Development of Attitude Determination for Student Pico-Satellite INNOSAT

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Abstract: Information of satellite’s attitude is very useful in autonomous satellite control system. For a small satellite particularly, attitude sensor cannot be installed in the satellite because of the limited weight and power consumption. So, the attitude should be obtained from other sensors, they are a combination of two satellite’s positioning sensors, sun sensor and magnetometer. This paper represents about the development of attitude determination system for INNOSAT by using deterministic and recursive approach. Since given the deterministic method cannot be used when satellite is in eclipse, the recursive method will be used during the eclipse [10][12]. The deterministic attitude determination is calculated by using q-method method. Because of dynamic nonlinearity of the satellite, for recursive estimator was used Extended Kalman filter (EKF). To analyze and develop the attitude determination on ground, the measured attitude data was produced from Satellite Tool Kit (STK). Development of attitude determination is performed by using linear and nonlinear quaternion satellite model for both methods and also observing the effect of attitude shifting from deterministic to recursive method in eclipse problem.

Keywords : Attitude determination, INNOSAT, EKF, q-method(QUEST)

1. Introduction

INNOSAT is a Pico-satellite which was built by four universities in Malaysia and a company engaged in satellite manufacturing. The INNOSAT’s structure has been developed mainly by Astronautic Technology Sdn Bhd (ATSB) and experimental payload systems are developed by University Sains Malaysia (USM), Universiti Malaysia Perlis (UNIMAP), Universiti Teknologi Malaysia (UTM), Universiti Kebangsaan Malaysia (UKM) and USM contributes mainly in development of space experimental attitude determination sensors and solar panel sun sensors.

The main mission of INNOSAT is to take photos of Malaysian region from its orbit. Therefore, INNOSAT was carrying a camera as payload that installed in satellite. The captured photos will be stored temporarily in the memory until the satellite enters the communication region, then satellite downlink the photos to ground station. Secondary mission of this satellite is a technology demonstrator for the fourth university in the development of hardware sub-systems and methodologies for attitude determination and control.

INNOSAT will orbit the earth at the altitude of 680 km with 1e-5 eccentricity and 9° inclination from equator line. INNOSAT is a small satellite which has a dimension of 10x10x30 cm³ with 3 Kg of maximum weight and 15W of maximum power. INNOSAT image was shown in Figure 1. Generally small satellite like INNOSAT has those limitations. So the satellite has to be built with inexpensive, lightweight, and low power consumption. With these limitations INNOSAT
should be able to carry out mission (taking photos of Malaysia region). Quality of photos was strongly influenced by the attitude, position, and light intensity when the satellite takes the shooting. So attitude determination and control play an important role for achieving the mission subject and sustainability of satellite on its orbit a long it designed[4][7][8][10].

Attitude of the satellite was expressed by the angular position and angular velocity in satellite body frame with relative to local coordinate orbit frame. Accuracy of attitude information is very important in satellite control problem. The problems of accuracy such as noise measurement, bias, and misalignment were affected the attitude estimation. So, in attitude determination was proposed to use two or more sensors as optimal combination to estimate the attitude as a solution to improve the accuracy [4][8][10].

Limitations of mass and power budget for INNOSAT, has been forced to use lightweight hardware, small and low power consumption. There for the use of attitude sensors such as gyroscope is not possible because of the limitations. So INNOSAT attitude is determined and estimated using optimal combination of two position sensors, they are sun sensors and magnetometers. The Attitude cannot be determined by using this technique when the sun sensors are in eclipse phase. This will push to use two methods simultaneously which widely used by various satellite to determine its attitude. They are deterministic method[10][11][12] and Extended Kalman Filter (EKF) [5][6][8][10][12].

This paper represents the development of attitude determination for INNOSAT. Deterministic method is used as main attitude determination and EKF is used only when the satellite is in eclipse. Satellite dynamics are modeled by using quaternion to avoid singularity when using Euler[1][2][3].To develop and analyzed the attitude determination on ground, the measured sun sensor and magnetometer data was produced from Satellite Tool Kit (STK) software and orbital data of sun vector and Earth magnetic field was produced by sun model and IGRF data. The process was shown in Development of attitude determination is performed by using linear and nonlinear satellite quaternion model for both methods and also observing the effect of attitude shifting from deterministic to EKF during the eclipse problem.

