MODELING AND SIMULATION OF PROLATE DUAL-SPIN SATELLITE DYNAMICS IN AN INCLINED ELLIPTICAL ORBIT: CASE STUDY OF PALAPA B2R SATELLITE

J. Muliadi*, Said D. Jenie**, A. Budiyono***
*) Department of Aeronautics and Astronautics, ITB
**) Department of Aeronautics and Astronautics, ITB
***) Department of Aeronautics and Astronautics, ITB

ABSTRACT

In response to the interest to re-use Palapa B2R satellite nearing its End of Life (EOL) time, an idea to incline the satellite orbit in order to cover a new region has emerged in the recent years. As a prolate dual-spin vehicle, Palapa B2R has to be stabilized against its internal energy dissipation effect. This work is focused on analyzing the dynamics of the reusable satellite in its inclined orbit. The study discusses in particular the stability of the prolate dual-spin satellite in the effect of perturbed field of gravity due to the inclination of its elliptical orbit. Palapa B2R physical data was substituted into the dual-spin's equation of motion. The coefficient of zonal harmonics J2 was induced into the gravity-gradient moment term that affects the satellite attitude. The satellite's motion and attitude were then simulated in the perturbed gravitational field by J2, with the variation of orbit's eccentricity and inclination. The analysis of the satellite dynamics and its stability was conducted for designing a control system for the vehicle in its new inclined orbit.

1 INTRODUCTION

A semi-rigid body is stable only when spinning about its major axis. In a related study, Bracewell and Garriott (Ref. [8] pp. 62-64) concluded that the four turnstile wire antennae of Explorer I were dissipating energy; thus, causing a transfer of body spin axis from the minimum inertia (prolate) to a transverse axis of maximum inertia (oblate). To meet this stability criterion, most of early dual-spin vehicles were designed in an oblate configuration.

As a case of study, this work used Palapa B2R physical data to analyze the dynamics of the vehicle. Palapa B2R is a communication satellite of Indonesia. In its orbit, it was operated by PT. Telkom (Indonesian State Telecommunication Company). Near the satellite’s End of Life
(EOL) time, several Africans and Polynesians countries have shown interest to buy and re-use Palapa B2R. Because of those countries' location in the southern latitudes, an idea emerged to incline the satellite's orbit. The current paper elaborates the analysis of the vehicle dynamics in its inclined orbit.

2 REFERENCE COORDINATE SYSTEM

2.1 Body Reference Coordinate System (Body Axes)

![Figure 2-1: Platform axis components](image)

The definition of Platform and Body axes is well-defined in the literature. Figure 2-1 are illustrated those axes with their origin at the satellite's e.g. while

Figure 2-2 are showed the axes in the space. In this paper, the Platform Axis Components will be identified as Body Reference Coordinate System or Body axis.

![Figure 2-2: Body reference coordinate system](image)

2.2 Stability reference coordinate system (stability axes)

Stability Reference Coordinate System (Stability Axes) was defined as a set of local horizon axes for the satellite. It is a target axes for the satellite's Body Axes to point its antennae to the Earth. All these axes are presented in Figure 2-3.

![Figure 2-3: Stability reference coordinate system](image)

2.3 Inertial reference coordinate system (inertial axes)

Inertial Reference Coordinate System (Inertial Axes) is defined as a geocentric non-rotating equatorial reference frame with Zi axis which coincides with the rotation axis of the Earth and points to the North Pole; the Xi axis lies in equatorial plane and points towards the vernal equinox. The Yi axis completes a right-handed Cartesian frame of reference (Figure 2-4). In this Inertial Axis, Newton's laws of motion are valid for the satellite's translation and rotation.

![Figure 2-4: Inertial reference coordinate system](image)
EULER ANGLES (ORIENTATION ANGLES)

3.1 Orientation of Body Axes in Inertial Axes

Figure 3-2: Euler angles of body axis in inertial axis

The yaw angle $ip$, pitch angle $0$ and roll angle $cp$, respectively defines the angle of rotation in Z-, Y- and X- axis of the Body frame with respect to its nominal condition in Inertial Axes. These angles are shown in Figure 3-2.

