (KM)	Error (%)	RMSE (KM)	MAE(KM)	Standard Deviation
X- axis	10.939	0.155	862.107	760.31	406.39
Y- axis	4.005	0.057	878.441	779.56	405.39
Z- axis	5.559	0.506	43.326	31.73	29.508

Figure 10 shows a comparison of the LLA around X, Y and Z-axes using the Keplerian orbit model, STK with RazakSAT. The latitude and the longitude positions are almost similar to that of RazakSAT; nevertheless, the value altitude of STK for maximum error is around 3.808 km.

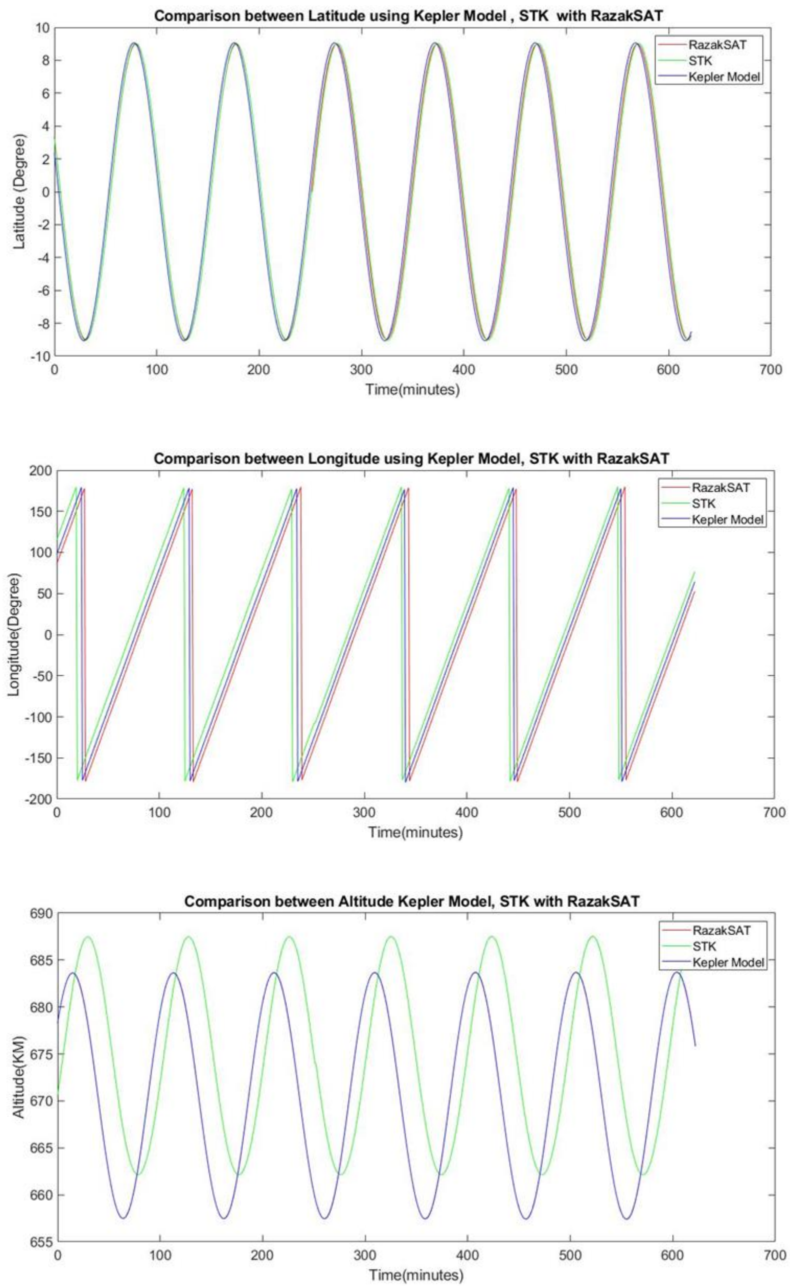


Fig. 10. Comparison of LLA around X,Y,Z-axes using Keplerian orbit model, STK and RazakSAT

From Table 7 and Table 8, the residual STK and Keplerian orbit model are smaller in terms of the latitude and longitude. However, in terms of the altitude, the STK result shows the maximum error of 3.808 km compared with the Keplerian orbit model around 0.024 km. The trend lines are still the same with RazakSAT and still relevant for verification that RazakSAT follows the requirement design on the satellite position.

Table 7
Residual analysis on LLA between Keplerian orbit model compared to RazakSAT

LLA	Maximum Error (KM)	Error (%)	RMSE (KM)	MAE(KM)	Standard Deviation
Latitude	0.084	0.930	0.745	0.670	0.325283
Longitude	0.519	0.289	64.97	23.00	60.76322
Altitude	0.024	0.003	0.031	0.025	0.018293

Table 8
Residual analysis on LLA between Satellite Tools Kit (STK) compared to RazakSAT

ECEF Frame	Maximum Error (KM)	Error (%)	RMSE (KM)	MAE(KM)	Standard Deviation
Latitude	0.054	0.604	0.354	0.261	0.238702
Longitude	0.222	0.124	93.986	48.78	80.33428
Altitude	3.808	0.557	9.418	8.122	4.767107

4. Conclusions

In this paper, the parameters of the Keplerian model were successfully implemented and enhanced as well as compared with its near equatorial orbit data of RazaSAT. The models were successful in estimating the position and orientation of the satellite in the NEqO orbit. ECI, ECEF and LLA coordinate vectors are used for analysing the position and orientation of the satellite. The outputs from the Keplerian model were verified using data from RazakSAT and ATSB orbit operational specification, i.e. with maximum percentage less than 5%. With the small error result from the Keplerian model by comparing it to the RazakSAT orbital data, the Keplerian model was successful in determining the ECI, ECEF and LLA of the satellite. The result from the Keplerian model has become important for the attitude determination method to estimate the attitude of the RazakSAT. The reliability analysis for the orbital data with the satellite analysis approach using Satellite Tools Kit software (STK) was successfully conducted by this researcher. RazakSAT is the world's first to be launched into Near Equatorial Orbit (NEqO) in 2009. Therefore, RazakSAT data is a reference orbital data verified by STK in terms of the position, orientation, magnetic field and sun model. By using the TLE data of RazakSAT provided from ATSB, STK software simulates the earth orbital satellite. Similar to the Keplerian orbit model, STK generated the ECI, ECEF, LLA results which were later corrected with RazakSAT modelling results. The result shows that the orbital data from RazakSAT is to follow the orbital specifications set by ATSB with maximum error less than 5%.

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