2. Attitude Dynamics

2.1. Quaternion Representation

Unit quaternion can be described as,

\[ q = \begin{bmatrix} q_1 & q_2 & q_3 & q_4 \end{bmatrix}^T \]  \hspace{1cm} (1)

In term of Euler axis, quaternion can be represented with \( \vec{e} = [e_x, e_y, e_z] \) and angle \( \theta \). This vector element are expressed as follow,

\[ q_1 = e_x \sin \left( \frac{\theta}{2} \right), q_2 = e_y \sin \left( \frac{\theta}{2} \right) \]
\[ q_3 = e_z \sin \left( \frac{\theta}{2} \right), q_4 = \cos \left( \frac{\theta}{2} \right) \]  \hspace{1cm} (2)

Direct cosine matrix of quaternion can be calculated as follows,

\[ R_0^b = (q_4 - q^T q) I_3 + 2qq^T - 2qSO(q) \]  \hspace{1cm} (3)

Which gives \( R_0^b \) as,

\[ R_0^b = \begin{bmatrix}
1 - 2(q_3^2 + q_4^2) & 2(q_1 q_2 - q_3 q_4) & 2(q_1 q_3 + q_2 q_4) \\
2(q_1 q_2 + q_3 q_4) & 1 - 2(q_1^2 + q_3^2) & 2(q_2 q_3 - q_1 q_4) \\
2(q_1 q_3 - q_2 q_4) & 2(q_2 q_3 + q_1 q_4) & 1 - 2(q_2^2 + q_3^2)
\end{bmatrix} \]  \hspace{1cm} (4)

And

\[ R_0^b = \left( R_0^b \right)^{-1} = \left( R_0^b \right)^T = \begin{bmatrix} c_1^b & c_2^b & c_3^b \end{bmatrix} \]  \hspace{1cm} (5)

And attitude conversion from quaternion to Euler becomes,

\[ \begin{bmatrix} \phi \\ \theta \\ \psi \end{bmatrix} = \begin{bmatrix}
\arctan \left( \frac{2(q_1 q_2 + q_3 q_4)}{1 - 2(q_3^2 + q_4^2)} \right) \\
\arcsin \left( 2(q_2 q_3 - q_1 q_4) \right) \\
\arctan \left( \frac{2(q_1 q_4 + q_2 q_3)}{1 - 2(q_2^2 + q_3^2)} \right)
\end{bmatrix} \]  \hspace{1cm} (6)

2.2. Attitude Kinematic

The kinematic represent the satellite’s orientation in space and found by integration of the angular velocity. Angular velocity of the satellite is described by using quaternion as given by [2],
\[ \dot{q}(t) = \left[ \dot{q}_{13}(t) \right] = \left[ \begin{array}{c} -\frac{1}{2} q_{13} \tau_{ob}^b \\ \frac{1}{2} q_{13} (I + S(q_{13})) \omega_{ob}^b \\ I^{-1}(\tau_{ob} - (\omega_{ob}^b (t) \times I \omega_{ob}^b (t))) \end{array} \right] \] (7)

Where \( q_{13}(t) \) is a part of quaternion vector, \( q_{13}(t) \) is the scalar part and \( \omega_{ob}^b \) is angular velocity of body with respect to orbit.

### 2.3. Attitude Dynamic

The dynamic equation of motion is derived from the change in angular momentum of the satellite and an expression for the change in angular velocity, as a function of the applied torques, is sought. A detailed in [1][9]. The satellite dynamic equation is given by

\[ I \omega_{ob}^b (t) + \omega_{ob}^b (t) \times I \omega_{ob}^b (t) = \tau_c (t) + \tau_g (t) \] (8)

Where \( I \) is the satellite’s inertia matrix given in body frame, \( \omega_{ob}^b (t) \) is the angular velocity of the body frame relative to the inertial frame given in body frame, \( \tau_c (t) \) is the control torque and \( \tau_g (t) \) is gravity gradient torque that applied to the satellite.