3.2 Orientation of body axes in stability axes

Figure 3-3: Deviation of body axes from stability axes

Euler Angles of Body Axes with respect to Stability Axes are defined to describe the attitude perturbation from its local horizon (its stationary or nominal condition) (Figure 3-3). If the satellite deviates to large, the antennae will point away from the Earth. The Euler Angles are:

Figure 3-4: Euler angles of body axis with respect to stability axis

*Yaw, Pitch and Roll Angle Deviation* ($tfs$, $0s$, and $cps$), which respectively denotes the angle of perturbation because of the rotation in Z-, Y-, and X- axis of the Body frame with respect to Stability Axes. These angles are shown in Figure 3-4.
It this paper the spacecraft will be treated as a non-axially symmetric vehicle. In addition, the satellite will be operated in inclined elliptical orbit, which means non-equatorial and non-circular orbit. So, by following and combining Kaplan's (Ref. [8] pp. 199-204) and Agrawal's techniques in deriving the equation of Gravity Gradient Moment for Spacecraft, the authors derived the equation of Gravity Gradient Moment for Prolate Dual-Spin Satellite in its inclined elliptical orbit.

The gravitational force (dFo) corresponding to a differential element of mass, dm, shown in Figure 4-1, is

\[ \text{d}F_G = g - diw \]  

Eq. 2

where \( g \) denotes gravity vector at dm. The unsymmetrical mass distribution of the Earth induced a zonal harmonic coefficient (Jk, Ref [8] pp. 273-282) that perturbs the homogenous of the Earth gravity's field. In this work, the oblateness of the Earth induced the zonal harmonic coefficient that is limited to second order, J2. The gravity vector equation at a point in space is

\[ g = \frac{\mu_E}{R^2} \left[ 1 - \frac{3}{2} J_2 \left( \frac{R_e}{R} \right)^2 \right] \mathbf{R} \]  

Eq. 3

With \( \mu_E \) = Earth gravitational parameter; \( R \) and \( \mathbf{R} \) = the distance from the center of the Earth in scalar and vector notation; \( R_e \) = the height measured perpendicular from the Earth equatorial plane; \( \mathbf{R} \) = the radius of the Earth equator.

Continuing to the moment equation,

\[ \mathbf{M}_G = \int_r \mathbf{r} \times \text{d}F_G = \int_r \mathbf{r} \times g \cdot \text{d}m \]  

Eq. 4

while the position of differential element of mass, dm, in Figure 4-1 is

\[ \mathbf{R}_m = \mathbf{R} + \mathbf{r} \]  

Eq. 5

Therefore, substituting Eq. 5 to replace \( \mathbf{R} \) in Eq. 3, then inserting the result to Eq. 4, transform Eq. 4 into:

\[ \mathbf{M}_G = -\int \mathbf{r} \times \]  

\[ \left[ \frac{\mu_E}{(\mathbf{R} + \mathbf{r})^2} \left[ 1 - \frac{3}{2} J_2 \left( \frac{R_e}{\mathbf{R} + \mathbf{r}} \right)^2 \right] \right] (\mathbf{R} + \mathbf{r}) \cdot \text{d}m \]  

Eq. 6

Figure 4-2: Deviation from steady attitude
The $R_\text{Zv}$ variable is the component of satellite's position at Z-Axis in Earth. The factor of inclination was induced in $R_{Zj}$ variable, because the height of the satellite will be varying in the inclined orbit. The factor of eccentricity was induced in $R$ variable. For $e>0$, the value of $R$ is varying along the orbit.

5 ORBITAL MOTIONS
5.1 Parameters of Keplerian Orbit

Astronomy defines 6 quantities to describe the orbit and position of heavenly body, namely a, e, i, $Q$, a, and r. The definition of those parameters can be found in many textbook in orbital mechanics. Figure 5-2 describes the geometry of the orbital parameters.

