Deployment Dynamics of Solar Array with Body Rates

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Abstract

Solar array used in satellites consists of a yoke and three to six panels is kept stowed during launch and deployed in orbit. The yoke and panels are interconnected by the hinges. The energy for the deployment is provided by the pre-loaded torsion springs mounted at each one of the hinges. The deployment is in a near accordion way to minimise the shock loads at the hinges. Close Control Loops (CCLs) are employed to achieve the accordion deployment. A case study for the deployment dynamics of a solar array of two wings has been considered. Each wing consists of a yoke and three panels. The satellite will have body rates due to separation disturbances at the time of injection into the orbit. The Solar Array Drive Assembly (SADA) is free to rotate at the time of deployment of solar array. During deployment of the array and because of the body rates, the solar array is acted upon by an induced torque about the SADA axis, which makes it to rotate. Because of this rotation, the solar array may hit any projection, which in turn affect the deployment and /or damage some sensitive devices. The deployment dynamics of solar array with spacecraft body rates has been carried out using ADAMS software. Simulation is carried out in ADAMS to study the possibility of hitting/interfering.

1. Introduction

A typical solar array used in satellite consists of a yoke and three to six panels, is kept stowed during launch and deployed in orbit. Figure 1 shows the configuration of solar array in stowed, partially deployed and fully deployed modes. The panels and yoke are inter connected by a pair of hinges. The energy for the deployment is provided by the torsion springs mounted at each one of the hinges. The deployment is carried out in an accordion way, so that different hinges latch up simultaneously, to minimise the shock loads at the hinges. This is because the energy gets countered between successive joints due to change in the direction of rotation between adjacent joints. CCLs are employed to achieve the accordion deployment. CCL provide direction control and help in converting a multi-degree of freedom system to a near single degree of freedom system. It may also be noted that the deployed solar array is rotated about pitch axis by SADA for orienting the array towards the sun.

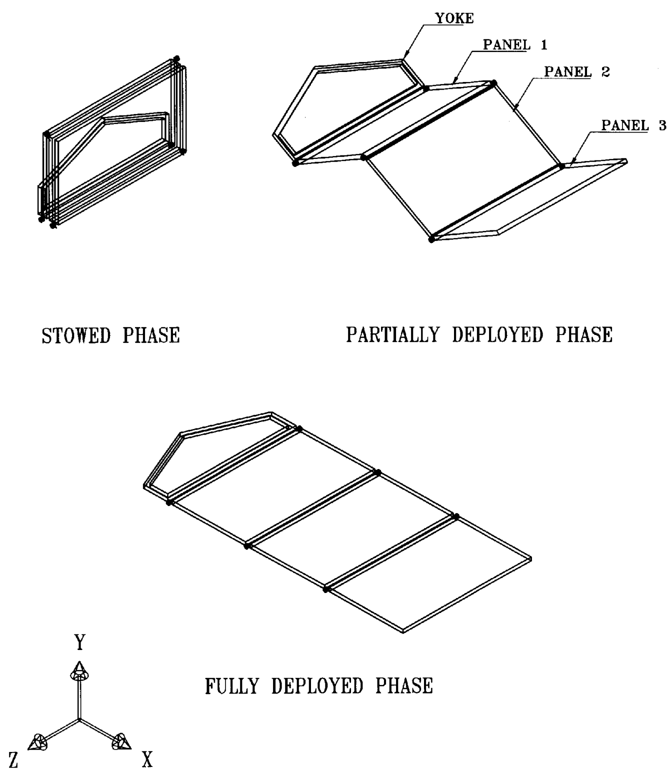

Figure 1: Schematic representation of solar array in stowed, partially deployed and fully deployed modes

A case study for deployment dynamics of a solar array with two wings has been considered. Each wing consists of a yoke and three panels. The satellite will have body rates due to separation disturbances at the time of injection in orbit. The deployment of the array follows with a pre-determined time. The SADA is free to rotate at the time of deployment of the array. During deployment of array and because of body rates, the solar array is acted upon by an induced torque about SADA axis, which makes it rotate. Because of this rotation, the solar array may hit any projection, which in turn can affect the deployment and/or damage some sensitive devices.

The deployment dynamic analysis of solar array with spacecraft body rates has been carried out using ADAMS software. The satellite body and two wings of solar array have been modelled as free floating. The objective of this simulation is to study motion of the solar array and this study would allow to find out any likely hood of array hitting/interfering projection of the satellite during deployment.

