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Design of Sliding Mode Attitude Control for Communication Spacecraft

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1. Introduction

Control problem of a spacecraft is an important topic in automatic control engineering. A body orbiting the Earth in geosynchronous orbit has instabilities in attitude dynamics and disturbances caused by the Earth, the Moon, the Sun and other bodies in space. These effects force the body to lose initial orbit and attitude. Here the control system takes important part of spacecraft missions where it keeps the body in designed orbit and desired attitude. The control system consists of control elements and control algorithms which are developed for the mission by a control engineer [1]. The commonly used control elements for a spacecraft in geosynchronous orbit are thrusters, reaction or momentum wheels, etc.

The sliding mode theory has an attention in the aerospace field. The technique permits the use of a lower order system model for generating control commands. On the other hand, the system is robust to the external disturbances and includes unmodelled dynamics, as well. The theory and methods of sliding mode control design principles are investigated [1]-[3], etc. Variable structure systems with nonlinear control techniques and dead-band on switching function for sliding mode controllers are introduced [4].

A variable structure control design for rigid body spacecraft attitude dynamics with quaternion representation for optimal sliding mode control which consists of three parts: equivalent control, sliding variable, and relay control where simulation results illustrate that the motion along the sliding mode is insensitive to parameter variations and unmodeled effects is given [5]. An automatic controller for active nutation damping in momentum biased stabilized spacecraft is introduced [6], where robust feedback stabilization of roll and yaw angular dynamics are achieved with prescribed qualitative characteristics for a spinning satellite. A smooth sliding mode control which requires well-estimated initial condition for quaternion based spacecraft attitude tracking maneuver is studied [7] where the chattering is eliminated by replacing saturation instead of signum function. A class of uncertain nonlinear systems decoupled by state variable feedback with sliding mode approach for attitude control of an orbiting spacecraft is considered [8] where simulation results show that precise attitude control is accomplished in spite of the uncertainty in the system. As seen from simulations spacecraft is stabilized approximately in 10 seconds. However there is a chattering in control action and thrusters are operating after stabilization of the spacecraft attitude dynamics. A reference book for various spacecraft attitude and

orbit dynamics, orbit transfer methods, and different control strategies such as PID and robust control, pulse modulation of thruster control etc. is covered [9]. An attitude control with reaction wheels is evaluated in [10].

There are many papers concerning control of flexible spacecraft. A maneuvering of a flexible spinning spacecraft is treated with variable structure control [11] where system is stabilized between 60-100 seconds for small and large angle maneuvers. An application with one sided dead-band for robust closed-loop control design for a flexible spacecraft slew maneuver using on-off thrusters is studied [12] where analytical simulations and experimental results demonstrate that the proposed switching function provides significant improvement in slew maneuver performance. The size of single-sided dead-band in switching function provides the capability of a tradeoff between maneuver time and fuel expenditure. Rotational maneuver and vibration suppression of an elastic spacecraft is considered [13] where pitch angle trajectories are asymptotically tracked by an adaptive controller. Variable structure control and active suspension of flexible spacecraft during attitude maneuver is studied [14] where positive position feedback technique is used to suspend vibration and variable sliding mode with pulse-width pulse-frequency modulation to eliminate chattering. An adaptive variable structure control of spacecraft dynamics with command input shaping which eliminate residual vibration is studied [15] where PD, conventional and adaptive variable structure output feedback controllers with and without input shaping are compared and simulated. Vibration suspension of flexible spacecraft during attitude maneuvers is considered [16] where PD controller with pulse-width pulse-frequency modulation with positive position feedback is considered for vibration reduction during on-off operation of thrusters. However, as seen from simulation results chattering occurs in control action. A comparison between linear and sliding mode controllers with reaction wheels is studied [17] where only small angle orientations are considered. Designed sliding mode controller stabilizes spacecraft attitude dynamics 30 times faster than output feedback controller with reaction wheels. Station keeping chattering free sliding mode controller design is designed in [18].

A body orbiting the Earth in geosynchronous orbit has instabilities in attitude dynamics and disturbances caused by the Earth, the Moon, the Sun and other bodies in space. These effects force the body to lose initial orbit and attitude. Here the control system takes important part in spacecraft missions where it keeps the body in designed orbit and desired attitude. The commonly used control elements for a spacecraft in geosynchronous orbit are thrusters, reaction, and momentum wheels. Dynamic model of a spacecraft is nonlinear, includes the rigid and flexible mode interaction, and the parameters of the spacecraft are not precisely known. The performance criteria for a spacecraft are fuel expenditure and vibration of flexible structures. The sliding mode technique permits usage of lower order system model for generating control commands, which includes unmodeled dynamics or uncertainties, and stabilizes the plant faster and robustly under bounded disturbance. The chattering at high frequencies is not desired because it may cause vibration. Chattering may be eliminated by replacing saturation instead of signum function. However, in that case non-zero tracking errors exist, which can be made small by taking a tiny region for saturation and also, saturation is limited with hardware capability and reduction of accuracy and robustness as introduced [7] and [8]. On the other hand, chattering may be eliminated by pulse modulation as done [14].

The chapter is organized as follows. Section 1 gives an introduction to sliding mode control of a satellite. Section 2 gives the system description of rigid body nonlinear attitude

dynamics. Section 3 evaluate rigid body in circular orbit with internal torquers and introduces equation of motion for a flexible spacecraft. Section 4 includes design of variable structure control systems for nonlinear attitude dynamics of a spacecraft and suspension of vibration of flexible solar arrays. Also design examples and performances comparison are studied. Section 5 concludes the chapter.

2. Rigid body dynamics

Consider a rigid-body with as body-fixed reference frame B with its origin at the center of mass of the rigid body as shown in Figure 1. \vec{R}_C is the position vector of the center of mass from an inertial origin of N , and \vec{R} is the position vector of dm from an inertial origin N .

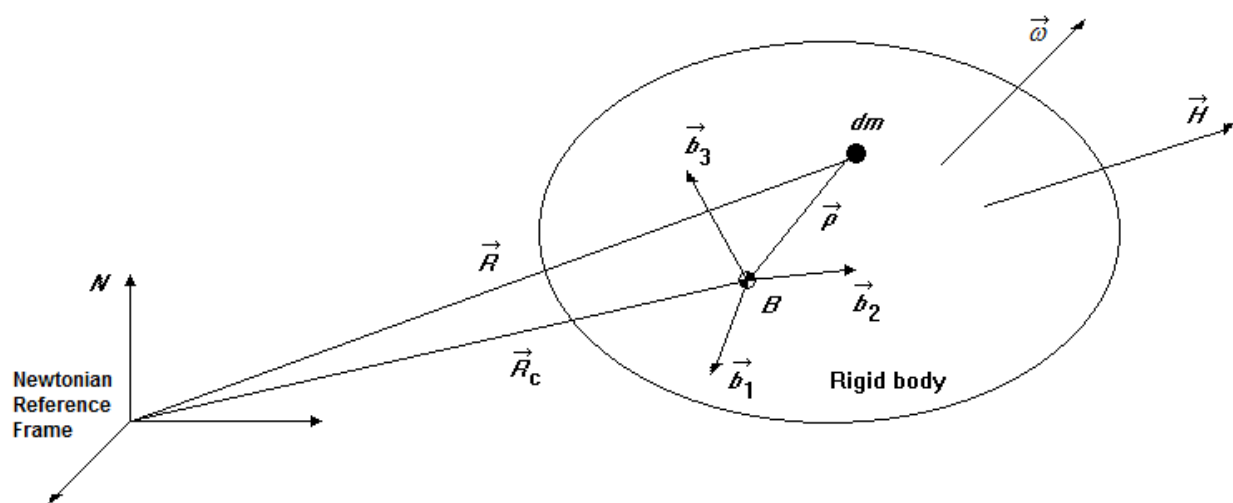


Fig. 1. Body fixed reference frame B at the center of mass of a rigid body.

