CHAPTER 2

ORBITAL DYNAMICS

2.1 INTRODUCTION

This chapter presents definitions of coordinate systems that are used in the satellite, brief description about satellite equations of motion and relative motion dynamics. Controllability issues of a magnetic actuated satellite and various disturbance torques acting on the satellite are explained in the following sections along with the attitude control modes.

2.2 COORDINATE SYSTEMS

Before presenting any mathematical description, the Coordinate Systems (CS) used in satellite controls are defined (Wisniewski 1996) as below:

Control CS: This CS is a right orthogonal CS coincident with the moment of inertia directions and with the origin placed at the centre of mass as in Figure 2.1. The X-axis is the axis of the maximum moment of inertia and Z-axis is the minimum.

Body CS: This CS is a right orthogonal CS with its origin at the centre of gravity. The Z-axis is parallel to the boom direction and points towards the boom tip. The X-axis is perpendicular to the shortest edge of the bottom satellite body and points away from the boom canister. The Y-axis is perpendicular to the longest edge of the bottom satellite body. It is the reference CS for attitude measurements and the magneto-torquers.

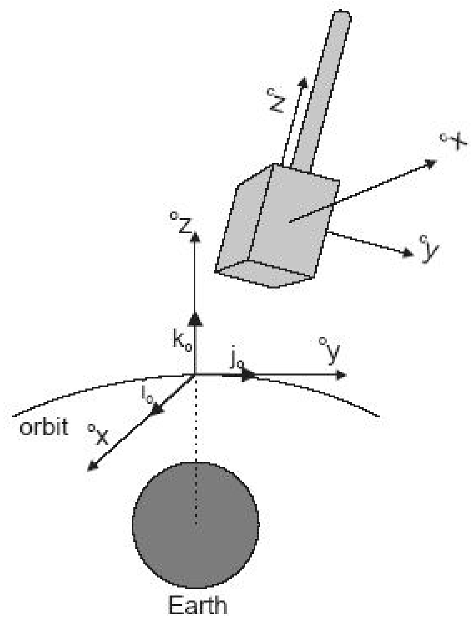

Figure 2.1 Definition of the control CS in the orbit CS

Orbital CS: This CS is a right orthogonal CS fixed at the centre of mass of the satellite. The Z-axis points at zenith (is aligned with the earth centre and points away from earth), the X-axis points in the orbit plane normal direction and its sense coincides with the sense of the orbital angular velocity vector. The orbit CS is the reference for the attitude control system.

Inertial CS: This CS is an inertial right orthogonal CS with its origin at the earth's centre of mass. The Z-axis is parallel to the earth rotation axis and points toward the North Pole. The X-axis is parallel to the line connecting the centre of the earth with vernal equinox and points towards vernal equinox (vernal equinox is the point where ecliptic crosses the earth equator going from south to north on the first day of spring).

In this work the satellite is considered to be homogeneous and axisymmetric so the body CS and the control CS are assumed to be the same.

The control CS is built on the principal axes of the satellite, whereas the orbit CS is fixed in orbit as shown in Figure 2.2.

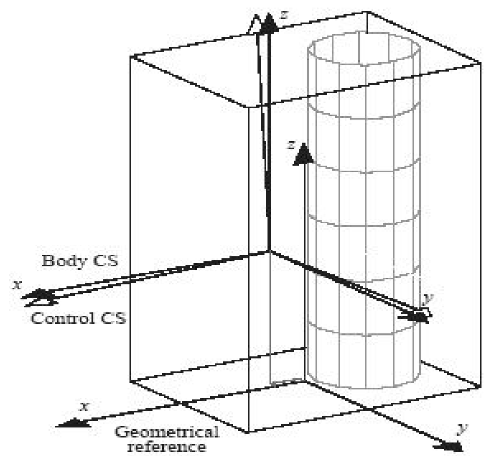

Figure 2.2 Definition of the body CS

The body CS in Figure 2.2 refers to the geometry of the satellite main body and its axes are perpendicular to the satellite facets.

2.3 EQUATIONS OF MOTION

The mathematical model of a satellite is described by dynamics and kinematics equations of motion (Wertz and Larson 1999). The dynamics relates torques acting on the satellite to the satellite's angular velocity in the world CS. The kinematics provides integration of the angular velocity.