### 2.4. Non-Linear Dynamics

By combining the kinematic equation and the dynamic equation, it will produce the nonlinear differential equation. By defining state vector of the system,

\[ x(t) = [q_s, q_1, q_2, q_3, \omega_{ob,x}^b, \omega_{ob,y}^b, \omega_{ob,z}^b]^T \] (9)

By inserting eq.(7) to eq. (8), the nonlinear dynamic becomes,

\[ \begin{bmatrix} \dot{q}_s \\ \dot{q}_{13} \\ \dot{\omega}_{ob}^b \\ \dot{\omega}_{ob}^b \\ \dot{\omega}_{ob}^b \end{bmatrix} = \begin{bmatrix} -\frac{1}{2} q_{13}(t)^T \omega_{ob}^b \\ \frac{1}{2} q_{13}(t) I + S(q_{13}(t)) \omega_{ob}^b \\ I^{-1}(\tau_{ob} - (\omega_{ob}^b (t) \times I \omega_{ob}^b (t))) \end{bmatrix} \] (10)

However, it is need to represent the attitude of satellite in body frame, so this can be done by exploiting the relation of

\[ \omega_{ob}^b = \omega_{ob}^b + \omega_{io}^b \\
\dot{\omega}_{ob}^b = \dot{\omega}_{ob}^b + \dot{R}_{ob}^b \omega_{io}^b \\
\dot{\omega}_{ob}^b = \dot{\omega}_{ob}^b - S(\omega_{ob}^b) R_{ob}^b \omega_{io}^b \] (11)

Where \( \omega_{io}^b = [0, -\omega_y, 0]^T \) is assume constant by substituting (5) to (11) and (11) to (10), nonlinear dynamic equation becomes,

\[ \begin{bmatrix} \dot{q}_s(t) \\ \dot{q}_{13}(t) \\ \dot{\omega}_{ob}^b(t) \\ \dot{\omega}_{ob}^b(t) \end{bmatrix} = \begin{bmatrix} -\frac{1}{2} q_{13}(t)^T \omega_{ob}^b \\ \frac{1}{2} q_{13}(t) I + S(q_{13}(t)) \omega_{ob}^b \\ I^{-1}(\tau_{ob} - (\omega_{ob}^b (t) \times I \omega_{ob}^b (t))) \end{bmatrix} \] (12)

And the state vector becomes,

\[ x(t) = [q_s, q_1, q_2, q_3, \omega_{ob,x}^b, \omega_{ob,y}^b, \omega_{ob,z}^b]^T \] (13)

### 2.5. Control Torque

Magnetic torque will produce magnetic dipole when current flow through its winding which is proportional with number of windings and cross section area of the coils [1][9][11]. Magnetic dipole will react with the perpendicular Earth’s magnetic field to generate torque. The torque generated by the magnetic torque can be modeled by,

\[ \tau_m^b = m^b \times B^b \] (14)

Where \( B^b = [B_x^b, B_y^b, B_z^b]^T \) is the local geomagnetic field vector relative to the satellite body and \( m^b \) is magnetic dipole vector which is define by,

\[ m^b = m_{x}^b + m_{y}^b + m_{z}^b = \begin{bmatrix} N_{ix} A_x \\ N_{ix} \end{bmatrix} = \begin{bmatrix} m_x \\ m_y \\ m_z \end{bmatrix} \] (15)

Where \( N_k \) is number of windings \( i_k \) is the coil current, and \( A_x \) is the span area of the coil. Based on vector property, equation (14) and (15) generated torque can be representing with,

\[ \tau_m^b = S(m^b) B^b = \begin{bmatrix} B_x^b m_y - B_y^b m_x \\ B_y^b m_z - B_z^b m_y \\ B_z^b m_x - B_x^b m_z \end{bmatrix} \] (16)

Because of local magnetic field vector around the satellite is calculated by using IGRF model, so calculated magnetic field vector is represented in orbit frame. From equation (16), magnetic field vector relative to body \( B^b = [B_x^b, B_y^b, B_z^b]^T \) will be provided from
magnetic field vector relative to orbit \( B^o = \left[ B^o_1 \ B^o_2 \ B^o_3 \right] \). So magnetic vector is satellite body become,

\[
B^b = R^b B^o
\]  

(17)

By substituting (17) to (16), magnetic torque equation becomes,

\[
\tau^m = S(m^b)B^b = S(m^b)R^b B^o
\]  

(18)

One of the effects of using magnetic torque is that they will contribute to the measurements of the Earth’s magnetic field [1][9][11].