Let $\vec{\omega} \equiv \vec{\omega}^{B/N}$ be the angular velocity vector of the rigid body in an inertial reference frame N . The angular momentum vector \vec{H} of rigid body about its center of mass can be defined as [9]:

$$\vec{H} = \int \vec{\rho} \times \dot{\vec{R}} dm = \int \vec{\rho} \times (\vec{\omega} \times \vec{\rho}) dm = I \vec{\omega} \quad (1)$$

The position vector $\vec{\rho}$ of very small mass element dm from the center of mass is defined as

$$\vec{\rho} = \rho_1 \vec{b}_1 + \rho_2 \vec{b}_2 + \rho_3 \vec{b}_3 \quad (2)$$

and finally angular velocity vector $\vec{\omega} \equiv \vec{\omega}^{B/N}$ of a rigid body in an inertial reference frame N can be written as:

$$\vec{\omega} = \omega_1 \vec{b}_1 + \omega_2 \vec{b}_2 + \omega_3 \vec{b}_3 \quad (3)$$

The angular momentum vector (1) can be rewritten as:

$$\vec{H} = H_1 \vec{b}_1 + H_2 \vec{b}_2 + H_3 \vec{b}_3 \quad (4)$$

where

$$\begin{aligned}
 H_1 &= I_{11}\omega_1 + I_{12}\omega_2 + I_{13}\omega_3 \\
 H_2 &= I_{21}\omega_1 + I_{22}\omega_2 + I_{23}\omega_3 \\
 H_3 &= I_{31}\omega_1 + I_{32}\omega_2 + I_{33}\omega_3
 \end{aligned} \tag{5}$$

and the matrix form of (5) is:

$$\begin{bmatrix} H_1 \\ H_2 \\ H_3 \end{bmatrix} = \begin{bmatrix} I_{11} & I_{12} & I_{13} \\ I_{21} & I_{22} & I_{23} \\ I_{31} & I_{32} & I_{33} \end{bmatrix} \begin{bmatrix} \omega_1 \\ \omega_2 \\ \omega_3 \end{bmatrix} \tag{6}$$

The rotational equation of motion of a rigid body in an inertial reference frame N about center of mass is [9]:

$$\int \vec{\rho} \times \ddot{\vec{R}} dm = \vec{M} \tag{7}$$

Form (1) and (7) we can write a relation between external moment acting on the body about its mass center and angular momentum as [9]-[11]:

$$\vec{M} = \dot{\vec{H}} \tag{8}$$

The relation above with an angular momentum \vec{H} and external moment \vec{M} is the rotational equation of motion of a rigid body in a circular orbit [9]-[10]:

$$\vec{M} = \dot{\vec{H}} \equiv \left\{ \frac{d\vec{H}}{dt} \right\}_N = \left\{ \frac{d\vec{H}}{dt} \right\}_B + \vec{\omega}^{B/N} \times \vec{H} \tag{9}$$

Substituting (1) into rotational equation of motion (9) we obtain:

$$\vec{M} = I \cdot \dot{\vec{\omega}} + \vec{\omega} \times I \cdot \vec{\omega} \tag{10}$$

which is known as Euler's rotational equation of motion.

3. Rigid body in circular orbit

Consider a rigid body orbiting the Earth with a constant radius. A local horizontal and local vertical reference frame A at the center of the mass of an orbiting spacecraft with unit vectors $\vec{a}_1, \vec{a}_2, \vec{a}_3$ as given in Figure 2. \vec{a}_1 is along the orbital direction, \vec{a}_2 is perpendicular to the orbital plane and \vec{a}_3 is always pointing the Earth. The angular velocity of A with respect to N is [10]:

$$\vec{\omega}^{A/N} = -n\vec{a}_2 \tag{11}$$

where n is the orbital rate defined as

$$n = \sqrt{\mu/R_C^3} \tag{12}$$

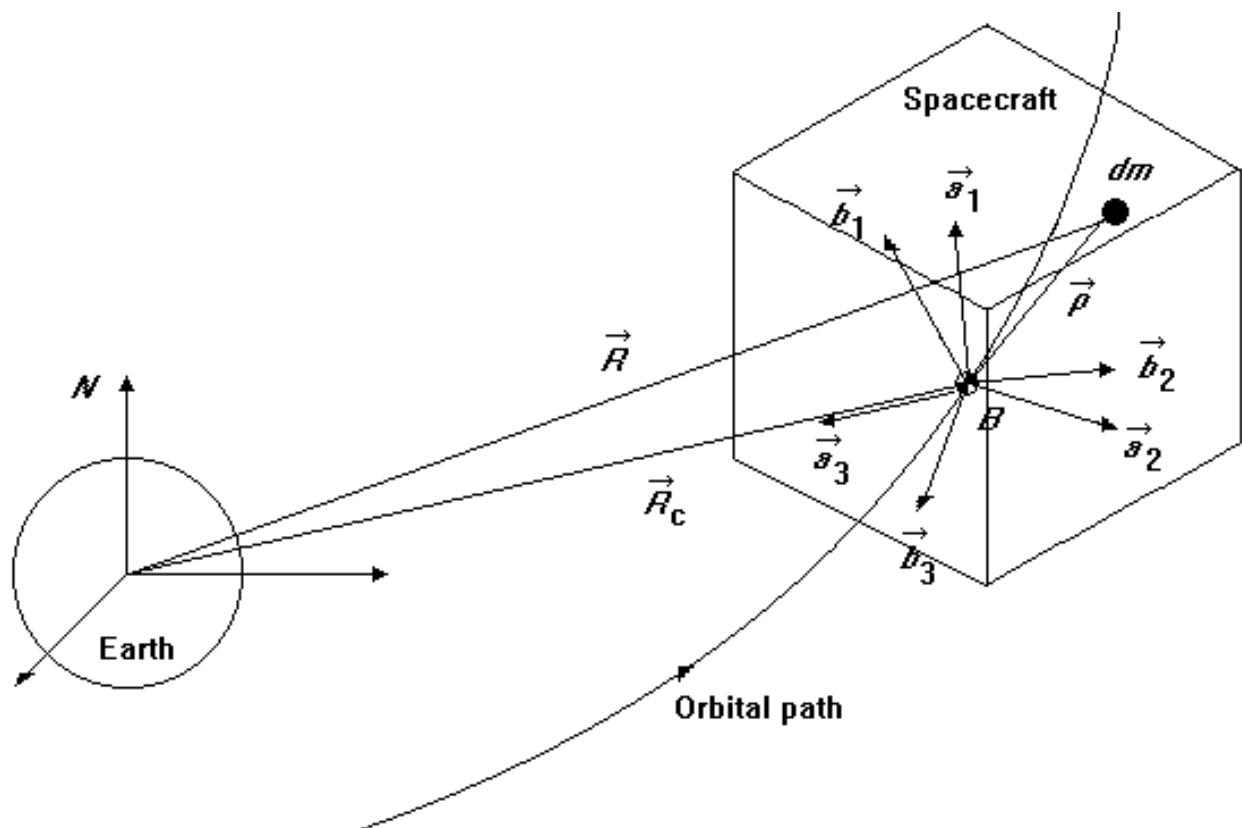


Fig. 2. Rigid body in circular orbit.

Note that, μ is the gravitational constant of the Earth and R_C is the radius of the orbit. The orbital rate for a spacecraft orbiting the Earth in a circular orbit with same angular velocity for one real day can be calculated from relation:

$$n = \frac{2\pi}{23h + 56m + 4.09054s} = 7.2921 \times 10^{-5} s^{-1} \quad (13)$$

The angular velocity of the body-fixed reference frame B with basis vectors $\vec{b}_1, \vec{b}_2, \vec{b}_3$ is given as [9]:

$$\vec{\omega}^{B/N} = \vec{\omega}^{B/A} + \vec{\omega}^{A/N} = \vec{\omega}^{B/A} - n\vec{a}_2 \quad (14)$$

where $\vec{\omega}^{B/A}$ is the angular velocity of B relative to A . To describe orientation of the body-fixed reference frame B with respect to the local vertical local horizontal reference frame A in terms of Euler angles θ_i ($i=1,2,3$), consider the rotational sequence of $C_1(\theta_1) \leftarrow C_2(\theta_2) \leftarrow C_3(\theta_3)$ to B from A . For this sequence relation is:

$$\begin{bmatrix} \vec{b}_1 \\ \vec{b}_2 \\ \vec{b}_3 \end{bmatrix} = \begin{bmatrix} c_2 c_3 & c_2 s_3 & -s_2 \\ s_1 s_2 c_3 - s_1 s_3 & s_1 s_2 s_3 - c_1 c_3 & s_1 c_2 \\ c_1 s_2 c_3 + s_1 s_3 & c_1 s_2 s_3 - s_1 c_3 & c_1 c_2 \end{bmatrix} \begin{bmatrix} \vec{a}_1 \\ \vec{a}_2 \\ \vec{a}_3 \end{bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} \\ C_{21} & C_{22} & C_{23} \\ C_{31} & C_{32} & C_{33} \end{bmatrix} \begin{bmatrix} \vec{a}_1 \\ \vec{a}_2 \\ \vec{a}_3 \end{bmatrix} \quad (15)$$

where $s_i = \sin \theta_i$, $c_i = \cos \theta_i$, $i=1,2,3$.