2. Details of Modelling

In this section the modelling of satellite with solar array is described. The mathematical modelling of CCL using subroutine of ADAMS is described.

2.1 Geometric Model

Figure 2 shows the model of the satellite with two wings of stowed solar array on either side of spacecraft body. The yoke and panels are connected by hinges. The spacecraft body with payload deck is modelled using box command. The yoke is modelled by extruding the cross section of yoke on the triangular shape. The panels are created by using box command. The revolute joint with a preloaded torsion spring created between yoke and panels. The similar joints are created between panels. As the yoke and panels rotate about SADA a revolute joint is created between the spacecraft body and the yoke about the pitch axis. The above procedure is repeated for the other wing of satellite.

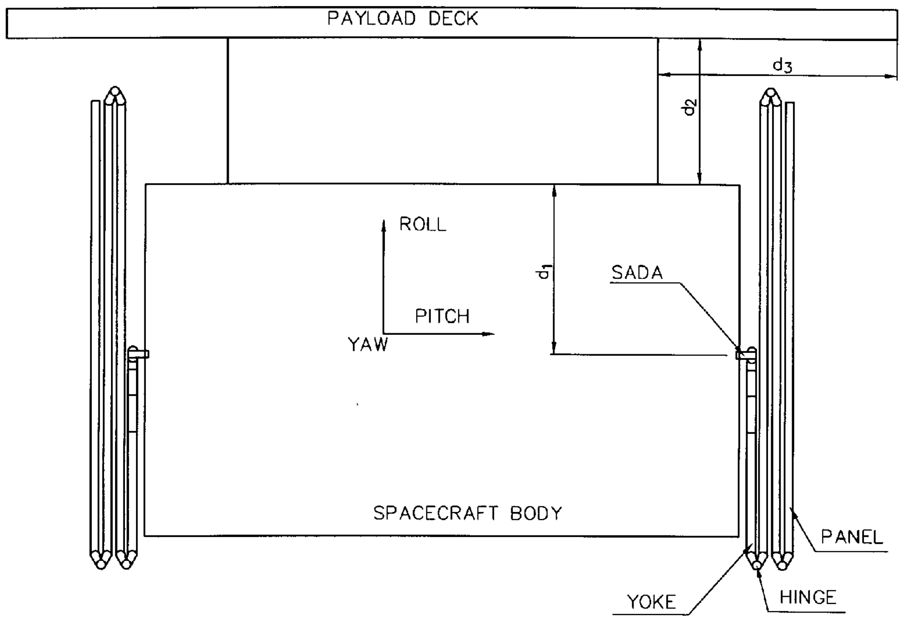

Figure 2: Schematic representation of stowed solar array with spacecraft

The satellite in orbit has 6 Degrees of freedom (DOF). The yoke and panels of each wing have 4 DOF. A degree of freedom corresponding to free rotation of yoke and panels about pitch axis is considered. The total DOF for each wing is 5. Hence, the satellite with two wings is having 16 DOF.

2.2 Modelling of CCL

A typical CCL used in solar array consists of pre-tensioned wire rope loop passing over two pulleys mounted between successive joints. A schematic of CCL connecting the yoke and panel 1 is shown in Figure 3. Similar CCLs are used at other joints. The generalised co-ordinate system used in ADAMS software is shown in Figure 4. In the absence of tension springs in CCL, for a finite rotation angle α of yoke, the panel rotates through an angle $\frac{R_{PY}}{R_{P1}}\alpha$, where R_{PY} is the Radius of SADA-yoke joint pulley and R_{P1} is the Radius of yoke-panel 1 joint pulley. However due to the presence of tension spring, the first panel rotates by an angle β. This difference in angle of rotation changes the length of CCL springs affecting the energy in tension springs, through which the torque between the successive joints gets adjusted.