2.3.1 Dynamics

The orbital motion of satellite is given by the position and linear velocity of the satellite in the orbit. The gravitational field governs equations of motion. A rigid body satellite is modeled for the low earth orbit. The dynamics equations and kinematics equations describe the satellite motion in the orbit. The dynamics equations relate the angular velocity of the satellite to the torques acting on the satellite. It is calculated in the body frame with respect to ECI frame, because Newton's laws are valid only in ECI frame.

A rigid body satellite has six degrees of freedom (3 translational, 3 angular).

$$F = m\frac{d\vec{v}}{dt} \quad \text{(Translational motion)} \tag{2.1}$$

$$T = \vec{M}\frac{d\vec{H}_b}{dt} \quad \text{(Angular motion)} \tag{2.2}$$

Torque, T is the sum of control torque, gravity gradient and disturbance torques. The satellite angular motion in terms of angular momentum is given by Euler moment equation (Piaski 1999).

$$T = \frac{d\vec{H}_b}{dt} = \frac{d\vec{H}_{bi}}{dt} + \vec{\omega}_{bi} \times \vec{H}_{bi} \tag{2.3}$$

where $$\vec{H} = I\vec{\omega} \tag{2.4}$$

$$I = \text{diag}[I_{xx} \ I_{yy} \ I_{zz}]^T \tag{2.5}$$

The derivative of angular momentum is,

$$\frac{d\vec{H}}{dt} = I\dot{\vec{\omega}} \tag{2.6}$$

$$T = I\dot{\vec{\omega}}_{bi} + \vec{\omega}_{bi} \times I\vec{\omega}_{bi} \tag{2.7}$$

$$I\dot{\vec{\omega}}_{bi} = -\vec{\omega}_{bi} \times I\vec{\omega}_{bi} + T \quad (2.8)$$

$$\dot{\vec{\omega}}_{bi} = -I^{-1}(\vec{\omega}_{bi} \times I\vec{\omega}_{bi}) + I^{-1}T \quad (2.9)$$

If $\vec{\omega}_{bi} = [\omega_x \ \omega_y \ \omega_z]$ then, (2.10)

$$\left.\begin{aligned} \dot{\omega}_x &= ((I_{yy} - I_{zz})\,\omega_y\omega_z + T_x)/I_{xx} \\ \dot{\omega}_y &= ((I_{zz} - I_{xx})\,\omega_z\omega_x + T_y)/I_{yy} \\ \dot{\omega}_z &= ((I_{xx} - I_{yy})\,\omega_x\omega_y + T_z)/I_{zz} \end{aligned}\right\} \quad (2.11)$$

Above equations are nonlinear-coupled differential equations. Control torque is generated by an interaction of the geomagnetic field with the magneto-torquer current $I_{coil}(t)$, which gives rise to a magnetic moment and is given by the following equation.

$$m(t) = n_{coil}\, I_{coil}(t) A_{coil} \quad (2.12)$$

The electromagnetic coils are placed perpendicular to the X, Y and Z axes of the body CS, thus the vector representing entire magnetic moment producible by all three magnetic coils are given in the body CS.

The control torque acting on the satellite is then given by,

$$T_{ctrl}(t) = m(t) \times B(t). \quad (2.13)$$

The magnetic moment m(t), will be considered as the control signal throughout the work.

The disturbance torque is due to aerodynamic drag, solar pressure, and several other effects.

2.3.2 Kinematics

The kinematics describes the orientation of the spacecraft body in space and is obtained through integration of the angular velocity (Wertz and Larson 1999).

The Euler angle parameterization of attitude will have singularity. This can be overcome by using kinematics equations in quaternion form.

The attitude quaternion scalar and vector can be derived from rotation matrix R as in equations (2.14) and (2.15) (Hall 2003).

$$q_4 = \frac{1}{2}\sqrt{(1 + \mathrm{trace}(R))} \tag{2.14}$$

$$q = \frac{1}{4q_4}\begin{bmatrix} R_{23} & R_{32} \\ R_{31} & R_{13} \\ R_{12} & R_{21} \end{bmatrix} \tag{2.15}$$

The orientation of the satellite in space is given by kinematics equations and is obtained through the integration of angular velocity. The kinematics equations are expressed by separate integrations of the vector and the scalar part of the attitude quaternion.