The gravity gradient torque can be written as [9]:

$$\vec{M}_G = -\mu \int \frac{\vec{\rho} \times \vec{R}_C}{|\vec{R}_C + \vec{\rho}|^3} dm = 3n^2 \vec{a}_3 \times I \cdot \vec{a}_3 \quad (16)$$

The basis vector on axis 3 of local horizontal and local vertical reference frame A is $\vec{a}_3 \equiv -\vec{R}_C/R_C$. The rotational equation of motion of a rigid body with an angular momentum in a circular orbit can be obtained by substituting $\vec{\omega}^{B/N} \equiv \vec{\omega}$ into (9) as

$$I \cdot \dot{\vec{\omega}} + \vec{\omega} \times I \cdot \vec{\omega} = 3n^2 \vec{a}_3 \times I \cdot \vec{a}_3 \quad (17)$$

$\vec{\omega}$ and \vec{a}_3 can be expressed in terms of basis vectors of the body-fixed reference frame B as:

$$\vec{\omega} = \omega_1 \vec{b}_1 + \omega_2 \vec{b}_2 + \omega_3 \vec{b}_3 \quad (18)$$

$$\vec{a}_3 = C_{13} \vec{b}_1 + C_{23} \vec{b}_2 + C_{33} \vec{b}_3 \quad (19)$$

The equation of motion with control torque and disturbance can be written as [5], [7], [9], [17]:

$$I \dot{\vec{\omega}} + \Omega I \vec{\omega} = M_G + u_B + d_E(t) \quad (20)$$

where u_B is the body (3×1)-control vector and $d_E(t)$ is an external disturbance (solar radiation, interaction with other bodies in space, etc.). Let us define gravity gradient torque (16) as

$$M_G = 3n^2 \Gamma C \quad (21)$$

where C is the third column of direction cosine (3×3)-dimensional matrix given in (15), $\Omega = \Omega(\omega)$ and $\Gamma = \Gamma(C)$ are (3×3)-dimensional skew symmetric matrices defined as:

$$C = [C_{13} \ C_{23} \ C_{33}]^T = [-s_2 \ s_1 c_2 \ c_1 c_2]^T \quad (22)$$

$$\Omega = \begin{bmatrix} 0 & -\omega_3 & \omega_2 \\ \omega_3 & 0 & -\omega_1 \\ -\omega_2 & \omega_1 & 0 \end{bmatrix} \quad (23)$$

$$\Gamma = \begin{bmatrix} 0 & -C_{33} & C_{23} \\ C_{33} & 0 & -C_{13} \\ -C_{23} & C_{13} & 0 \end{bmatrix} \quad (24)$$

Relation between angular velocity ω and attitude angles and their rates for circular orbit can be written as [9]:

$$\begin{bmatrix} \omega_1 \\ \omega_2 \\ \omega_3 \end{bmatrix} = \begin{bmatrix} 1 & 0 & -s_2 \\ 0 & c_1 & s_1 c_2 \\ 0 & -s_1 & c_1 c_2 \end{bmatrix} \begin{bmatrix} \dot{\theta}_1 \\ \dot{\theta}_2 \\ \dot{\theta}_3 \end{bmatrix} - n \begin{bmatrix} c_2 s_3 \\ s_1 s_2 s_3 + c_1 c_3 \\ c_1 s_2 s_3 - s_1 c_3 \end{bmatrix} \cong n_1 \begin{bmatrix} \dot{\theta}_1 \\ \dot{\theta}_2 \\ \dot{\theta}_3 \end{bmatrix} + n_3 \quad (25)$$

or vice versa of (25):

$$\begin{bmatrix} \dot{\theta}_1 \\ \dot{\theta}_2 \\ \dot{\theta}_3 \end{bmatrix} = \frac{1}{c_2} \begin{bmatrix} c_2 & s_1 s_2 & c_1 s_2 \\ 0 & c_1 c_2 & -s_1 c_2 \\ 0 & s_1 & c_1 \end{bmatrix} \begin{bmatrix} \omega_1 \\ \omega_2 \\ \omega_3 \end{bmatrix} + \frac{n}{c_2} \begin{bmatrix} s_3 \\ c_2 c_3 \\ s_2 s_3 \end{bmatrix} \cong n_2 \begin{bmatrix} \omega_1 \\ \omega_2 \\ \omega_3 \end{bmatrix} + n_4 \quad (26)$$

Then the nonlinear equation of motion of three axes stabilized rigid body spacecraft (20) reduces to:

$$I n_1 \ddot{\theta} + (\dot{I} n_1 - \Omega I n_1) \dot{\theta} - (n(I \dot{n}_3 - n_3) + 3n^2 \Gamma I C) = u_B + d_E(t) \quad (27)$$

where

$$\dot{n}_1 = \begin{bmatrix} 0 & 0 & -c_2 \\ 0 & -s_1 & c_1 c_2 - s_1 s_2 \\ 0 & -c_1 & -c_2 s_1 - c_1 s_2 \end{bmatrix} \quad (28)$$

$$\dot{n}_3 = \begin{bmatrix} c_2 c_3 - s_2 s_3 \\ c_1 s_2 s_3 + c_2 s_1 s_3 + c_3 s_1 s_2 - c_1 s_3 - c_3 s_1 \\ c_1 c_2 s_3 + c_1 c_3 s_2 - s_1 s_2 s_3 + s_1 s_3 - c_1 c_3 \end{bmatrix} \quad (29)$$

Note that, (27) will be used in simulation of attitude dynamics.

3.1 Rigid spacecraft with internal torquers in circular orbit

Here we have considered a three axis stabilized communication satellite with a bias momentum wheel. Some parameters of considered communication satellite are given in Table 1. Internal control torquers for the satellite are reaction wheels mounted on roll and yaw axes, and a momentum wheel is set up on pitch axis which spins along negative direction, see Figure 3. The total angular momentum of spacecraft can be written as [9]:

$$\vec{H} = (H_1 + h_1) \vec{b}_1 + (H_2 - H_0 + h_2) \vec{b}_2 + (H_3 + h_3) \vec{b}_3 \quad (30)$$

where H_1, H_2, H_3 were defined in (5). We can obtain equation of motion for principle axis frame B from rotational equation of motion (9) with considered gravity gradient torque (21), external disturbances and internal torquers as:

$$I \dot{\omega} + \Omega I \omega = M - (\dot{h} + \Omega h - \Omega H_0) + d_E(t) \quad (31)$$

Or in term of attitude angles

$$I n_1 \ddot{\theta} + (\dot{I} n_1 - \Omega I n_1) \dot{\theta} - (n(I \dot{n}_3 - n_3) + 3n^2 \Gamma I C) = -(\dot{h} + \Omega h - \Omega H_0) + d_E(t) \quad (32)$$

Properties	Values	Units
Principle moments of inertias, I_{11}, I_{22}, I_{33}	3026 / 440 / 3164	kg.m ²
Main body dimensions, x-y-z.....	1.5 / 1.7 / 2.2	m
Solar arrays (tip-to-tip).....	20	m
Maximum thrust force of thrusters.....	10	N
Bias momentum.....	91.4	Nms
Array power.....	1.5	kW
Liquid of bi-propellant thrusters.....	N ₂ O ₄ /MMH	-

Table 1. Spacecraft Parameters [9].

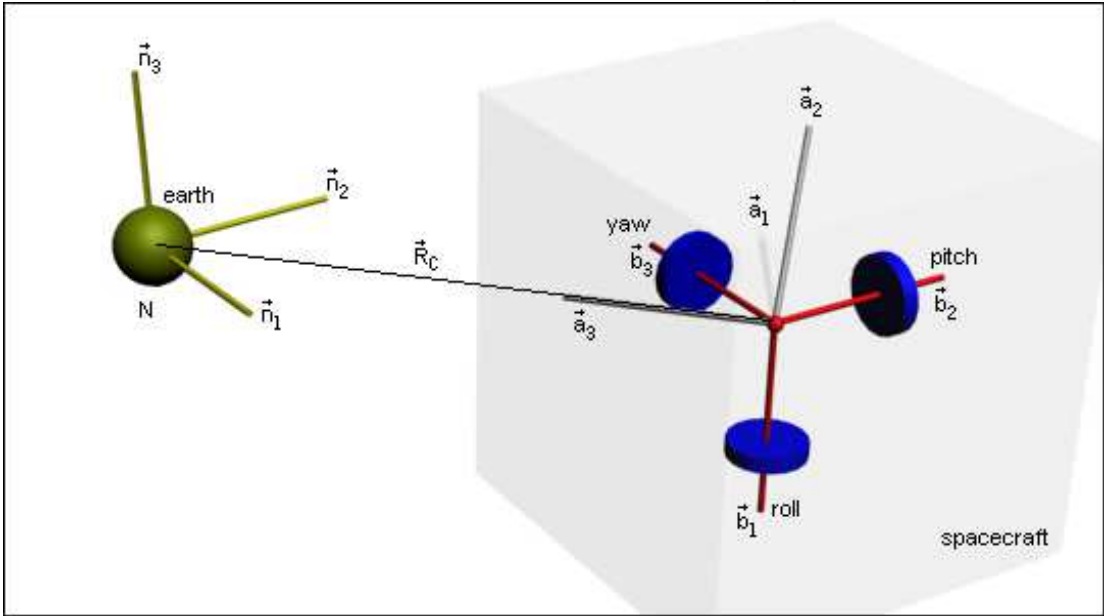


Fig. 3. Rigid spacecraft with internal torquers.