2.6. Gravity Gradient Torque

The satellite which orbiting the Earth, will affect by gravitational forces of Earth and the forces will produce torque on satellite body. The torque is derived in [9][11] by assuming that Earth’s mass distribution is homogenous, the gravitational torque is derived as,

\[
\tau_{grav} = 3\mu u_x \times (Iu_x)
\]  

(19)

Where \( \mu \) is Earth’s gravitational coefficient, \( R_o \) is distance from Earth’s center (m), \( u_x \) unit vector toward nadir, and \( I \) is inertia matrix. Equation (19) in body will be,

\[
\tau_{grav} = 3\omega_3^2 c_3^b \times (Ic_3^b)
\]  

(20)

Where \( \omega_3^2 = \frac{\mu}{R_o^3} \) is angular velocity of orbit and \( c_3^b = \left[ c_{13}^b \ c_{23}^b \ c_{33}^b \right] \) is the third column of \( R^b \) (because only z-axis will be affected to Earth’s forces).

2.7. Complete Attitude Dynamic

Complete model is constructed by substituting equation (18) and (20) to (12), by assuming that internal torque is negligible. The nonlinear complete model becomes,

\[
\begin{bmatrix}
\dot{\phi}_1(t) \\
\dot{\phi}_2(t) \\
\dot{\phi}_3(t)
\end{bmatrix} =
\begin{bmatrix}
-\frac{1}{2}\gamma \omega_3(t)\sigma_3^b \\
\frac{1}{2}\gamma \omega_3(t)\sigma_3^b \\
\omega_3(t) - \dot{\omega}_3 - \dot{\omega}_3 - \dot{\omega}_3 + 3 \omega_3 S(\epsilon) \sigma_3^b + S(\omega_3) R^b B^o - S(\sigma_3) u_x^b
\end{bmatrix}
\]  

(21)

3. Attitude Determination Method

3.1. q-Method (QUEST)

Wahba’s problem is formulated as an eigenvector problem and directly estimates an optimal attitude quaternion minimizing Wahba’s problem [10][12]. Given a set of \( n \geq 2 \) vector observation, a loss function is formulated, known as Wahba’s problem given by

\[
L(A_{ob}^b) = \frac{1}{2} \sum_{j=1}^{n} w_j \left| b_j - A_{ob}^b r_j \right|^2
\]  

(22)

Where \( w_j \) is the weight of the \( j^{th} \) vector observation, \( r_j \) is vector in the reference frame and \( A_{ob}^b \) is the orthonormal rotation matrix, representing the rotation from the reference frame to the body frame, which is sought [10][12]. The loss function is a weighted sum squared of the difference between the measured and the reference vectors in the body frame. By minimizing the loss function, given in equation (22) an optimal attitude may be estimated. Wahba’s loss function can also be written as

\[
L(A_{ob}^b) = \sum_{j=1}^{n} w_j - tr(A_{ob}^b B^o)
\]  

(23)

Where \( B \) is given by

\[
B = \sum_{j=1}^{n} w_j b_j r_j^T
\]  

(24)

Based on equation (23), the loss function \( L(A_{ob}^b) \) is minimized by maximizing \( tr(A_{ob}^b B^o) \). In the q-method, the rotation matrix \( A_{ob}^b \) is determined as a quaternion by computing the symmetric \( 4 \times 4 \) matrix \( K \).

\[
K = \begin{bmatrix}
S & -\sigma I & Z \\
Z^T & \sigma
\end{bmatrix}
\]  

(25)

With

\[
S = B + B^T
\]  

(26)

\[
Z = \begin{bmatrix}
B_{23} - B_{32} \\
B_{31} - B_{13} \\
B_{12} - B_{21}
\end{bmatrix}
\]  

(27)
The optimal quaternion by using QUEST method is,

\[ q_{opt} = \frac{1}{\sqrt{v^2 + |x|^2}} \begin{bmatrix} x \\ y \end{bmatrix} \]  

(31)

In which \( \gamma \) and \( x \) are given by,

\[ \gamma = \left( \lambda_{\max}^2 - (trB)^2 + tr(adjS) \right) (\lambda_{\max} + trB) - detS \]  

(32)

\[ x = \left[ \left( \lambda_{\max}^2 - (trB)^2 + tr(adjS) \right) I + (\lambda_{\max} + trB)S + S^2 \right] z \]  

(33)

\[ L(A_{opt}) = \lambda_o - \lambda_{\max} \]  

(34)

As a note, that angular velocity of the satellite cannot be determined by using q-method.