$$\dot{q} = \frac{1}{2}\ \omega_{bo} q_4 + \frac{1}{2}\omega_{bo} q \tag{2.16}$$

$$q_4 = -\frac{1}{2}\,\omega_{bo} \cdot q_4 \tag{2.17}$$

The relation between satellite angular velocity in ECI frame and angular velocity with respect to orbit frame is obtained by,

$$\omega_{bo} = \omega_{bi} - \omega_0 \hat{i}_o \tag{2.18}$$

2.4 ORBITAL PARAMETERS

For the earth orbiting spacecraft, it is common to define an inertial coordinate system with the center of mass of the earth as its origin and whose direction in space is fixed relative to the solar system. The X-axis is the axis of rotation of earth in positive direction, which intersects the celestial sphere at the celestial pole. The Y-Z plane of this co-ordinate system is taken as the equatorial plane of the earth, which is perpendicular to the earth axis of rotation.

The equatorial plane is inclined to the ecliptic plane, which is the plane of the earth orbit around the sun, by an angle 23.5^0. Either of two points on the celestial sphere at which the ecliptic intersects the celestial equator is called as equinox. The vernal equinox, also known as “the first point of Aries”, is the point at which the sun appears to cross the celestial equator from south to north.

The six major parameters, called classical orbit parameters (Wertz and Larson 1992) define the location in space of a body moving in a keplerian orbit.

The first two elements describe the shape of the orbit.

i) a, semi major axis

ii) e, eccentricity

The next two elements describe the position of the plane of the orbit.

iii) i, inclination

iv) α, right ascension of the ascending node

The next two elements describe the rotation of the elliptical orbit in its plane.

v) ω, argument of perigee and

vi) M, is mean anomaly

The various orbital elements are shown in Figure 2.3. The plane of the orbit is inclined to the equatorial plane of the earth by an angle i, the inclination of the orbit. The orbital plane and the equatorial plane intersect at the node line. The angle in the equatorial plane that separates the node line Y, Ω is called the right ascension, r is the radius vector of the moving body and r_p is the radius vector to the perigee of the orbit. The angle between r_p and node line is ω, the argument of perigee.

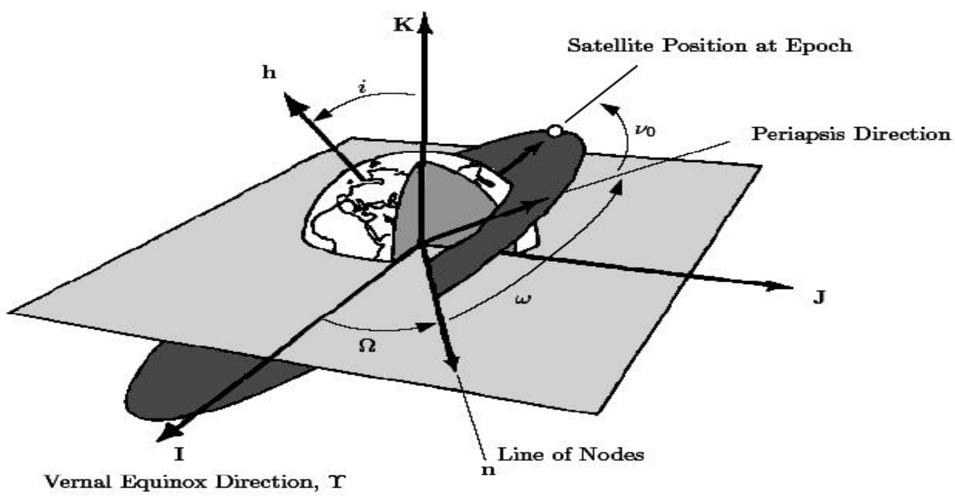

Figure 2.3 Orbital elements

2.5 AXES DEFINITION OF THE SAELLITE

The axis definition for the satellite is given in Figure 2.4.

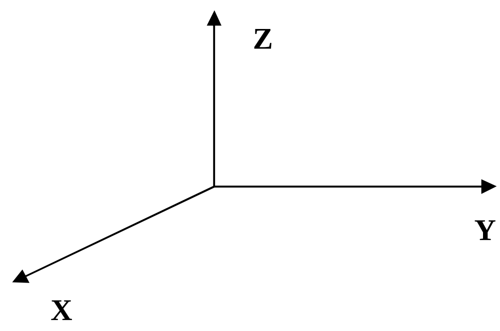

Figure 2.4 Axes definition

- X and Y are transverse axes
- Z is spin axis.