3.2 Communication satellite: rigid body with flexible solar arrays

During on-orbit normal mode operations, both solar arrays always point towards the sun, whereas the main body points towards the Earth. This results in a very slow change of modal frequencies and modal shapes. For control design purposes, however, the spacecraft model will be treated as a time-invariant but nonlinear system with a known range of modal characteristics. The equation of attitude motion of the three axis stabilized with flexible solar array is given [9]-[16]. Extending the rigid body equation (27) with the flexible solar arrays, following set of equations can be obtained for main body:

$$In_1\ddot{\theta} + (\dot{In}_1 - \Omega In_1)\dot{\theta} - \left(n(\dot{In}_3 - n_3) + 3n^2\Gamma IC \right) + \sqrt{2}\delta\ddot{q} = u_B + d_E(t)$$

(33)

or in terms of ω

$$I\dot{\omega} + \Omega I\omega + \sqrt{2}\delta\ddot{q} = 3n^2\Gamma IC + u_B + d_E(t)$$

(34)

and two solar arrays [9] with control force:

$$\ddot{q} + \sigma^2 q + \sqrt{2} \delta \ddot{\theta} = u_{SA}$$

(35)

where $\delta = diag(\delta_1, \delta_2, \delta_3)$ represents rigid-elastic coupling diagonal matrix of a single solar array (see Table 2), $\sigma = diag(\sigma_1, \sigma_2, \sigma_3)$ are the modal frequencies diagonal matrix, and $q = [q_1 \ q_2 \ q_3]^T$ are the modal coordinates, and u_{SA} is the control command produced by solar array drivers. Note that, in this study only the first modes of modal characteristics of flexible solar arrays are taken into account.

Cantilever mode description	Cantilever frequency σ , rad/s	Coupling scalars, $\sqrt{kg \cdot m^2}$		
		Roll, δ_1	Pitch, δ_2	Yaw, δ_3
OP-1	0.885	0	0	35.372
OP-2	6.852	0	0	4.772
OP-3	16.658	0	0	2.347
OP-4	33.326	0	0	0.548
T-1	5.534	0	2.532	0
T-2	17.668	0	0.864	0
T-3	33.805	0	0.381	0
IP-1	1.112	35.865	0	0
IP-2	36.362	2.768	0	0

[1]^a: OP is out-of plane, T is torsion and IP is in-plane

Table 2. Single solar array modes at 6 a.m. [9].

Note that, two solar arrays point towards the Sun. Hence, the solar driving mechanism actuated by Sun Sensors causes the control torque to point solar arrays towards the Sun. In our case we have considered an station keeping controller that can activate the solar array driver mechanism to suspend vibration of flexible solar arrays caused by attitude angle acceleration during maneuvering. The control law will be designed in Section 4.2.

4. Sliding mode control

The sliding mode technique permits usage of lower order system model for generating control commands, which includes unmodeled dynamics or uncertainties, and stabilizes the plant faster and robustly under bounded disturbance. The chattering at high frequencies is not desired because it may cause vibration. Chattering may be eliminated by replacing saturation instead of signum function. However, in that case non-zero tracking errors exist, which can be made small by taking a tiny region for saturation and also, saturation is limited with hardware capability and reduction of accuracy and robustness as introduced [7] and [8]. On the other hand, chattering may be eliminated by pulse modulation as done [16]. In this section we suggest variable structure attitude and station keeping control system design for a communication satellite.

4.1 Sliding mode attitude control system design of a rigid spacecraft

Let s represent a sliding manifold as

$$s = [s_1 \dots s_m]^T = \omega + k\theta = 0$$

(36)

where k is a constant to be selected. The attitude dynamics dominated on the sliding manifold $s = 0$ can be written from [5], [17] as:

$$\dot{\theta} = \begin{bmatrix} -k\theta_3\theta_2 + n\theta_3 - k\theta_1 \\ k\theta_3\theta_1 - k\theta_2 + n \\ -k\theta_2\theta_1 - k\theta_3 \end{bmatrix} = T\theta \quad (37)$$

Assume that the discontinuous control is given by

$$u_B = -I(N_1s + N_2\text{sign}(s)) \quad (38)$$

where N_1, N_2 are constant parameters to be selected. To analyze the stability of the system, consider a Lyapunov function candidate

$$V = \frac{1}{2}s^Ts \quad (39)$$

Then,

$$\begin{aligned} \dot{V} &= s^T\dot{s} = s^T(\dot{\omega} + k\dot{\theta}) \\ &= s^T(-I^{-1}\Omega I\omega + 3n^2I^{-1}\Gamma IC + u + I^{-1}d(t) + k\dot{\theta}) \\ &= s^T(-I^{-1}\Omega Ik\theta + 3n^2I^{-1}\Gamma IC + u + I^{-1}d(t) + kT\theta) \end{aligned} \quad (40)$$

Taking the norm of (40) we have

$$\dot{V} \leq k\|s\|\|I^{-1}\Omega I\|\|\theta\| + 3n^2\|s\|\|I^{-1}\Gamma IC\| + \|s\|\|I^{-1}\| \|d(t)\| - \|s\|N_1\|s\| - \|s\|N_2 \quad (41)$$

Some matrix and vector norms in (41) satisfy inequalities as below:

$$k\|I^{-1}\Omega I\| \leq L_0 \quad (42)$$

$$3n^2\|I^{-1}\Gamma IC\| \leq M_0 \quad (43)$$

$$\|I^{-1}d_E(t)\| \leq d_0 \quad (44)$$

Therefore, substituting norm values (42)-(44) into (41) we obtain:

$$\begin{aligned} \dot{V} &\leq -(N_1\|s\| - L_0\|\theta\|)\|s\| - (N_2 - M_0 - d_0)\|s\| \\ &< 0 \end{aligned} \quad (45)$$

Hence, the sliding mode controller forces the system trajectories toward the sliding manifold asymptotically for $N_2 > d_0 + M_0$ and $N_1 > L_0\|\theta\|/\|s\|$.

4.1.1 Example

Sliding mode control system design is performed firstly with determining proper switching function where the system trajectory is caused to follow the sliding manifold, $s = 0$. Then proposed discontinuous control term is employed to model reaction wheels. Physical sliding surface consists of inputs from the Earth sensor for yaw and pitch attitude angles, from the star tracker for yaw, roll and pitch attitude angles, and from the rate gyro for attitude angle rates. Considered control function (38) and sliding manifold (36) stabilizes the dynamic equation of attitude angles presented via (31) for small attitude angles errors $\theta_1 = 0.3^\circ$, $\theta_2 = 0.5^\circ$, $\theta_3 = -0.3^\circ$ gravity gradient torque and bounded external disturbances as shown in Figure 4.

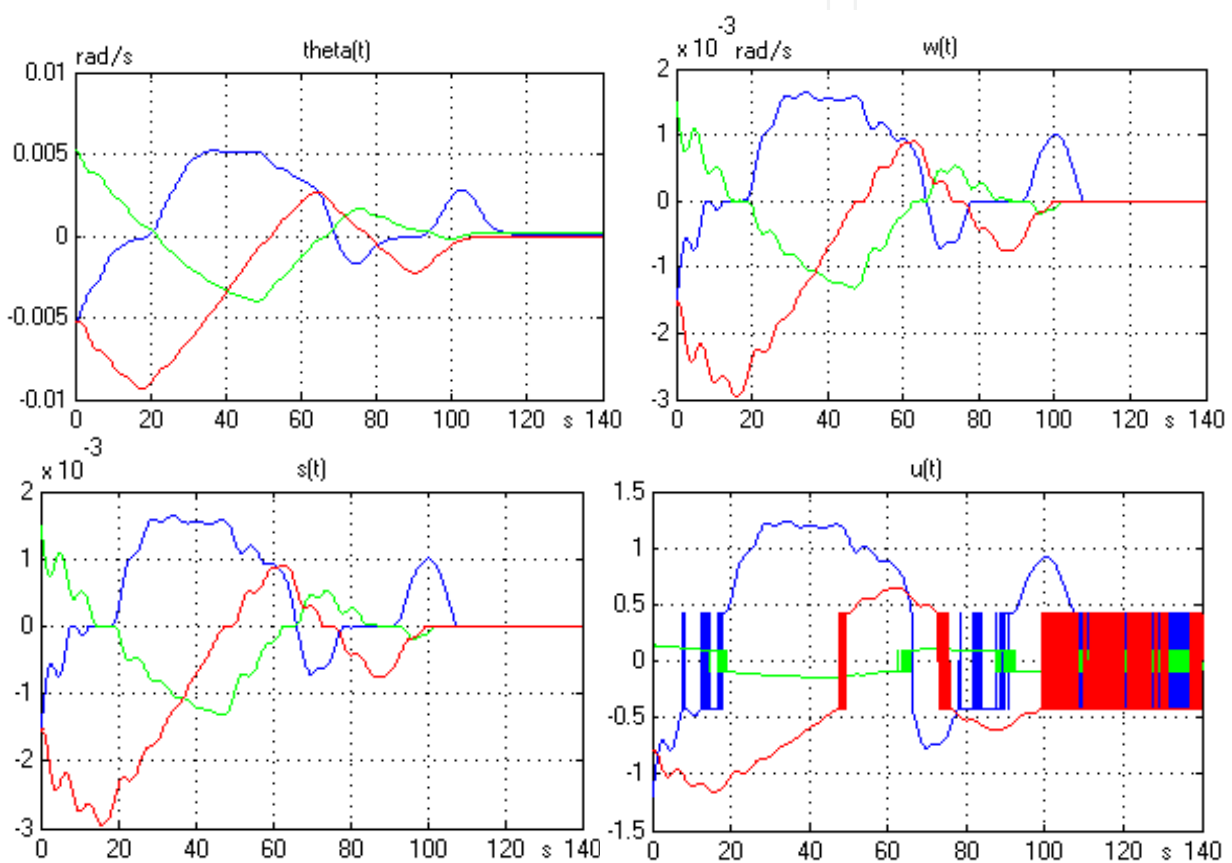


Fig. 4. Time responses of attitude sliding mode control system.