The equation was derived by relating Newton's Second Law of Motion and Newton's Law for Gravitation. To sketch the satellite's orbit, Eq. 11 were integrated two times with 6 initial values, initial velocity $J_X$, $R_{Yj}$, $R_{Zj}$, 3 initial position $R_{Xj}$, $R_{Yi}$, and $R_{Zi}$ when $t=0$. From Jenie (Ref. [6]), initial velocity can be expressed by

$$
\begin{align*}
R_X &= R \cdot \sin \theta_s \\
R_Y &= -R \cdot (\sin \phi_s \cdot \cos \theta_s) \\
R_Z &= -R \cdot (\cos \phi_s \cdot \cos \theta_s) \\
\end{align*}
\text{Eq. 7}
$$

By linearizing the equations, the Linearized Gravity Moment Equations read,

$$
\begin{align*}
M_{0X} &= G_X \cdot \phi_s \\
M_{0Y} &= G_Y \cdot \theta_s \\
M_{0Z} &= G_Z \cdot \theta_s \\
\end{align*}
\text{Eq. 8}
$$

Where coefficients of $\phi_s$ and $\theta_s$ are:

$$
\begin{align*}
G_X &= g_0 \cdot \pm (l_2^b - l_2^c) \\
G_Y &= g_0 \cdot \pm (l_1^b - l_1^c) \\
G_Z &= g_0 \cdot \pm (l_3^b - l_3^c) \\
\end{align*}
\text{Eq. 9}
$$

And coefficient $g_0$ is,

$$
g_0 = \left\{ 3 \frac{\mu_p}{R^2} + \frac{105}{2} \frac{\mu_p}{R^2} \cdot J_2 \cdot \Re_s^2 \cdot R_{Zj}^2 \right\} \text{Eq. 10}
$$

The $R_{Zv}$ variable is the component of satellite's position at Z-Axis in Earth. The factor of inclination was induced in $R_{Zj}$ variable, because the height of the satellite will be varying in the inclined orbit. The factor of eccentricity was induced in $R$ variable. For $e>0$, the value of $R$ is varying along the orbit.

5.2 Orbital Motion Equations

Simulation of the orbital motion uses scalar differentials equation in $X_j$, $Y_j$, and $Z_j$-axis as follows:

$$
\begin{align*}
\ddot{R}_{Xj} &= \frac{\mu_p}{R^3} \cdot R_{Xj} \\
\ddot{R}_{Yj} &= \frac{\mu_p}{R^3} \cdot R_{Yj} \\
\ddot{R}_{Zj} &= \frac{\mu_p}{R^3} \cdot R_{Zj} \\
\end{align*}
\text{Eq. 11}
$$

The equation was derived by relating Newton's Second Law of Motion and Newton's Law for Gravitation. To sketch the satellite's orbit, Eq. 11 were integrated two times with 6 initial values, initial velocity $J_X$, $R_{Yj}$, $R_{Zj}$, 3 initial position $R_{Xj}$, $R_{Yi}$, and $R_{Zi}$ when $t=0$. From Jenie (Ref. [6]), initial velocity can be expressed by
with transformation matrix $C^{O_6}_t$, and its elements as follows:

$$
C^{O_6}_t = \begin{bmatrix}
C_{11} & C_{12} & C_{13} \\
C_{21} & C_{22} & C_{23} \\
C_{31} & C_{32} & C_{33}
\end{bmatrix}
$$

$$
C_{11} = (\cos \omega \cos \Omega - \sin \omega \cos i \sin \Omega) \\
C_{12} = (-\sin \omega \cos \Omega - \cos \omega \cos i \sin \Omega) \\
C_{13} = (\sin i) \\
C_{21} = (\cos \omega \sin \Omega + \sin \omega \cos i \cos \Omega) \\
C_{22} = (-\sin \omega \sin \Omega + \cos \omega \cos i \cos \Omega) \\
C_{23} = (\cos \cos i) \\
C_{31} = (\sin \omega \sin i) \\
C_{32} = (\cos \omega \sin i) \\
C_{33} = (\cos i)
$$

When simulating the satellite's orbit, the orbital parameters are:

$$
\bar{\theta}_0 = 0^\circ, \ \bar{\theta}_0 = 0^\circ, \omega_0 = 0^\circ \\
\alpha_0 = 8078.14 \text{ km for circular orbit} \\
\bar{R} = 1700 \text{ km above the sea level; for elliptic orbit with } e=0.2, \text{ perigee, } R_{\text{perigee}} = 85 \text{ km above the sea level.}
$$

In the state space model for attitude dynamics, the satellite transversal velocity that perpendicular to its radius to the Earth, $V_\theta$, will be needed.