2.8. Extended Kalman Filter (EKF)

The operation of the EKF is illustrated as shown in Figure 3. To apply EKF techniques, it is first necessary to describe the real behavior by a set of nonlinear models. The system can be represented by a nonlinear continuous system model and a nonlinear discrete measurement model in equations (35) and (36), respectively.

\[ \Phi_k \approx I + \frac{\partial f(x_i,u_i) \partial x_k}{x_i \theta_k} T_i \]  

(41)

where \( T_i \) is sample time and \( \Phi_k \) is the measurement matrix derived from rotation matrix from orbit to body reference \( R_o \), and define with,

\[ H_k = [h_i \ h_{i+1} \ldots \ h_{i+n} \ 0] \]  

and,

\[ h_i = \begin{bmatrix} \frac{\partial R_o \ (\hat{q}_i)}{\hat{q}_k} B_{orb,k+1} \\ \frac{\partial R_o \ (\hat{q}_i)}{\hat{q}_k} S_{orb,k+1} \end{bmatrix} \]  

component of quaternion at time k then \( B_{orb,k+1} \) is model magnetic field (IGRF) at time k+1 and \( S_{orb,k+1} \) is model sun vector at time k+1.

The EKF algorithm for the system is defined in equation (35) according to [8]. [12] and [14] given by,

\[ K_k = P_k H_k^T \left( H_k P_k H_k^T + R \right)^{-1} \]  

(42)

\[ \hat{x}_k = \hat{x}_k + K_k \left( y_{\text{meas},k} - H_k \hat{x}_k \right) \]  

or

\[ \hat{x}_k = \hat{x}_k + K_k \left( y_{\text{meas},k} - R_o (\hat{q}_k)^{\text{meas},k} \right) \]  

(43)

The process noise \( w(t) \) is described by equation (37) where \( Q(t) \) is the strength of process noise. The discrete noise \( v_k \) has the covariance described by equation (38), with \( R_k \) representing the measurement noise covariance matrix.

\[ E[w(t)w^T(t)] = Q(t) \]  

(37)

\[ E[v_k v_k^T] = R_k \]  

(38)
\[P_{k+1} = \left[ I - K_k H_k \right] P_k \left[ I - K_k H_k \right]^T + K_k R K_k^T \] (44)
\[\bar{x}_{k+1} = \Phi_k \hat{x}_k \] (45)

where \(K_k\) is the Kalman gain matrix, \(P_k\) is the error covariance, and \(\bar{P}_{k+1}\) is the error covariance propagation.
The quaternion update can be interpreted as a rotation and the quaternion product given by,
\[\hat{q}_k = \bar{q}_k \otimes K_{q,k} v_k \] (46)
The angular velocity is updated using equation
\[\hat{\omega}_{ob,k} = \bar{\omega}_{ob,k} + K_{\omega,k} v_k \] (47)

where \(v_k\) is the result of innovation process given by,
\[v_k = y_k - H_k \bar{x}_k \] (48)
which is different between real and the predicted measurement.

4. Simulation Results and Analysis of Attitude Determination

Attitude determination calculation is started by inserting initial values from Table 1 to Table 3 to equations in Section 2.7 and 3.1 for deterministic method. And by inserting initial values in Table 1 to Table 4 to Section 2.7 and 3.2 for EKF. Measurement data of magnetic field and sun vector are provided from Satellite Tool Kit (STK) and magnetic field and sun vectors surrounding of satellite are provided from IGRF and sun model which are used in [10]. The simulations were performed for two orbits to determine the effect of the presence of eclipses.