Required parameter k is selected as $k = 0.3$. The disturbance is assumed to be as:

$$d_E(t) = 0.0005 \sin(\omega t) \quad (46)$$

The control torque is produced by reaction wheels which have approximately 6000 rpm angular velocity and can produce 1.5 Nm control torque. Therefore, control inputs can be developed for each pitch, roll and yaw axes by sliding mode control approach. Thus attitude sliding mode controller can be obtained for $N_2 > d_0 + M_0$ and $N_1 > L_0 \|\theta\|/\|s\|$ as:

$$u_B = -J(1000s + \text{sign}(s)) \quad (47)$$

where J is the inertia matrix of reaction wheels.

As seen from Figure 4, spacecraft attitude errors are eliminated in 120 seconds by sliding mode control approach where some chattering appears in control action. Since control function is applied to electro-motor of reaction wheel, the electro-motors are driven at nominal speeds on chattering phenomena for which sliding motion is conventional [2]. In general, chattering effect can be eliminated by using a saturation element.

4.2 Discontinuous station keeping sliding mode control design of flexible spacecraft

Station keeping sliding mode control algorithm for a three axes stabilized geosynchronous communication satellite is considered in this subsection. The spacecraft is assumed to be controlled externally with small attitude thrust jets. Classical sliding mode technique with chattering free correction and elimination of operation for small attitude angles via dead-band function will be used to model thrust jets variable on-off operation for stabilization of the spacecraft. The performances, modeling and simulation are discussed on a design example by using MATLAB-Simulink programming. The attitude sliding mode controller for geosynchronous satellite with flexible solar arrays will use fuel optimally and adequately as little as possible with proposed control algorithm. Also, attitude sliding mode controller is robust to bounded external disturbances and includes unmodelled dynamics as well.

Two types of simple and easy-to-apply variable structure P+relay controllers different from existing (for example from [5] which includes the linear equivalent and sliding terms plus relay) are proposed for the stabilization of full nonlinear attitude dynamics and vibration control of flexible solar arrays during station keeping maneuvering. Variable structure P+relay control law has only two design parameters. A modified sliding function with a dead-band instead of conventional one is considered to reduce fuel expenditure for small attitude corrections that can be stabilized by internal torquers (reaction or momentum wheels). The size of dead-band provides the capability of a tradeoff between maneuver time and fuel expenditure. The limits of the dead-band of switching function can be determined from maximum available torque produced by reaction wheels. On the other hand, large angle orientation of spacecraft induces structural deformation in the flexible solar arrays. Dynamical models of satellites are nonlinear and include rigid and flexible mode interaction. Therefore, vibration suppression of flexible solar arrays is required. For this case variable structure P+relay algorithm is proposed to eliminate vibration of flexible solar panels.

Desired sliding manifold on which the system equation of motion has good transient performances need to be selected before form a control law. The switching surfaces can be selected [1]-[5], [7], [11], [13]:

$$s = \omega + K\theta \quad (48)$$

where $K = \text{diag}(k_1, k_2, k_3)$, in general is a diagonal design matrix to be selected. Particularly, these parameters are selected $k_1 = k_2 = k_3 = k$. Then the sliding manifold is:

$$s = \omega + k\theta \quad (49)$$

After selecting sliding manifold, a variable structure P+relay control algorithm can be formed as follows:

$$u_B = -I(N_1\|\theta\| + N_2)\text{sign}(s) \quad (50)$$

where N_1 and N_2 are design parameters to be selected. The control law consists of two terms: P term and relay term. The first term is used for the compensation of the model nonlinearities, model and parameter uncertainties. The second term is used to compensate the bounded external disturbances, flexibility effects of solar arrays, and gravity gradient torque. This controller should provide the existence of the sliding mode motion on the selected sliding manifold. So, consider a Lyapunov function candidate:

$$V = \frac{1}{2} s^T s \quad (51)$$

Now the sliding mode existence condition for the nonlinear satellite equation of motion in large will be investigated. The time derivative of (51) along the state trajectories of dynamics system defined by nonlinear equations (33) or (34) and (35) can be calculated as follows:

$$\begin{aligned} \dot{V} &= s^T \dot{s} = s^T (\dot{\omega} + k\dot{\theta}) = \\ &= s^T \left(I^{-1} \left[u_B + d_E(t) - \sqrt{2}\delta\ddot{q} + 3n^2\Gamma IC \right] - I^{-1}\Omega I n_1 \dot{\theta} + k\dot{\theta} \right) \end{aligned} \quad (52)$$

Taking the norm of (52) where

$$\|kT - kI^{-1}\Omega I\| \leq R_0 \quad (53)$$

$$\|3n^2 I^{-1}\Gamma IC\| \leq M_0 \quad (54)$$

$$\|I^{-1}d_E(t)\| \leq d_0 \quad (55)$$

we have

$$\begin{aligned} \dot{V} = s^T \dot{s} &\leq -(N_1 - R_0)\|\theta\|\|s\| - (N_2 - d_0 - \delta_0 - M_0)\|s\| \\ &\leq -(N_1 - R_0)\|\theta\|\|s\| - \eta\|s\| \end{aligned} \quad (56)$$

where

$$\eta = N_2 - d_0 - d_I \quad (57)$$

and the internal disturbance is:

$$d_I = \delta_0 - M_0 \quad (58)$$

For providing negativeness of the \dot{V} it is required that the following sliding mode existence conditions should be satisfied:

$$N_1 - R_0 \geq 0 \text{ or } N_1 \geq R_0 \quad (59)$$

$$\eta \geq 0 \text{ or } N_2 \geq d_0 + d_I \quad (60)$$

Moreover, from (53) and (54) after some evaluations it is easy to design sliding gain constant k as:

$$k \leq \sqrt{R_0/M_0} \quad (61)$$

In result, (56) can be evaluated as:

$$\dot{V} = s^T \dot{s} \leq -(N_1 - R_0) \|\theta\| \|s\| - \eta \|s\| \leq -\eta \|s\| < 0 \quad (62)$$

Therefore, the sliding manifold $s(t) = 0$ is reached in finite time [2]: $t_{s_1} \leq \|s(0)\|/\eta$.

4.2.1 Modification of switching function

Thrusters apply discontinuous external force for stabilization of the nonlinear attitude dynamics of the spacecraft in finite time with limited thrust force. Control system using attitude thrusters is operated for large attitude angle orientations and its faster stabilization. Note that, thrusters are not required to operate for small attitude correction. Hence, here sliding function s can be modified to two-side dead-band (see Figure 5) to stop thruster operation for small attitude errors:

$$s = \gamma(s) = \begin{cases} s - s_d & s \geq s_d \\ 0 & -s_d < s < s_d \\ s + s_d & s \leq -s_d \end{cases} \quad (63)$$

where $\pm s_d$ is the upper and lower limit of dead-band. Therefore, the control action (50) forces the system to the dead-band limits of sliding manifold $s \pm s_d = 0$ and keeps it in dead zone. As shown [4], dead-zones can have a number of possible effects on control system. Their most common effect is to decrease static output accuracy. They can actually stabilize a system or suppress self-oscillations. For example, if a dead-zone is incorporated into ideal relay, it may lead to the avoidance of the oscillation at the contact point of the relay.

However, the sliding mode existence conditions should be investigated for the following three cases. If $s \geq s_d > 0$, $N \geq R_0$, and $N_2 \geq d_0 + d_I$ then

$$\begin{aligned} \dot{V} &\leq -\|\gamma(s)\| (N_1 - R_0) \|\theta\| - \|\gamma(s)\| (N_2 - d_0 - d_I) \\ &\leq -\|s - s_d\| (N_1 - R_0) \|\theta\| - \|s - s_d\| (N_2 - d_0 - d_I) \\ &\leq -\eta \|s - s_d\| \end{aligned} \quad (64)$$

If $-s_d < s < 0$ or $s_d > s > 0$, which corresponds to no control action with $\gamma(s) = 0$ we have

$$\dot{V} = [R_0 \|\theta\| + (d_0 + d_I)] \|\gamma(s)\| \quad (65)$$

If $s \leq -s_d < 0$, $N_1 \geq R_0$, and $N_2 \geq d_0 + d_I$ then

$$\begin{aligned} \dot{V} &\leq -\|\gamma(s)\| (N_1 - R_0) \|\theta\| - \|\gamma(s)\| (N_2 - d_0 - d_I) \\ &\leq -\|s + s_d\| (N_1 - R_0) \|\theta\| - \|s + s_d\| (N_2 - d_0 - d_I) \\ &\leq -\eta \|s + s_d\| \end{aligned} \quad (66)$$

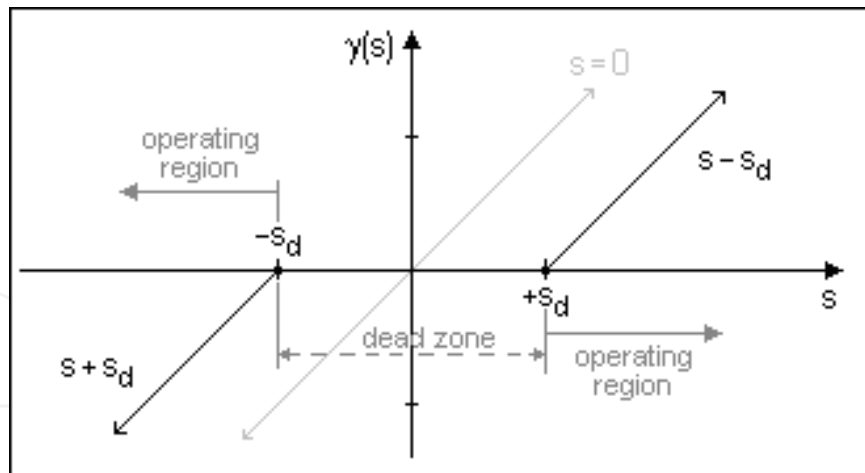


Fig. 5. Dead-band.