A. E. Roy (Ref. [9] p. 81) relates $\sin \bar{\omega}$ in Figure 4-2 with eccentricity and semimajor axis as follows:

$$
\sin \bar{\omega} = \left[ \frac{a^2 \cdot (1-e^2)}{R \cdot (2-a-R)} \right]^{1/2} \quad \text{Eq. 14}
$$

With Eq. 14, the component of velocity in theta direction can be stated as

$$
V_\theta = V \cdot \sqrt{\frac{a^2 \cdot (1-e^2)}{R \cdot (2-a-R)}} \quad \text{Eq. 15}
$$

## 6 DYNAMICS OF PROLATE DUAL SPIN SATELLITE

Palapa B2R is a prolate configuration Communication Satellite (HS 376). In order to stabilize its attitude and pointing direction, Palapa B2R uses its rotor spinning. Control moments were produced by the angular acceleration and deceleration of the rotor's spin. Because of the rotor's spinning motion and the configuration of satellites inertia, the satellite's motion in the yaw mode is coupled with its roll mode. In addition, by the imbalance from the satellite's antennae reflector configuration (hz), the satellite's motion in the pitch mode is coupled with its yaw mode.

### 6.1 Dynamic Equations of Motion

Bryson has shown the equation of spacecraft motion in De-Spin Active Nutation Damping in Ref. [2 pp. 62-68]. In this work, the author will insert gravity gradient moment as an external moment that perturbs the satellite's attitude. Using Bryson's technique in deriving the De-Spin Active Nutation Damping equations, the author adds several modifications. The results are:

**The Dynamics Equation at X-axis,**

\[
(I_X + I_Y) \cdot \ddot{p} - (I_5 \cdot \Omega_{R5}) \cdot \dot{r} = M_X \quad \text{Eq. 16}
\]

**The Dynamics Equation at Y-axis,**

\[
I_Y \cdot \ddot{q} + I_{YZ} \cdot \ddot{r} + \left( \frac{N \cdot K_Y}{R_{zz}} + c \right) \cdot \left( I_Z - 1 \right) \cdot q + \left( \frac{N \cdot K_Y}{R_{zz}} \right) \cdot \dot{r} = M_Y \quad \text{Eq. 17}
\]

**The Dynamics Equation at Z-axis,**

\[
I_{YZ} \cdot \ddot{q} + (I_Z + I_Y) \cdot \ddot{r} + (I_5 \cdot \Omega_{R5}) \cdot \dot{r} = M_Z \quad \text{Eq. 18}
\]
6.2 Dynamic Equations with Gravity Gradient Moments

Substituting $M_x$ in Eq. 16 with $M_0x$ in Eq. 8 yields the Dynamics Equation at X-axis in Inclined Elliptical Orbit,

$$I_x + I_{yz} \cdot \dot{\theta} - (I_S \cdot \Omega_{xS}) \cdot r$$

$$-G_x \cdot \dot{\phi}_s = 0 \quad \text{Eq. 19}$$

Substituting $M_y$ in Eq. 17 with $M_0y$ in Eq. 8 yields the Dynamics Equation with Gravity Gradient Moments at Y-axis in Inclined Elliptical Orbit,

$$I_y + I_{yz} \cdot \dot{\theta} + \left( \frac{N \cdot K_y}{R_{de}} + c \right) \left( I_y + I_z \right) q + I_{yz} \cdot I_y + I_{yz} \cdot \dot{r} - G_y \cdot \dot{\theta}_s + \frac{N}{R_{de}} \cdot \delta e = 0 \quad \text{Eq. 20}$$

Finally, substituting $M_z$ in Eq. 18 with $M_0z$ in Eq. 8 yields the Dynamics Equation at Z-axis in Inclined Elliptical Orbit,

$$(I_x + I_{yz}) \cdot \dot{\theta} - (I_S \cdot \Omega_{zS}) \cdot r$$

$$-G_x \cdot \dot{\phi}_s = 0 \quad \text{Eq. 21}$$

6.3 Kinematics Equations of Motion
6.3.1 Kinematics equations in inertial axes

Bryson has shown that the equation of kinematics in relationship between Euler Angle Rates and Angular Velocity (Ref. [2] pp. 8-9) can be approximated as

$$\dot{\phi} \equiv p + n \cdot \psi$$

$$\dot{\theta} \equiv q + n$$

$$\dot{\psi} \equiv r - n \cdot \phi$$

Figure 6-1: Rotation of local horizon Axes (stability axes), $n$

where $n = \text{orbital angular velocity}$. These kinematics equations are derived in Inertial Axis by Newton's first and second law of motion.