2.6 RELATIVE MOTION DYNAMICS

Relative motion between satellites is a subject of interest to rendezvous and formation flying. The relative motion of 3 satellites is shown in Figure 2.5.

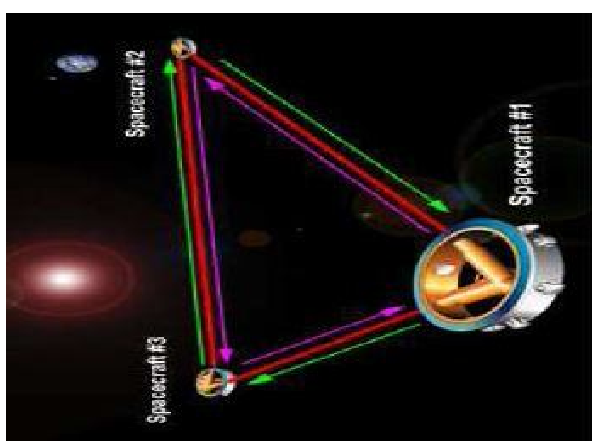

Figure 2.5 Relative motion of satellites in formation

Relative orbits of interest for formation flying are described in a rotating frame of reference, which is attached to the chief satellite (LVLH coordinates). Hence, it is desirable to write down the relative motion dynamics in this frame of reference. A Local-Horizontal Local-Vertical (LVLH) frame is denoted by the radial, circumferential and normal (r-θ-h) directions. Figure 2.6 shows a LVLH frame.

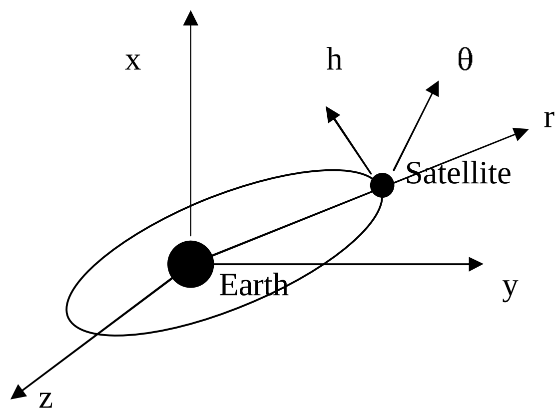

Figure 2.6 LVLH frame

The Euler-Hill's equations are the simplest set of equations in LVLH coordinates. The Euler-Hill's equations are obtained for a circular chief orbit by making the assumption of spherical earth, linearizing the differential gravity accelerations and neglecting all other perturbations to the two-body problem. These equations are derived by considering the chief satellite to be in circular orbit. The Euler-Hill's equations constitute a system of sixth order constant coefficient linear ordinary differential equations.

The Euler-Hill's equations are:

$$\left.\begin{aligned} &\ddot{x} - 2n\dot{y} - 3n^2x = 0 \\ &\ddot{y} + 2n\dot{x} = 0 \\ &\ddot{z} + n^2z = 0 \end{aligned}\right\} \tag{2.19}$$

where n is mean motion of the chief satellite and is given by $\sqrt{(\mu/a^3)}$ (Sidi 1997)

$$\mu = GM_{\oplus} \tag{2.20}$$

where G - Gravitational constant

$M_{\oplus}$ - Mass of the earth

a - Semi major axis of the chief's orbit

2.7 DISTURBANCE TORQUES

Disturbance torques is of two types, internal and external disturbance torques.

2.7.1 External Disturbance Torques

The different types of disturbance torques acting on the satellite are given in this section.

2.7.1.1 Aerodynamic drag

In a low earth orbit a satellite is subjected to an aerodynamic drag torque, due to atmospheric molecules. The collision of the satellite and the molecules produce a torque about the satellite center of mass. The aerodynamic torque acting on a satellite is composed by n surface elements (Wertz and Larson 1992). The drag acts in the opposite direction of the velocity vector and removes energy from the orbit. This energy reduction causes the orbit to get smaller, leading to further increase in drag. Eventually, the altitude of the orbit becomes so small that the satellite re-enters the atmosphere.