Note that, (64) and (66) are operating regime of the attitude sliding mode controller that produces external control torques for stabilization via thrusters. Thus, the controller (50) forces the system to reach dead-band on switching function at finite time. On the other hand, if sliding variable has a value between $-s_d \leq s \leq s_d$ then the sliding mode controller do not take any action. So, the system behavior is determined by attitude dynamics (33) with $u_B = 0$. Therefore, strictly, $\dot{V} \leq 0$ for two operating cases of $\gamma(s)$ (63). This means that system is convergent.

4.2.2 Example

The design of attitude sliding mode controller begins with selection of appropriate sliding manifold. A design parameter k can be selected from (61):

$$k_{\max} = \sqrt{\frac{R}{M_0}} = \sqrt{\frac{0.0189}{0.3029}} = 0.25 \quad (67)$$

$$k = 0.25 \leq k_{\max} \quad (68)$$

we chose $k = 0.25$ which usually gives better performance for sliding manifold [5]. Inertia matrix can be constructed form Table 1 as:

$$I = \begin{bmatrix} I_{11} & 0 & 0 \\ 0 & I_{22} & 0 \\ 0 & 0 & I_{33} \end{bmatrix} = \begin{bmatrix} 3026 & 0 & 0 \\ 0 & 440 & 0 \\ 0 & 0 & 3164 \end{bmatrix} \quad (69)$$

The design parameters N_1 and N_2 of attitude controller can be determined from conditions (59) and (60). Since control torque is limited by maximum available thrust and geometric configuration of thrusters, attitude controller parameters for a station keeping maneuver must satisfy the following physical condition:

$$I(N_1 \|\theta\| + N_2) \leq 10 \quad (70)$$

with bounded external disturbance:

$$d_E(t) = 0.5\sin(\omega_d t) \leq d_E \tag{71}$$

where ω_d is the frequency of external disturbance, and according to (55) Frobenius norm is $d_E = 0.5$. Consider that a station keeping maneuver at 6 a.m. is required with initial attitude errors as $\theta_1 = 0\text{deg}$, $\theta_2 = 10\text{deg}$ (0.1745rad), $\theta_3 = 0\text{deg}$. The variable structure controller parameters can be calculated for considered station keeping maneuver as:

$$N_1 = 0.07 \geq R = 0.0683$$
$$\|\theta\|_F = \left\| \begin{bmatrix} 0 & 0.1745 & 0 \end{bmatrix}^T \right\|_F = 0.1745$$
$$N_2 = 0.005 \geq \|I^{-1}d_E\| + \|d_I\| \cong 0.0048$$

$$(72)$$

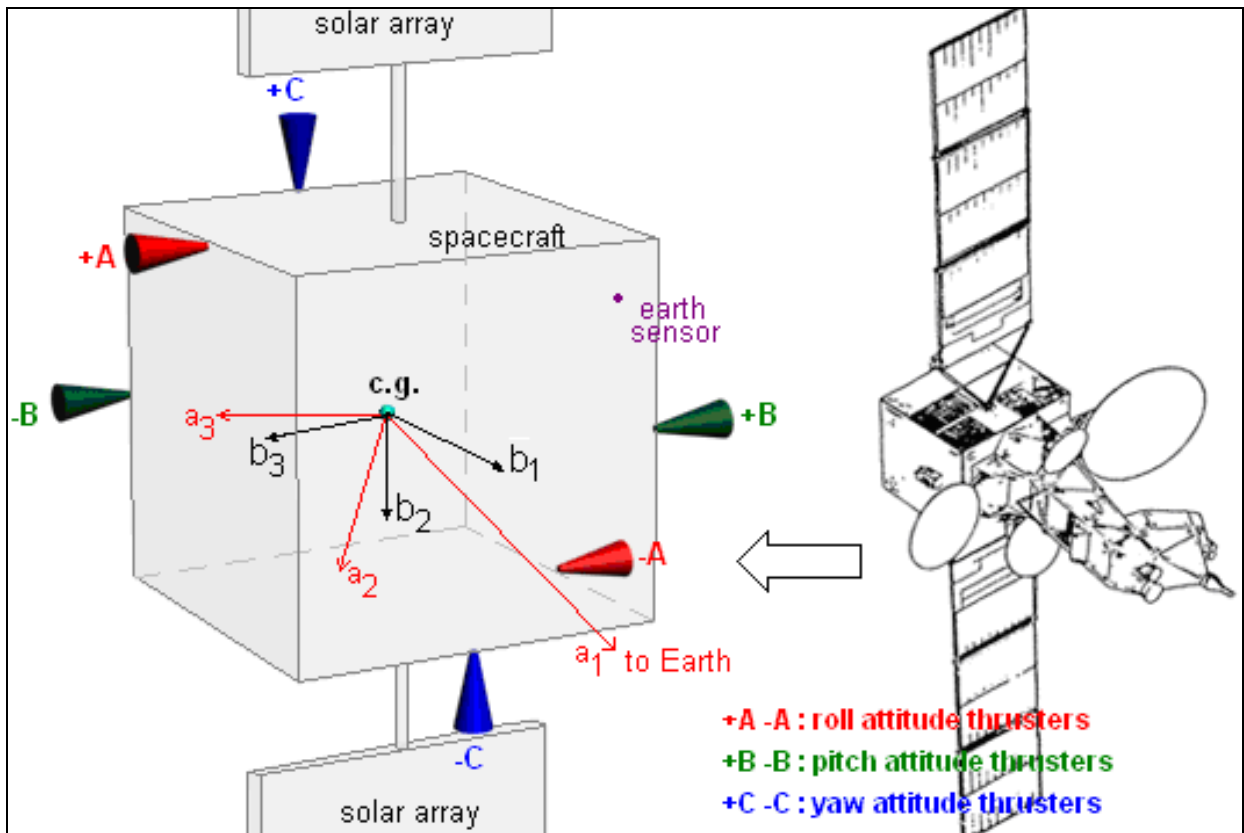


Fig. 6. Schematic view of geosynchronous satellite Intelsat-V and thrusters configuration: +A/-A corresponds to roll attitude thrusters, +B/-B corresponds to pitch attitude thrusters, and +C/-C corresponds to yaw attitude thrusters.

Note that, design parameters N_1 , N_2 and $\|\theta\|$ satisfy condition (59) and (60) for considered maneuver. Additionally, the dead-band limit is practically chosen according to thruster performance as $s_d = 0.012$. Therefore, variable structure control algorithm (50) with sliding manifold (49) can be formed as below. Also Figure 7 illustrates behavior of switching function with dead zone

$$u_B = -I(0.07\|\theta\| + 0.005)sign(\gamma(s))$$

$$\gamma(s) = \begin{cases} s - 0.012 & s \geq 0.012 \\ 0 & -0.012 < s < 0.012 \\ s + 0.012 & s \leq -0.012 \end{cases} \tag{73}$$