6.3.2 Kinematics equations in inertial axes

In the beginning of simulation, the satellite's antennae still points to the Earth for a moment. By the inertia (the Newton first law of motion) the satellite's attitude will drift at rotor spin axis with drift rate equals to orbital angular rate, $9$, or equals to initial value of $n$, no. Therefore, to measure the deviation in Stability Axes, the kinematics equation needs to be reduced by the effect of Local Horizon Axes (Stability Axes) initial angular drift with respect to Inertial Axes, no. Following Byson's techniques in deriving the kinematics equation, the Kinematics Equation of Dual Spin Satellite in Stability Axis can be written as follows

$$\dot{\phi}_s \equiv p + \delta n \cdot \psi_s$$

$$\dot{\theta}_s \equiv q + \delta n$$

$$\psi_s \equiv r - \delta n \cdot \phi_s$$

$$\text{Eq. 23}$$

where $\delta n = (n - n_0)$.

6.4 State Space Model of Prolate Dual-Spin

Combining The Dynamics Equation in Inclined Elliptical Orbit and Kinematics Equation of Dual Spin Satellite in Stability Axis yields State Space Model for Dual-Spin Satellite in Stability Axes as follows,

$$\begin{pmatrix} p \\ q \\ r \\ \phi_s \\ \theta_s \\ \psi_s \end{pmatrix} = [A] \cdot \begin{pmatrix} p \\ q \\ r \\ \phi_s \\ \theta_s \\ \psi_s \end{pmatrix} + [B] \cdot \begin{pmatrix} \delta e \\ \delta n \end{pmatrix} \quad \text{Eq. 24}$$
where the [A] and [B] matrices are:

\[
[A] = \begin{bmatrix}
0 & 0 & A_{13} & A_{23} & 0 & 0 \\
A_{31} & A_{12} & A_{22} & 0 & A_{32} & 0 \\
A_{31} & A_{12} & A_{22} & 0 & A_{32} & 0 \\
1 & 0 & 0 & 0 & 0 & 0 \\
0 & 1 & 0 & 0 & 0 & 0 \\
0 & 0 & 1 & -\delta n & 0 & 0
\end{bmatrix}
\]

\[
[B] = \begin{bmatrix}
0 & 0 \\
B_{31} & 0 \\
B_{31} & 0 \\
0 & 0 \\
0 & 1 \\
0 & 0
\end{bmatrix}
\]

By defining \( \Delta t \) as follows,

\( \Delta t = (I_Y I_Z + I_Y I_T - I_{YZ}^2) \)

the [A] matrix elements in 1st line are:

\[
A_{13} = -\frac{1}{(I_X + I_T)}(I_S \cdot \Omega_{R0})
\]

\[
A_{14} = -\frac{1}{(I_X + I_T)} G_X
\]

the [A] matrix elements in 2nd line are:

\[
A_{21} = -\frac{I_{YZ}}{\Delta t} (I_S \cdot \Omega_{R0})
\]

\[
A_{22} = \frac{I_T + I_T}{\Delta t} \left( \frac{N \cdot K_V}{R_{d e}} + c \right) \left( \frac{I_Y}{I_S} + 1 \right)
\]

\[
A_{23} = \frac{I_T + I_T}{\Delta t} \left( \frac{N \cdot K_V}{R_{d e}} + c \right) \left( \frac{I_{YZ}}{I_S} \right)
\]

\[
A_{24} = -\frac{(I_T + I_T)}{\Delta t} G_Y + \frac{I_{YZ}}{\Delta t} G_Z
\]

the [A] matrix elements in 3rd line are:

\[
A_{31} = \frac{I_Y}{\Delta t} (I_S \cdot \Omega_{R0})
\]

\[
A_{32} = -\frac{I_{YZ}}{\Delta t} \left( \frac{N \cdot K_V}{R_{d e}} + c \right) \left( \frac{I_Y}{I_S} + 1 \right)
\]