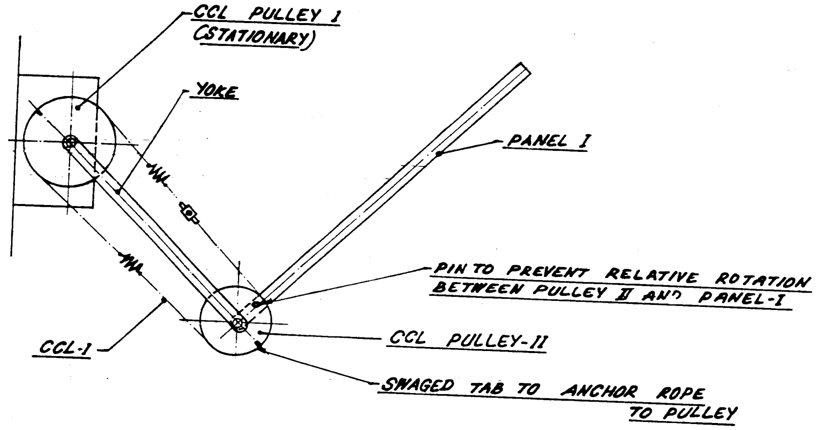

Figure 3: Schematic of CCL connecting yoke and panel 1

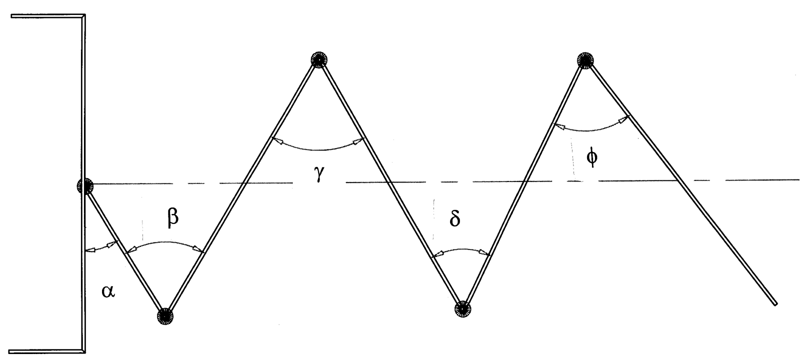

Figure 4: Generalised co-ordinates.

The change in the energy of tension springs of each wing is given by

$$U_{C1} = \frac{1}{2}\left[\frac{R_{PY}}{R_{P1}}\alpha + \beta\right]^2 R_{P1}^2 (K_{C11}+K_{C12})$$

$$U_{C2} = \frac{1}{2}\left[\frac{R_{P2}}{R_{P3}}\beta + \gamma\right]^2 R_{P3}^2 (K_{C21}+K_{C22})$$

$$U_{C3} = \frac{1}{2}\left[\frac{R_{P4}}{R_{P5}}\gamma + \delta\right]^2 R_{P5}^2 (K_{C31}+K_{C32})$$

Total potential energy U of CCLs is given by

$$U=U_{C1}+ U_{C2}+ U_{C3}$$

using, $\frac{R_{PY}}{R_{P1}} = 2$, $R_{P1} = R_{P2} = R_{P3} = R_{P4} = R_{P5} = R$ and $K_{CIJ} = K$, in the above equation the generalised forces at various DOF are given by

$$-\frac{\partial U}{\partial \alpha} = -4K\,R^2\,(2\,\alpha + \beta)$$

$$-\frac{\partial U}{\partial \beta} = -4K\,R^2\,(2\alpha + 2\beta + \gamma)$$

$$-\frac{\partial U}{\partial \lambda} = -4K\,R^2\,(\beta + 2\gamma + \delta)$$

$$-\frac{\partial U}{\partial \delta} = -4K\,R^2\,(\gamma + \delta\,)$$

The above generalised forces have been modelled using GFOSUB by passing the required marker IDs. As the GFOSUB resides outside ADAMS/VIEW this can be linked to any other model of solar array.

3. Results and Discussion

The physical parameters of solar array and the spring characteristics are given in Table-1a, Table-1b and Table-2 respectively. Simulation studies have been performed for various combinations of body rates of satellite. The typical results for the body rates of spacecraft of 3 degrees/sec about all the three axis is presented here. The objective is to study the possibility of solar array hitting the pay load deck projection during deployment in orbit. This will affect the performance of some sensitive devices and also can hinder the deployment. Latch up occurs when yoke rotates through 90 degrees or panels rotate through 180 degrees. The sensor statements supported by ADAMS senses latch up and stops the simulation. The path traced by the farthest point from the centre of spacecraft on the panel was plotted as a function of time. Figure 5 shows the angle of opening of yoke and panels with time. It is observed from the above figure that the yoke has oscillations during deployment. But the panels will not have any significant oscillatory motion during deployment. The yoke locks when the deployment angle is 90 degrees and the panels lock when the relative angle of deployment is 180 degrees as described earlier.