$$s = \omega + 0.25\theta$$

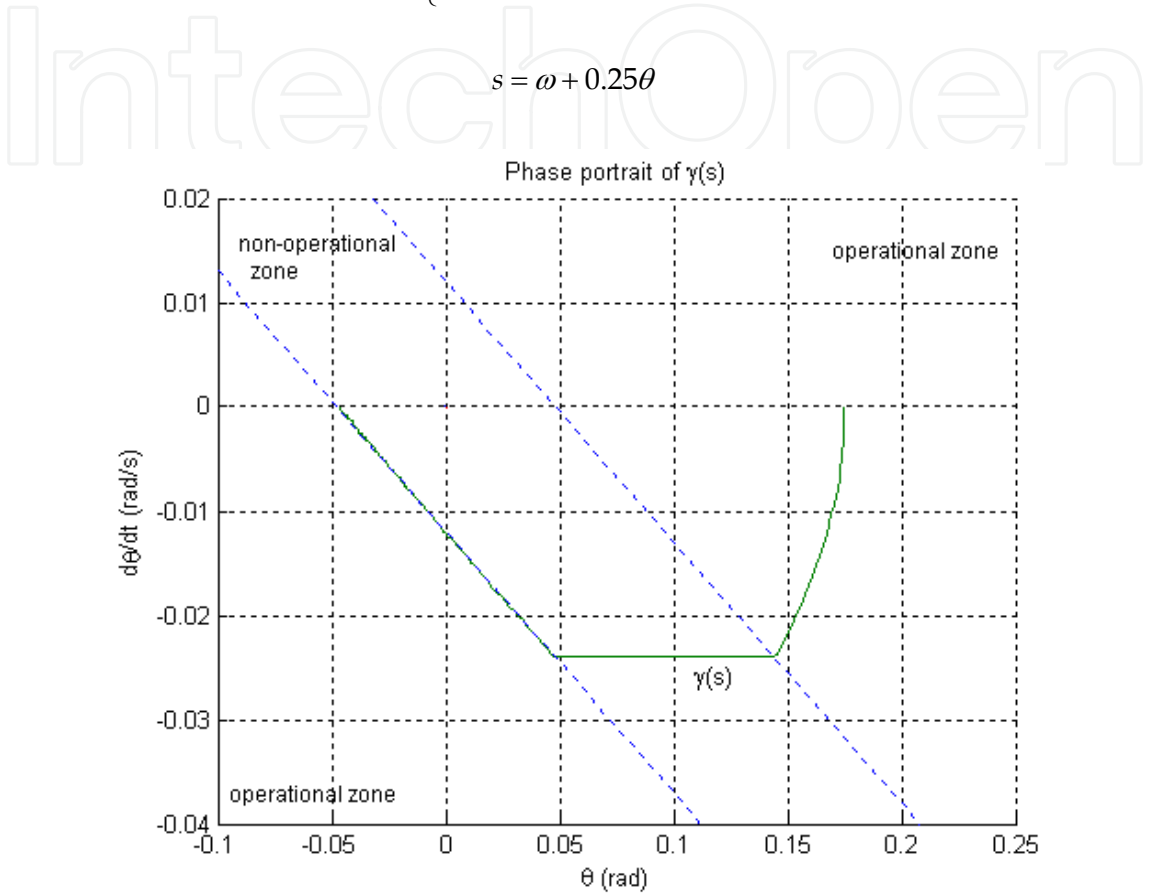


Fig. 7. Phase portrait of switching function of controlled body.

In this section nonlinear spacecraft dynamics (33) with flexible solar arrays (35) of a geosynchronous communication satellite are simulated with variable structure attitude and vibration controllers by Matlab-Simulink with iteration step of 0.1 seconds. Block diagram of satellite control system is shown in Figure 8. The time responses of attitude angles, angular velocities and accelerations; Sliding function and vibration control action and generated by solar array driving mechanism; modal coordinates are given in Figure 9. Note that, control command is illustrated in $(1/s^2)$.

As seen from Figure 9 attitude controller stabilizes the nonlinear model of flexible spacecraft approximately in 20 seconds and the sliding manifold is reached in 5 seconds at left side of dead zone. Vibration suppression of flexible solar arrays is achieved about 3-5 seconds and the sliding manifold is reached in 0.8 seconds. The station keeping attitude control performances and vibration suppressions with designed controllers are sufficient for faster stabilization and limited firings.

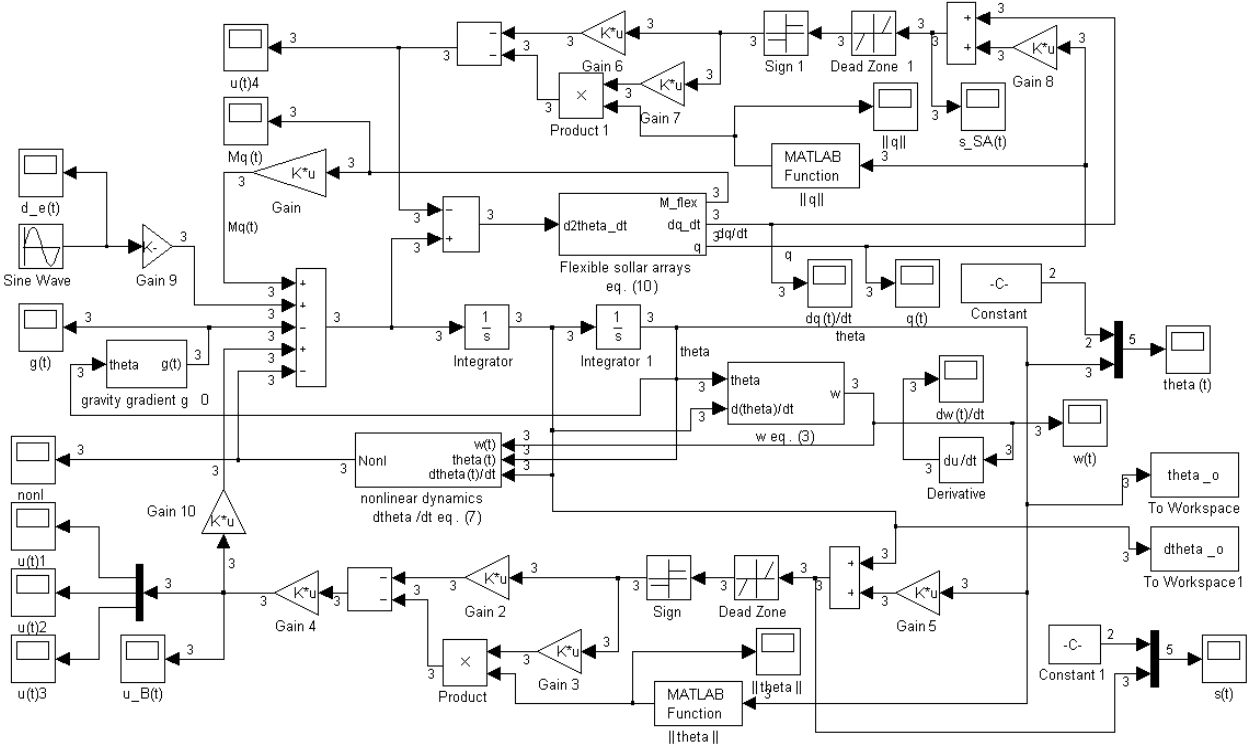
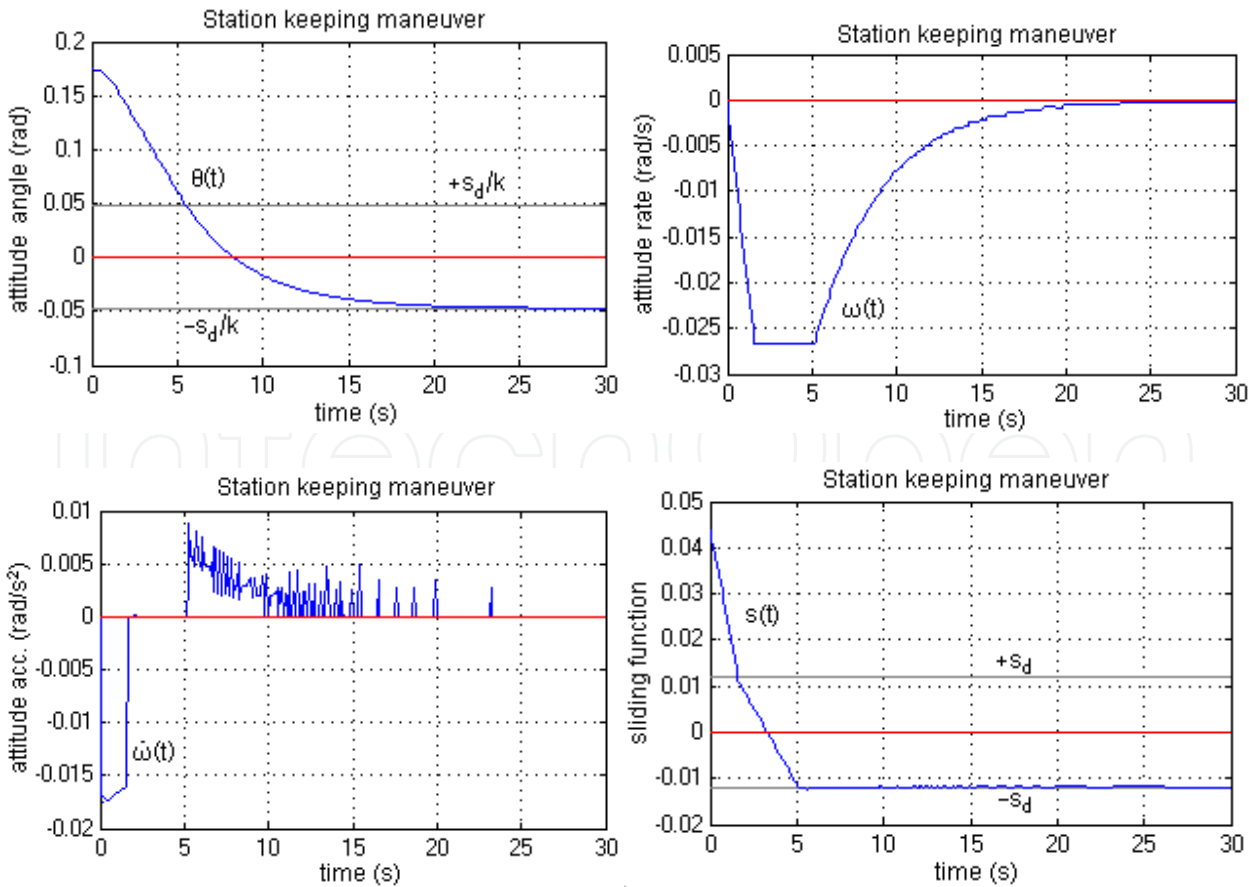


Fig. 8. Block diagram of satellite control system.