\[
A_{33} = -\frac{I_{YZ}}{\Delta t} \left( \frac{N \cdot K_V}{R_{d e}} + c \right) \left( \frac{1}{I_S} \right)
\]

\[
A_{34} = \frac{I_{YZ}}{\Delta t} G_Y - \frac{I_{YZ}}{\Delta t} G_Y
\]

the [B] matrix elements in 1st column are:

\[
B_{21} = \frac{I_T + I_T}{\Delta t} \left( \frac{N}{R_{d e}} \right)
\]

and

\[
n = -\dot{\theta} = \frac{-V_2}{R}
\]

\[
\delta n = (n - n_0)
\]

\[
\dot{\theta} = \dot{\theta} = \left( \frac{V_0}{R} \right) - \left( \frac{V_0}{R} \right)
\]

7 RESULTS OF OPEN LOOP SIMULATION AND INTERPRETATIONS

The values of [A]'s and [B]'s elements are shown in Error! Reference source not found.. The value of \( A_{14}, A_{25}, A_{35} \) and \( \delta n \) will be time-varying for elliptical or inclined orbit. However, the elements of [A] are constant value only for circular orbit at equatorial plane.

\[
[A] = \begin{bmatrix}
9 & 0 & 0 & 0 & 0 & 0 \\
-0.0073 & 9.714 \times 10^{-6} & -2.495 \times 10^{-5} & 0 & A_{14} & 0 \\
0 & 0 & 0 & 0 & 0 & 0 \\
-5.1218 \times 10^{-5} & 0 & 1.7735 \times 10^{-5} & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0
\end{bmatrix}
\]

\[
[B] = \begin{bmatrix}
0 & 0 \\
0 & 0 \\
0 & 0 \\
0 & 0 \\
0 & 0 \\
0 & 0
\end{bmatrix}
\]

(4-25)

7.1 Simulation in Longitudinal Mode

7.1.1 Effect of eccentricity in inclined orbit (\( i = 30^\circ \))

- Plot of \( \theta_s \) because of impulsive input \( \delta \theta_{ref} \) (Figure 7-1)

- Plot of \( \theta_s \) due to elliptic orbital drift input \( \delta n \) (Figure 7-2)

Figure 7-1: Plot of \( \theta_s \); input \( \delta \theta_{ref} \); \( i = 30^\circ \)

Figure 7-2: Plot of \( \theta_s \) due to elliptic orbital drift input \( \delta n \)
After impulsive disturbance were applied, Gravity Gradient Moment induced subsidence mode in circular orbit. While the increment of eccentricity, drove that subsidence mode into long periodic oscillation mode, which had the same period with orbital period. This long periodic oscillation mode oscillates from -20° to +90° as superposition of many oscillation mode.

7.1.2 Effect of inclination in elliptic orbit (e = 0.2)

- Plot of 0s due to impulsive input 5e_ref (Figure 7-3)
- Plot of 0s due to elliptic orbital drift input 6n (Figure 7-4)

Figure 7-3: Plot of 0s and 0n; e=0 & e=0.2; 2 orbital periods; i=30deg

Figure 7-4: Plot of 0s; i=0 deg, 30deg dan 60 deg; 1 Orbit and 2 Orbits; e=0.2

Inclination increment effect in longitudinal motion mode was negligible for Dual-Spin Satellite. For inclination at 0°, 30°, and 60°, the graphic curves for 0s were almost aligned.

7.2 Simulation in Lateral Mode
7.2.1 Effect of eccentricity in inclined orbit (i = 30°)

W Plot of ϕs due to impulsive input 6e_ref (Figure 7-5)

Figure 7-5: Plot of ϕs; e=0 and e=0.2; i=30deg
After impulsive disturbance were applied, Gravity Gradient Moment induced roll-librations mode in for $\phi_8$, which oscillates from $-1 \times 10^{-3}$ to $+1 \times 10^{-3}$ deg. In circular orbit, this roll-librations mode in for $\phi_8$ was damped after $\pm 400$ sec. While in elliptic orbit, the eccentricity induced long periodic oscillation mode, which oscillates from $-4 \times 10^{-3}$ to $+4 \times 10^{-3}$ deg.