The equation for the acceleration due to drag on satellite is given below.

$$T_a = F\,(C_{pa} - C_g) \tag{2.21}$$

where $F = 0.5 \times \rho\, C_d A V^2$ (2.22)

T_a	-	Aerodynamic torque
F	-	Aerodynamic force
ρ	-	Atmospheric density
C_d	-	Drag co-efficient
A	-	Exposed surface area
V	-	Spacecraft velocity
C_{pa}	-	Center of aerodynamic pressure
C_g	-	Center of gravity
ρ	-	7.584×10^{-15} kg/m^3
C_d	-	Usually between 2 and 2.5
A	-	0.509 m^2
C_{pa}-C_g	-	0.1 (approximation that spacecraft is geometrically symmetric)

For this work, aerodynamic disturbance torque is calculated as $T_a = 3.238\times10^{-12}$Nm.

The orbit altitude, the satellite structure and the location of center of mass mainly influence the aerodynamic drag.

2.7.1.2 Solar radiation Pressure

Radiation incident on a spacecraft's surface produces a force, which results in a torque about the spacecraft's center of mass. The surface is subjected to radiation pressure or force/unit area equal to the vector difference between the incident and reflected momentum flux. The factors determining the radiation torque on a spacecraft are:

- Intensity and spectral distribution of the incident radiation
- Geometry of the surface and its optical properties
- Orientation of the sun vector relative to the spacecraft.

The mean momentum flux, P acting on the surface normal to the sun's radiation is given by,

$$P = F_e/c \qquad (2.23)$$

where F_e - Solar constant

c - Speed of light

In the model of the solar radiation torque it is assumed that the radiation is completely absorbed. The solar radiation pressure causes periodic variations in all orbital elements. Its effect is strongest for satellite with low ballistic coefficients. The solar radiation can be expressed as (Wertz and Larson 1992).

$$T_{sp} = F(C_{sp} - C_g) \qquad (2.24)$$

where $$F = Fs/c\ A_s\,(1+q)\,\mathrm{Cos}(i) \qquad (2.25)$$

T_{sp} - Solar radiation torque

F_s - Solar constant

c - Speed of light

A_s - Surface area

C_{sp} - Location of the center of solar pressure

C_g - Center of gravity location

q - Surface reflectance

i - Angle of incidence to the sun

A_s - $\sqrt{(2)} \times 0.6 \times 0.6 = 0.509 m^2$ (worst case)

q - Range 0 –1 and taken to be 0.6 (semi reflective)

Cos (i) - 1 (i=0, worst case)

C_{sp}-C_g = 0.075 m (approximation based on spacecraft is highly symmetric)

For this work, solar radiation torque is calculated to be $T_{sp} = 2.783 \times 10^{-7}$Nm.

The primary influence of solar radiation torque is on the satellite structure, the surface reflectivity and the satellite center of mass. The type, source and characteristic of the disturbances discussed above are listed in Table 2.1 (Wertz and Larson 1992).

Table 2.1 External disturbances

Type	Source	Characteristic
Aerodynamic	Earth atmosphere	Drag and lift on the satellite. Torque is cyclic for inertial pointing satellite.
Gravity gradient	Distribution of inertia	If the satellite has off-diagonal inertia elements it will produce a body fixed torque. The torque will be cyclic for inertial pointing satellite
Solar radiation	The Sun	The torques and forces cause by the sun are harmonics of the orbit rate, and dependent on the satellite geometry and the surface reflectivity. The torque is constant for a solar oriented satellite.

2.7.1.3 Earth's magnetic field

Magnetic disturbance torques result from the interaction between the spacecraft's residual magnetic field and the geomagnetic field. The primary sources of magnetic disturbance torques are:

i) Spacecraft magnetic moments

ii) Eddy currents

iii) Hysteresis

Spacecraft's magnetic moment is the dominant source of disturbance torques. The spacecraft is usually designed of the material selected to make disturbances from the other sources negligible (Droll and Iuler 1967). The instantaneous magnetic disturbances torque, N_{mag} (in Nm) due to spacecraft effective magnetic moment m (in AM^2) is given by,

$$N_{mag} = m \times B \qquad (2.26)$$

where B - Geocentric magnetic flux density

The torques caused by the induced eddy currents and the irreversible magnetizations of permeable material and hysteresis, are due to the spinning motion of the spacecraft (Visti 1957).