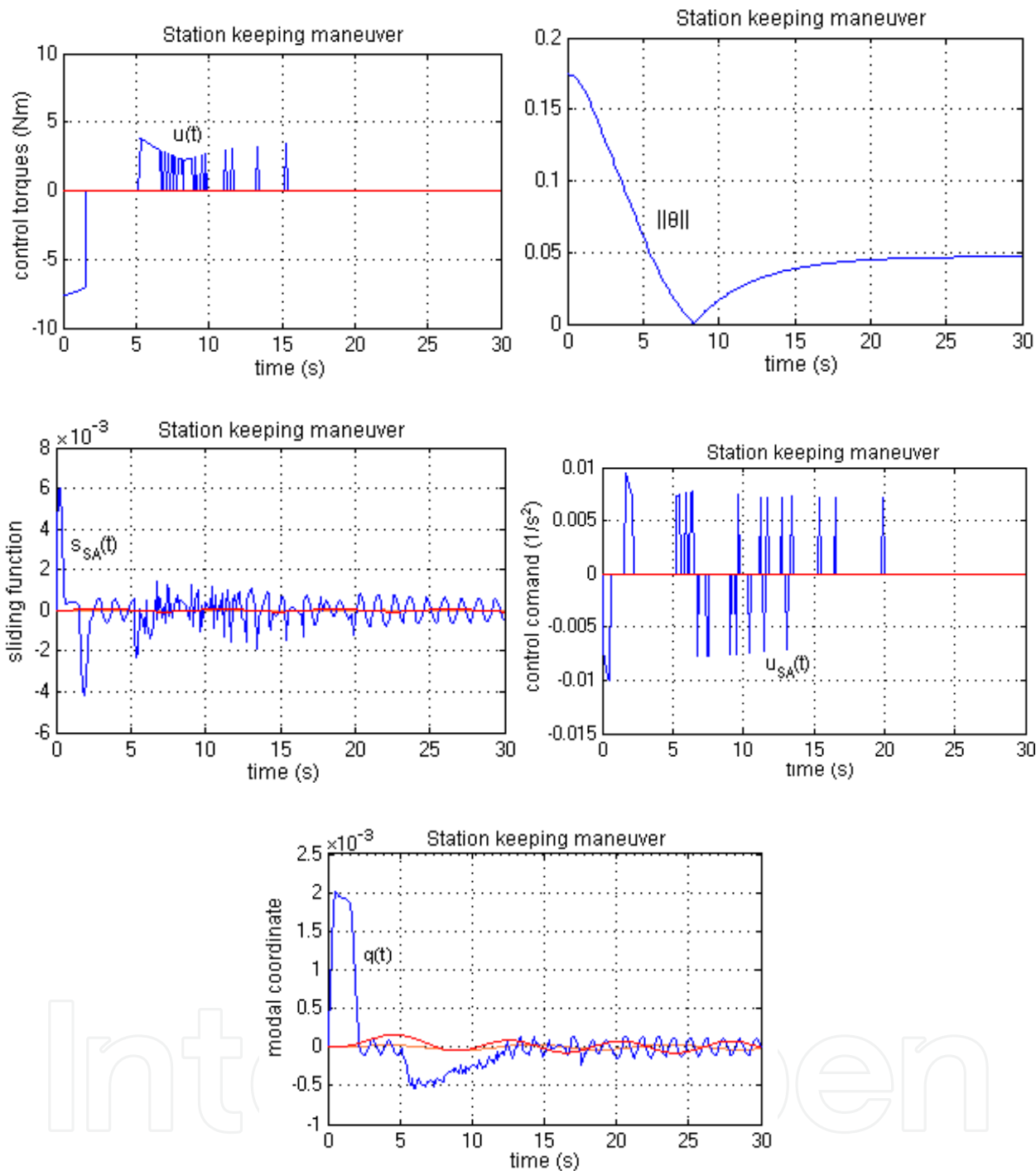


Fig. 9. Time responses of station keeping & vibration controller.

4.3 Comparison

Comparison of simulation results for both designed satellite control systems are done in the table. As seen from first part of Table 3 proposed variable structure controller in Section 4.1 has a large settling time then other considered attitude control systems. Since considered internal actuators has maximum 1.5 Nm torque capability, the settling time is three times shorter than [9] where same satellite (see Table 1) has been considered.

Design	Attitude controller properties			Vibration controller properties	
	controller-type	chattering	settling time	controller-type	settling time
[5]	VSC	no	45 s	-	-
[6]	VSC	-	60 s	-	-
[7]	VSC	no	50 s	-	-
[8]	VSC	yes	10 s	-	-
[9]	PID	-	300 s	-	-
[12]	VSC	yes	15-20 s	-	-
Sec. 4.1	VSC	yes	120 s	-	-
[11]	VSC	little	100 s	VSC	10-15 s
[13]	Adaptive VSC	no	20 s	Linear	15-20 s
[14]	Bang-Bang, nl. s(t)	-	50 s	Pos.Pos.FB	100 s
[15]	Adaptive VSC	no	30 s	Input shaping	10-20 s
[16]	PD+PWPF	yes	20 s	Pos.Pos.FB	10 s
Sec. 4.2	VS P+relay	little	15-20 s	VS P+relay	5 s

Pos.Pos.FB: Positive position feedback, nl: nonlinear

Table 3. Comparison analyses.

Proposed variable structure P+relay attitude controller in Section 4.2 has a little chattering and settling time about 15-20 seconds for a simple station keeping attitude maneuver and VS P+relay vibration controller has also a little chattering (because of introducing dead-band) and settling time about 5 seconds. From second part of Table 3 proposed P+relay controller provides relatively good control performances. Obtained results are preferable for both designs with considered reaction wheels and thrust-jets than that of [6], [7], [8], [11], [14], [15] nevertheless the other results shown in Table 3 also are acceptable for their operational conditions.

5. Conclusions

In this chapter, we have first introduced rigid-body dynamics of orbiting spacecraft. Then we have developed equation of motion of a rigid-body in circular orbit with internal torquers. We have also considered solar arrays and write equation of motion for a flexible spacecraft model. At the second stage we have designed variable structure control system for a rigid body controlled with reaction wheels. Then we have proposed P+relay control technique to design attitude and vibration control systems for a satellite with solar arrays. Finally we had a comparison of simulation results of proposed control techniques with literature.

Modeling and simulation results show that proposed variable structure attitude control system for a rigid body stabilizes nonlinear dynamics successfully and performs satisfactory settling time and control torque with compared results. On the other hand, proposed P+relay attitude and vibration controller for geosynchronous satellite with flexible solar arrays successfully stabilizes the nonlinear model with external disturbances by using minimum fuel for considered initial conditions. We suggest using of P+relay control technique for station keeping maneuvering of flexible satellites.

6. Nomenclature

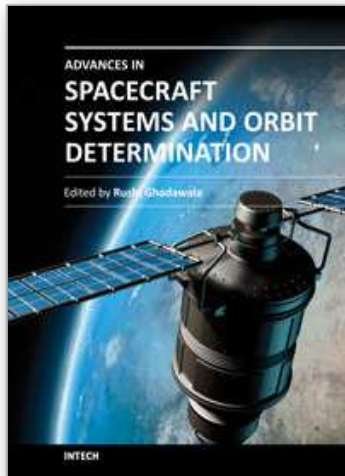
$\vec{\rho}$: position vector from center of mass of a small mass element
\vec{R}	: position vector from an inertial origin
dm	: a small mass element
n	: orbital rate
M_G	: gravity gradient torque
\vec{H}	: angular momentum vector
$\vec{\omega}$: angular velocity vector
$\vec{\theta}$: attitude angle vector
I	: inertia matrix
C	: direction cosine matrix
Γ	: skew symmetric matrix of angular velocity
Ω	: skew symmetric matrix of direction matrix
d_E	: external disturbance
q	: modal coordinates vector
a_1, a_2, a_3	: unit vectors of a local horizontal and a local vertical ref. frame
b_1, b_2, b_3	: unit vectors of body ref. frame
s	: sliding manifold
u_B	: control vector
V	: Lyapunov function candidate

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"Advances in Spacecraft Systems and Orbit Determinations", discusses the development of new technologies and the limitations of the present technology, used for interplanetary missions. Various experts have contributed to develop the bridge between present limitations and technology growth to overcome the limitations. Key features of this book inform us about the orbit determination techniques based on a smooth research based on astrophysics. The book also provides a detailed overview on Spacecraft Systems including reliability of low-cost AOCS, sliding mode controlling and a new view on attitude controller design based on sliding mode, with thrusters. It also provides a technological roadmap for HVAC optimization. The book also gives an excellent overview of resolving the difficulties for interplanetary missions with the comparison of present technologies and new advancements. Overall, this will be very much interesting book to explore the roadmap of technological growth in spacecraft systems.

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