7.2.2 Effect of inclination in elliptic orbit ($e = 0.2$)

- Plot of $\phi_8$ due to impulsive input $\delta e_{ref}$ (Figure 7-7)

![Plot of $\phi_8$ due to impulsive input $\delta e_{ref}$](image)

Figure 7-7: Plot of $\phi_8$; $i=0$deg, 30deg dan 60deg; $e=0.2$

- Plot of $\phi_8$ due to impulsive input $\delta e_{ref}$ (Figure 7-9)

![Plot of $\psi_8$ due to impulsive input $\delta e_{ref}$](image)

Figure 7-9: Plot of $\psi_8$; $e=0$ and $e=0.2$; $i=30$deg

7.3 Simulation in Directional Mode

7.3.1 Effect of eccentricity in inclined orbit ($i = 30^\circ$)

- Plot of $\psi_8$ due to impulsive input $\delta e$ (Figure 7-8)

![Plot of $\phi_8$ and $\delta n$](image)

Figure 7-8: Plot of $\phi_8$ and $\delta n$; $i=0$deg, 30deg and 60deg; 2 Orbits $e=0.2$

Inclination increment effect in longitudinal motion mode was negligible for Dual-Spin Satellite. For inclination at 0°, 30°, and 60°, the graphic curves for $\phi_8$ were almost aligned.
After impulsive disturbance were applied, Gravity Gradient Moment induced yaw-librations mode in for $\psi_s$. In circular orbit, this yaw-librations mode in for $\psi_s$ was damped after ±400 sec. In the elliptic orbit, the eccentricity induces long periodic oscillation mode, which oscillates from $-5.5 \times 10^{-3}$ to $+1.5 \times 10^{-3}$ (deg).

### 7.3.2 Effect of inclination in elliptic orbit ($e = 0.2$)

Plot of $\psi_s$ due to impulsive input $<\delta s_e>$ (Figure 7-11)

Inclination increment effect in longitudinal motion mode was negligible for Dual-Spin Satellite. For inclination at 0°, 30°, and 60°, the graphic curves for $\psi_s$ were almost aligned.

### 8 CONCLUDING REMARKS

In Open-Loop simulation, the librations and subsidence mode are present in longitudinal, lateral and directional motion. However, in lateral and directional motion, the Gravity Gradient moment induced a divergence mode if the simulation were run for more than 4 orbital periods (1 Orbital Periods is 7225 sec).

The effect of Gravity Gradient Moments is destabilizing the lateral and directional motion for the dual-spin satellite. In reverse, the effect of Gravity Gradient Moments is stabilizing longitudinal motion. The effect of increasing the orbital eccentricity, $e$, is the presence of the long period oscillation mode in the longitudinal and lateral directional motion.

The effect of increasing the inclination of Orbital Plane, $i$, can be neglected in the longitudinal motion. In lateral directional motion, increasing the inclination will reduce steady state error of $\psi_s$ and $\psi_{ps}$.
REFERENCES

A. E. Biyson, 1994. Control of Spacecraft and Aircraft, Princeton University Press, Princeton (New Jersey).

B. N. Agrawal, 1986. Design of Geosynchronous Spacecraft, Prentice-Hall Inc., Englewood Cliffs (New Jersey).

Cornelisse, J. W. with H. F. R. Schoyer and K-F. Wakker, 1979. Rocket Propulsion and Spaceflight Dynamics, Pitman Publishing Ltd., London.

Hughes Aircraft Company, 1982. Palapa-B System Summary, Hughes Aircraft Company.

Hughes Aircraft Company, 1990. Palapa-B2R System Summary HS-376E-22868, Hughes Aircraft Company.

Jenie, Said D., 2002. Astrodynamics 1 (Lecture Notes PN-350), In Indonesian. Department of Aeronautics and Astronautics, Bandung Institute of Technology, Bandung.

Jenie, Said D., 2003. Flight Control (Lecture Notes PN-4041), In Indonesian. Department of Aeronautics and Astronautics, Bandung Institute of Technology, Bandung.

M. H. Kaplan, 1976. Modern Spacecraft Dynamics & Control, John Wiley & Sons, New York.

Roy, Archie E., 1982. Orbital Motion, Adam Hilger Ltd., Bristol (England).