Table-1a: Physical parameters of spacecraft and solar array

Length of Panel (m)	1.40
Width of Panel (m)	1.80
Length of Yoke (m)	0.70
Spacecraft body Dimensions (m)	
Width (Pitch direction)	1.55
Height (Roll direction)	1.085
Length (Yaw direction)	1.65
d_1	0.504
d_2	0.35
d_3	0.55

Table -1b: Mass and Inertia of spacecraft and solar array

Part	Mass (Kg)	Mass moment of Inertia (Kgm^2)		
		I_{XX}	I_{YY}	I_{ZZ}
Satellite	1290.0	987.0	564.0	941.0
Yoke	3.3	1.08	0.92	0.164
Panel	8.7	3.77	2.35	1.42

Table-2: Characteristics of deployment springs

Joint	Stiffness (N.m/rad)	Pre Rotation Angle (rad)
SADA-Yoke	0.2498	7.854
Yoke - Panel 1	0.0936	11.082
Panel 1- Panel 2	0.0936	11.082
Panel 2 - Panel 3	0.0936	11.082

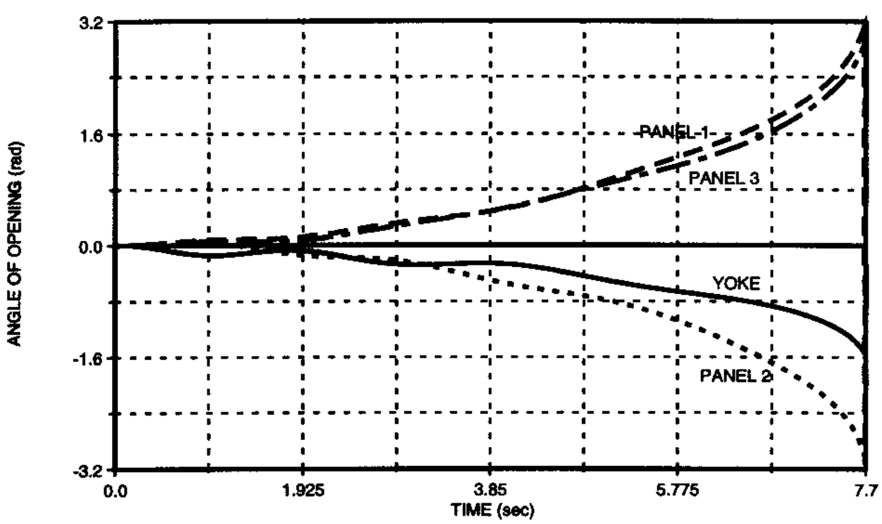

Figure 5: Angle of opening of yoke and panels versus time

Figure 6 shows the variation of rotation of the yoke and panels about pitch axis during deployment. As the spacecraft is subjected to body rates the yoke and panels will be subjected to Coriolis component of acceleration. This induces torque about the pitch axis. It is observed from the above Figure that the pitch angle increases to 6.15 degrees at the end of deployment. The magnitude of this pitch angle depends upon the body rates of the spacecraft. The pitch angle can increase further with increase in body rates of spacecraft.

In studying the motion of panels during deployment, the farthest point on the panel from the pitch axis is to be considered as the panels are free to rotate about that axis. Hence, the motion of corner of the panels is taken. The two corners along the width of the panel is taken, as one of the corners may hit the projection during deployment. The radius vector **R** one of the corner of the panel-1 from the coordinate system attached to SADA yoke joint is given by

$$\mathbf{R} = X\,\mathbf{i} + Y\,\mathbf{j} + Z\,\mathbf{k}$$

Where, X, Y and Z are the coordinates of the corner of panel-1 from the SADA yoke joint.