$$N_{eddy} = K_e (\omega \times B) \times B \qquad (2.27)$$

where ω - Spacecraft's angular velocity

K_e - Constant coefficient depends on spacecraft geometry and conductivity

In a permeable material rotating in a magnetic field, H energy is dissipated in the form of heat due to frictional motion of the magnetic domains. The energy loss over one rotation period is given by,

$$\Delta E_H \quad = \quad V\text{¢}\, H.dB \qquad (2.28)$$

where V - Volume of the permeable material

dB - Induced magnetic induction flux in the material

The hysteresis effects are appreciable only in very elongated "soft" magnetic material. The torque due to hysteresis is given by,

$$N_{hyst} \quad = \quad \frac{\omega \Delta E_H}{\omega^2 \Delta t} \qquad (2.29)$$

where Δt - Time over which the torque is being evaluated.

The magnitude of the above mentioned torques will be computed showing the effect of the various disturbance torques on the stability of the spacecraft justifying the spin rate of 8 rpm and the spin axis orientation within $\pm$ 3 degrees.

2.7.1.4 Gravity gradient torque

Any non-symmetrical object of finite dimensions in orbit is subjected to a gravitational torque because of the variations in the earth's gravitational force over the object. This gravity gradient torque results from the inverse square gravitational force field.

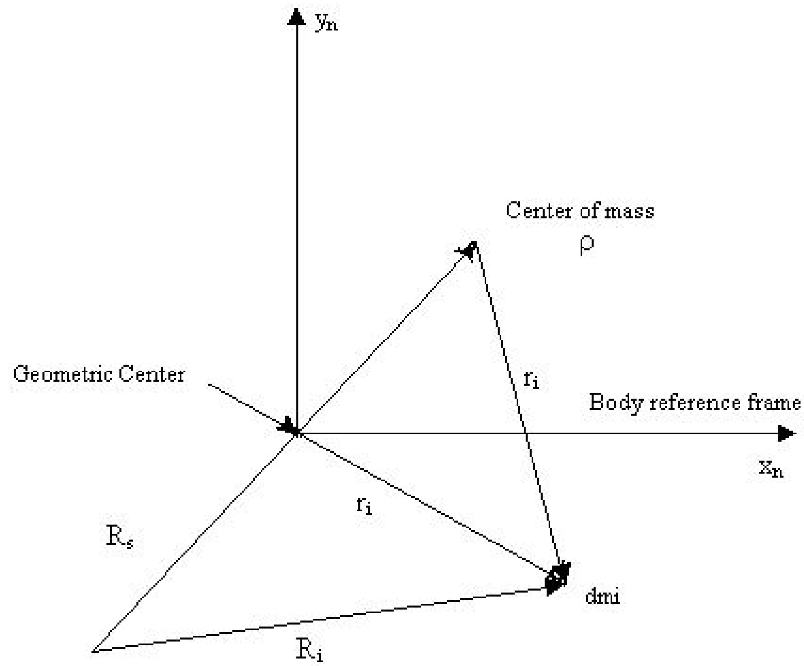

Figure 2.7 Gravity gradient torque

General expressions for the gravity-gradient torque on a satellite of arbitrary shape have been calculated for both spherical (Nidey 1960 and Hultquist 1961) and non-spherical (Roberson 1958) earth models. For most applications, it is sufficient to assume a spherical mass distribution for the earth or the effect of the earth's oblateness can be accounted for in the motion of the orbital plane (Hultquist 1961and Holland and Sperling 1969).

Referring to Figure 2.7 and assuming that the spacecraft's moment of inertia tensor is known for some arbitrary body reference frame and also by assuming the spacecraft to be orbiting a spherical earth, the gravitational force dF_i acting on the spacecraft mass element dm_i, located at a position R_i relative to the geocenter is given by,

$$dF_i = -\mu R_i \, dm_i / R_i^3 \tag{2.30}$$

where μ - Earth's gravitational constant

The torque about the spacecraft's geometric center due to force dF_i at a position r_i, relative to the spacecraft's geometric center is given by,

$$dN_i = r_i \times dF_i = (\rho + r'_i) \times dF_i \tag{2.31}$$

where ρ - Vector measured from the geometric center of mass

r'_i - Vector measured from the center of mass to the mass element dm_i

The gravity gradient torque on the entire spacecraft is obtained by integrating the equation (2.31).

$$N_{gg} = \int r_i \times dF_i = \int (\rho - \dot{r}_i) \times (-\mu R_i / R_i^3)\, dm_i \tag{2.32}$$

$$N_{gg} = (3\mu/R_s^3)\, [R_s \times (I.\, R_s)] \tag{2.33}$$

where I is the moment of inertia tensor.