<p>| Table 1. Structure parameters of INNOSAT |</p>
<table>
<thead>
<tr>
<th>Parameters</th>
<th>Values</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>(l_{xx})</td>
<td>32716516.64e-9</td>
<td>Kgm²</td>
</tr>
<tr>
<td>(l_{xy})</td>
<td>-518537.85e-9</td>
<td>Kgm²</td>
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<td>(l_{xz})</td>
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<td>(l_{yy})</td>
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<td>c.g (Z_m)</td>
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<p>| Table 2. Satellite Two Line Elements |</p>
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</tr>
<tr>
<td>Inclination</td>
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<td>deg</td>
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<tr>
<td>Eccentricity</td>
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<tr>
<td>Argument of perigee</td>
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<td>RAAN</td>
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<td>Mean Motion in Revolution</td>
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<p>| Table 3. Initial attitude of INNOSAT |</p>
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<td>(q_{20})</td>
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<td>(q_{30})</td>
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<td>(q_{40})</td>
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<p>| Table 4. EKF initial parameters |</p>
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<th>Values</th>
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<td>(P_k)</td>
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<tr>
<td>(R)</td>
<td>(R=diag(8.5e-4 8.5e-4 8.5e-4 8.5e-4 8.5e-4 8.5e-4 8.5e-4 8.5e-4 8.5e-4 8.5e-4))</td>
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</table>

Estimation outputs of q-method and EKF are attitude in quaternion, as stated in eq.(13), then the outputs are converted to Euler angle (\(\phi, \Theta, \Psi\)) by using eq.(6). The Euler outputs from q-method and EKF are compared with Euler from STK software which called as true attitude.

[Figure 4. Estimated roll (\(\phi\)) attitude]
methods is 0.01. This means the problems in switching estimated attitude data from QUEST to EKF or vice versa at the eclipse will produce insignificant effect on the satellite dynamics. The same analysis was done for Figure 5 and Figure 6, the maximum error is 0.28 degree and 4.8 degree respectively.

Figure 4 shows the difference between true and estimated roll (\(\varphi\)) attitude. It’s about 2.5 deg of steady state attitude error can be made when applying QUEST and EKF. Maximum errors from both methods were relatively large. This is due to the differences in the measurement data of the sun’s and magnetic field vector with the actual values (Figure 7). This happen particularly from EKF. This also could be caused by difference parameters of mathematical model and physical model. Inertial moment of INNOSAT is coupled, so the Euler attitude on each axis is influenced by the attitude of the other axis, so it caused difficulty in obtaining pure attitude estimation on each axis.

Figure 4 to Figure 6 shows the estimated attitude of the satellite and the analysis shows that the maximum error between QUEST and EKF vector

Figure 7. Normalized magnetic field and sun vector

The QUEST method cannot be used to estimate the angular velocity of satellite. In this case the EKF method is used instead. Figure 8 and Figure 10 shows the angular velocity for \(\omega_x\) and \(\omega_z\) which is estimated using the EKF method and this is possible because it has steady state error of 0.07.
rad/s and 1.7e-5 rad/s respectively. However, estimation of $\omega_1$ using EKF is not adequate because the steady state error value is 0.4 rad/s. Specifically for $\omega_1$, estimation is obtained using a differential EKF calculation of estimated pitch ($\theta$) from Figure 5 using equation (49) and obtained the results in Figure 11.

$$\frac{\partial \theta}{\partial t} = \theta(t + 1) - \theta(t)$$

(49)

Figure 11. Estimate $\omega_1$ by using differential equation

5. Conclusion and Further Works

Attitude determination for INNOSAT has been developed by using deterministic q-Method (QUEST) and recursive using EKF. It has been shown that attitude of INNOSAT can be determined using two position sensors. The maximum error which can be obtained using the QUEST and the EKF is 4.8 deg for yaw attitude and 2.5 deg for roll and only 0.28 deg of error for pitch attitude. Estimated angular velocity in X and Z-axis are acceptable since using EKF but for Y-axis is only possible calculated using difference of estimated pitch angle.

For further works, its need to attempt using the Euler dynamic model for the EKF to know better possibility of estimated results. Continuous EKF is also possible to use to reduce computational load. Improvement of inertial moment is also possible to attempt in order to reduce attitude coupling.

6. Acknowledgement

Many thanks to ATSB Sdn Bhd that has been providing much information and data of INNOSAT and also team of SEADS USM that has given many helped in the success of this project.

7. Reference