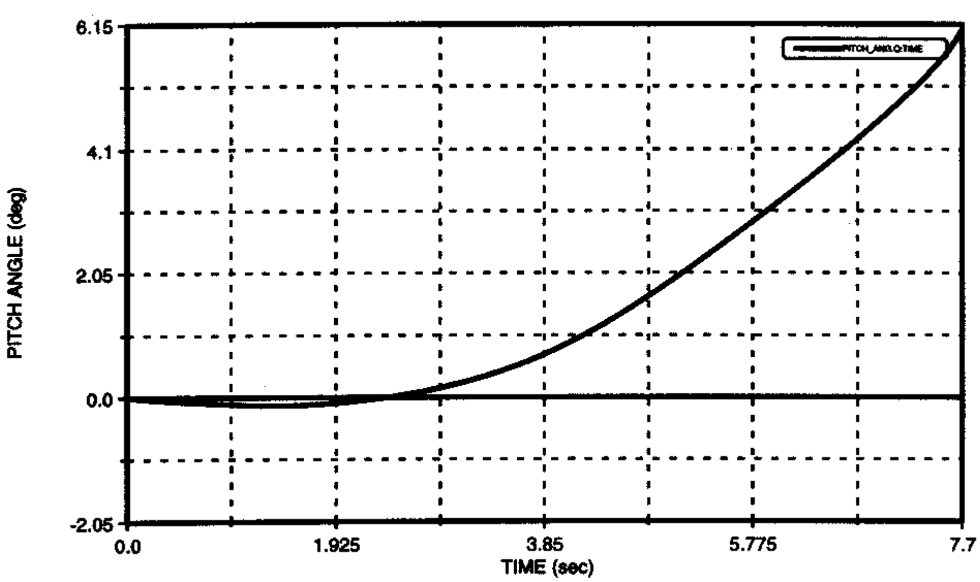

Figure 6: Variation of pitch angle with time

Figure 7, Figure 8 and Figure 9 shows the trajectory X, Y and Z coordinates of the first panel during deployment. The other corner is not presented as it moves away from the projection during deployment. From the trajectory of the panel, the motion in X direction generally increases with time due to increase in the deployment angle of yoke and panel with respect to time. However, in the initial phase of motion the X coordinate shows oscillations due to forward and backward motion of yoke. It is observed from Figure 8 that the Y co-ordinate shows oscillations during initial phases of motion and increases to a maximum value. This is due to the rotation of spacecraft and rapid increase in the angle of opening of yoke and at the same time the panel 1 does not open as fast as yoke during initial phase of deployment. Further, the Y coordinate decreases with time during deployment. It is observed from Figure 9 that the Z coordinate decreases during initial phase of deployment due to the combined effect of rotation of spacecraft and the angle of opening of yoke and panel.

During deployment, the yoke and panels are free to rotate about pitch axis, the Y coordinate plays an important role in deciding the projection of the spacecraft along pitch axis. In order to avoid interference of panel-1 during deployment the following constraints must be satisfied.

$$X \leq d_3$$

$$Y \leq d_1 + d_2$$

In addition, the following constraints can be satisfied by reducing the spacecraft body rates and also by choosing the stiffness and pre rotation angle of the springs.