The characteristics can be deduced as,

i) The torque is normal to the local vertical

ii) The torque is inversely proportional to the cube of the geocentric distance

2.7.2 Internal Disturbance Torques

Internal torques are defined as torques exerted on the main body of a spacecraft by internal moving parts like reaction wheels, flexible booms or solar arrays, scanning instruments, tape recorder reels, liquids inside partially filled tanks, or astronauts inside a manned space station. In the absence of external torques, the total angular momentum of the spacecraft remains constant. However internal torques can alter the system's kinetic energy and redistribute the spacecraft angular momentum among its component parts in ways, which can change its dynamic characteristics. For example, in a spinning spacecraft, angular momentum can be transferred from the nominal spin axis to another principal axis resulting in nutation, uncontrolled tumbling or flat spin. These undesirable results are best countered by attitude stabilization systems based on internally generated torques.

2.8 ATTITUDE CONTROL SYSTEM

In the context of spacecraft, attitude control is to maintain the angular position and rotation of the spacecraft, either relative to the object that it is orbiting, or relative to the celestial sphere. The function of the attitude control system is to control and maintain the angular orientation or direction of pointing of the spacecraft within acceptable limits and to overcome any disturbances acting on the satellite. Spacecraft attitude is defined by the angular relationship between two coordinate systems; one is based on the spacecraft geometry and the other is based on some external reference frame, such as that defined by the rotating earth (Sidi 1997). There are different methods used in attitude control, some are passive and determined by the physical properties of the satellite and others are active such as magnetic torquers, momentum wheels and thrusters. ACS is mainly classified into passive and active stabilization methods.

2.8.1 Passive Stabilization Methods

The passive methods chosen to take into account are gravity gradient, spin stabilization and passive magnets. A common feature of all passive stabilization is that the satellite will only be stabilized in two axes. Passive control requires no power and no actuators. It is simple and very inexpensive to implement and is appropriate for missions having loose pointing requirements. Passive control techniques take advantage of basic physical principles and/or naturally occurring forces by designing the spacecraft so as to enhance the effect of one force, while reducing the effect of others.

2.8.1.1 Gravity gradient stabilization

The gravity gradient uses the earth gravitational force to stabilize the satellite.

2.8.1.2 Spin stabilization

A spin-stabilized satellite is passively stable, when the satellite is spinning around the axis with the largest momentum of inertia. For a mid-range of attitude control accuracy, the satellite itself can spin on a particular axis to maintain its orientation. This creates an axis of momentum, which will maintain its orientation despite small perturbations. Precision control can then be enhanced with other actuators. This is only applicable to certain missions with a primary axis of orientation that does not need to change dramatically over lifetime of the satellite and in cases where there is no need for extremely high precision pointing. It is also useful for missions with instruments that must scan the star field or the earth's surface or atmosphere.

A body spinning about its major axis or minor axis will keep the direction of its spinning axis fixed with respect to the inertial space. Spin is used to stabilize the attitude of the satellite during its linear velocity augmentation while transferring between orbits. This direction will change only if external moments are applied to its center of mass and perpendicularly to the spin axis. In the proposed spin-stabilized micro-satellite, the spacecraft will be imparted a spin rate of 8±0.5 RPM, which is found to be a good compromise between the stability of the satellite, rate of heat dissipation and the maneuverability of the satellite.

2.8.1.3 Dual spin stabilization

Dual spin stabilization is a derivative of the previously mentioned passive spin stabilization. In dual spin only part of the satellite is spinning as opposed to passive spin stabilization, where the entire satellite is spinning. Dual spin can be accomplished by making part of the satellite spin and keeping the payload platform stationary or by placing a momentum wheel along the axis that is to be stabilized.

2.8.1.4 Passive magnets

Passive magnet will stabilize the satellite with the earth's magnetic field and the movement of the satellite will follow the magnetic field.