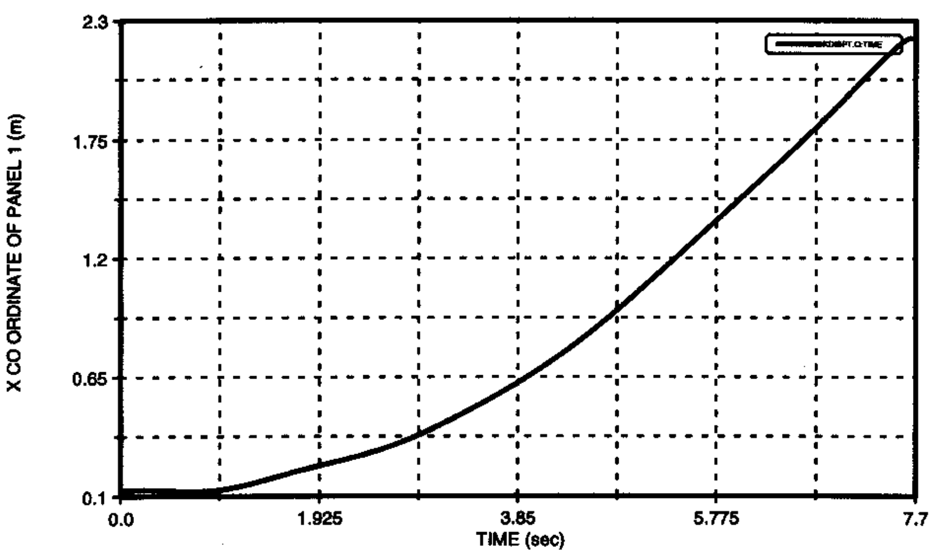

Figure 7: Trajectory of X co ordinate of panel 1

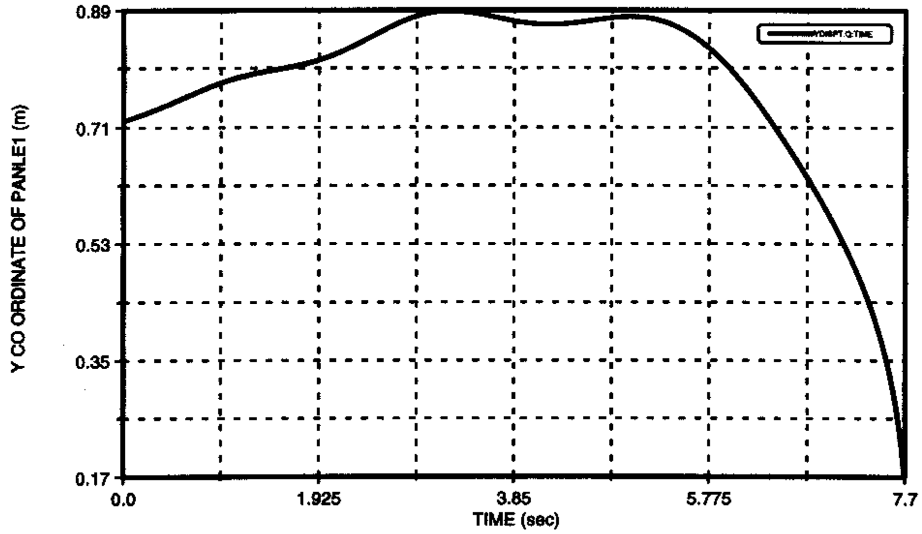

Figure 8: Trajectory of Y coordinate panel 1

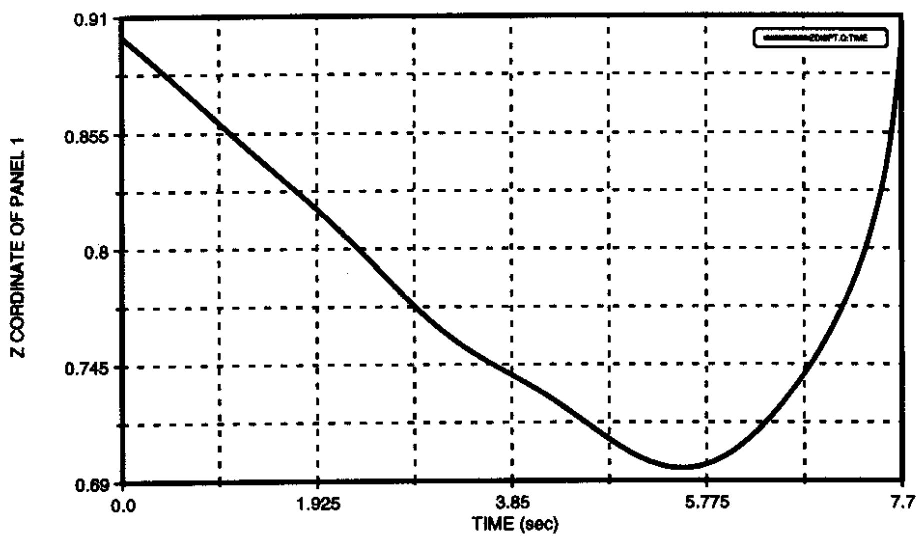

Figure 9: Trajectory of Z coordinate of panel 1

The study of dynamics of solar arrays simulating the body rates of spacecraft using ADAMS is useful in deciding the layout of the spacecraft deck. This is also useful in deciding the spring stiffness and length of each panel and yoke.

4. Conclusion

The ADAMS software is used in modelling the motion of the satellite with two wings of solar array attached to the two faces of the spacecraft. The deployment of yoke and panel was studied with the spacecraft having the body rates about all the three axis. The generalised forces of CCLs of the solar array is modelled using the subroutine GFOSUB. The above analysis was useful in choosing the various physical parameters of yoke and array. This study was also useful in deciding/limiting the dimensions of the spacecraft payload deck projection for the free motion of array in orbit.

Acknowledgment

The authors sincerely acknowledge the support given by Sri M. Nageswara Rao, Group Director, Spacecraft Mechanisms group. The authors also thank Prof. A.V Patki, Deputy Director, Mechanical systems area and Dr. P. S. Goel Director, ISAC for their constant encouragement.