Depending on the position of the centre of gravity, the gravity gradient might not have a stabilizing effect. The spin stabilization is dependent on the distribution of the momentum of inertia. Ideally only the inertia tensor would have values different from zero and these values should further more be distinctly different from one another. This would make it easier to control the satellite, as a torque applied in one axis will have no effect on the others. Passive magnets will align the satellite to the earth's magnetic field. However this is not desirable for most control purposes, since the satellite will only point with sufficient accuracy over the poles. Further more it will make it much harder to use a magnetometer for attitude determination, as the magnetic field from the passive magnets will be much stronger than the earth's magnetic field, thereby possibly saturating the magnetic sensors.

2.8.2 Active Stabilization Methods

Active stabilization makes it possible to control the satellite in three axes. Each of the actuators only have an effect in one axis, therefore 3 independent actuators are needed for three-axis control. Three types of actuators are commonly used are magnetic torquers, momentum wheels and thrusters. A spacecraft with active attitude control has the ability to change its angular orientation using actuators such as thrusters or reaction wheels. This allows the spacecraft to turn, to point an antenna, prepare for a burn, or perform any other task that requires a specific orientation.

2.8.2.1 Three-axis stabilization

With three-axis stabilization, satellites have small spinning wheels, called reaction wheels or momentum wheels that rotate so as to keep the satellite in the desired orientation in relation to the earth and the sun. If satellite sensors detects the satellite to be moving away from the proper orientation, the spinning wheels speed up or slow down to return the satellite to its correct position. Some spacecraft may also use small propulsion system thrusters to continually nudge the spacecraft back and forth to keep it within a range of allowed positions. An advantage of 3-axis stabilization is that optical instruments and antennas can point at desired targets without having to perform “de-spin” maneuvers**.**

2.9 ACS PERFORMANCE COMPARISION

The performance of various attitude control techniques with typical accuracy, number of axes and their remarks are given in Table 2.2.

Table 2.2 Attitude control systems performance

Method	Accuracy (deg)	Axes	Notes
Spin stabilization	0.1-1.0	1, 2	Passive, simple, cheap, inertially oriented
Gravity gradient	1-5	2	Passive, simple, cheap, central body oriented
Reaction control system	0.01-1	3	Expensive, quick response, consumables
Magnetic torquers	1-2	2	Cheap, slow, lightweight, LEO only
Momentum wheel	0.1-1	2	Expensive, similar to dual spin
Reaction wheels	0.001-1	3	Expensive, precise, faster slew
Control moment gyro	0.001-1	3	Expensive, heavy, quick, for fast slew, 3-axes

2.10 ACS FUNCTIONS

The different modes of ACS functions discussed in the thesis are detumbling and initial spin up, spin down, spin rate control and spin axis orientation control.

2.10.1 Detumbling and Initial Spin Up - Spin Down Mode

Detumbling is done after the satellite is released from the launch vehicle. The satellite will be tumbling around having very high body rates and needs to be detumbled in order to bring down the body rates to low values so that the coning motion will be within the specified limits and thus to stabilize the satellite. The torquers mounted along two transverse axes are actuated to spin up or down the satellite along the spin axis (maximum moment of inertia axis). Transverse axes torquers are actuated till the satellite starts spinning with the desired rate of 8 rpm. Initially Spin up/down is done along with detumbling.

2.10.2 Spin Rate Control

During normal mission the spacecraft spin rate along the maximum moment of inertia axis is required to be maintained within the specified band of 8 ± 0.5 rpm. Transverse axes torquers are actuated till the specified rate is reached.

2.10.3 Spin Axis Orientation Control

The spin axis must be pointed accurately, so the control of spin axis orientation becomes important. The spin axis orientation is maintained within ± 3 deg from the negative orbit normal. The spin axis deviation is corrected using the magnetic torquer mounted along the spin axis. The dipole moment created by the torquer interacts with the local earth’s magnetic field and produce a torque in the transverse plane to reduce the deviation of spin axis from the negative orbit normal.

2.11 CONCLUSION

Brief insights of the orbital dynamics were given including the orbital elements, their perturbation forces and their calculation. The Hill’s equation for relative motion and the satellite equations of motion were also discussed. The stabilization techniques for satellites were also stated along with the various attitude control modes for micro satellites. These fundamentals form the basics for the forth coming proposed work.