2

Satellite Orbits and Inclination

2.1. Introduction

The orbit of satellite in use for communication purposes has special significance. It is geostationary and has to be maintained geostationary at all costs. Thus there are two main problems involved with the communication satellites regarding the orbit, namely the (*i*) launching and putting the satellite into geostationary orbit and (*ii*) maintaining it (station keeping). The importance of geostationary orbit has already been discussed in chapter 1. Once the satellite is placed in geostationary orbit, it has to be parked in the desired slot till its whole life. Various kinds of manoeuvre are needed to do that. There are standard techniques to achieve them and all these are specified and recommended by the International Telecommunication Union (ITU) and International Consulative Committee (CCIR) which also prescribe the parking slots for individual countries. This helps in avoiding unnecessary overlapping of the satellite coverage area and thus avoiding the unnecessary losses of information or traffic capacity. The communication satellite is a very large investment and its life is limited. It is therefore very essential that its full and proper utilization be made. In this chapter therefore the orbital aspects of the communication satellite are discussed.

Communication satellites move around the earth as planets do around the sun and therefore the three Keplers' laws apply to them also. These Kepler's laws are : (*i*) the orbit of satellites is an ellipse with the center of the earth at one focus, (*ii*) the line joining the center of earth and the satellite sweeps over equal areas in equal time intervals and (*iii*) the squares of the orbital periods of two satellite have the same ratio as the cubes of their mean distances from the center of the earth. The satellite orbit may be either elliptical or circular and its characteristics are governed by these laws. Infact these are the laws that initially developed the techniques for launching and putting the communication satellite in geostationary or any

 $D = V \times t$

orbit (*e.g.* low altitude satellites for special research purposes). The tracking of the satellite, its station keeping etc all depend on these Kepler's laws. The Newton's gravitational law of force also controls the satellite's orbits and infact a very simple calculation regarding the geostationary satellite distance etc may be carried out from Newton's law of force.

The orbital aspects which are of importance are the determination of the orbit of the satellite, the distance between the satellite and earth stations, coverage angle, earth station pointing angles, eclipses and solar interference. The orbits may be far, near, equatorial, polar or inclined. The earth station antenna that communicates with the geostationary satellite requires a knowledge of azimuth angle and the elevation angle with respect to latitude and longitude of the location of the earth station. A knowledge of the polarization angle is also necessary. The study of coverage areas requires the knowledge of the coverage angle and slant range. The duration of eclipse is essential to be calculated because during these periods the satellite has to depend on its own stored power as solar batteries remain idle then. The eclipse may be caused by the earth and the moon. The solar interference appears in a variety of ways. It increases the antenna noise temperature when the sun is within the earth station beam. There is also a continuous drift in the satellite orbit both in longitude and latitude due to various disturbances. This drift is to be constantly nullified so that the orbit be stationary but perfect stationarity is not possible and so a satellite is constrained to remain within a 'window' whose limits are defined by an angular shift as seen from the center of the earth around the required nominal position. Usually this window is about 75 km on the sphere containing the geostationary satellite orbit.

2.2. Synchronous Orbit

A *geostationary satellite* is synchronous and has an equatorial, circular and direct orbit. It should be noted that synchronous is often confused with stationary. Polar and inclined orbits may be synchronous but never stationary. At the geosynchronous altitude and orbital velocity of the satellite equals the velocity of a point on the earth's equator. The geosynchronous altitude can be easily calculated. The movement of the satellite with respect to the earth can be considered as that shown in Fig. 2.1. Let M and m be the mass of the earth and the satellite respectively. Similarly let R_E be the radius of synchronous circular orbit of the satellite. If ω is the angular velocity of the circular motion of the satellite in radians per second then the centripetal force acting on the satellite would be

$\boxed{mR\omega^2}$ This centripetal force equal to the gravitational force (Newtons gravitational law) between the satellite and the earth expressed by $\boxed{GmM/R^2}$ where G is the gravitational constant. Thus

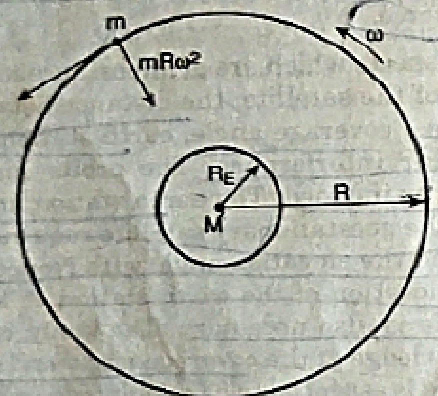

Fig. 2.1. Geostationary Orbit.

$$mR\omega^2 = \frac{GmM}{R^2} \quad \text{...(2.1)}$$

At the earth's surface the gravitational force in mg where g is the acceleration due to gravity. Then

$$\frac{GmM}{R_E^2} = mg \quad \text{...(2.2)}$$

From Eqs. (2.1) and (2.2) one can have

$$R = \left(\frac{gR_E^2}{\omega^2}\right)^{1/3} = \left(\frac{gR_E^2}{4\pi^2 f^2}\right)^{1/3} \quad \text{...(2.3)}$$

where $f = \omega/2\pi$ is the rotation rate in revolutions per second. Putting $f = 1$ revolution per day and $R_E = 6370$ km, $g = 9.9$ m/s, one obtains $R = 42{,}208$ km. Subtracting R_E form R one gets 35,838 km as the orbital height above the equator which is quite close to the precise value of 35,860 km. The maximum illumination of earth by a geosynchronous satellite is as that shown in Fig. 2.2. It is evident that polar regions are not illuminated by the geosynchronous satellite. The angle subtended by the earth is 17.3° and the highest latitude at which satellite is visible is 81.3°. Further for all practical purposes an earth station requires a minimum elevation angle of atleast 5° above the horizon. This reduces the highest latitude to about 76° at the longitude of the satellite. Corresponding to this 76° latitude of illumination the maximum distance illuminated by the geostationary satellite is about 16,900 km which is around 42%

of the equatorial distance. To cover the polar regions therefore a satellite with an elliptical and inclined orbit is needed. A typical example is Molniya series of satellite of USSR. Molniya orbit is elliptical, inclined 65° with respect to the earth with a 40,000 km apogee in the northern hemisphere and a 500 km perigee in the southern hemisphere. 3 Moliniya satellites are required to provide 24 hour coverage of the polar regions.

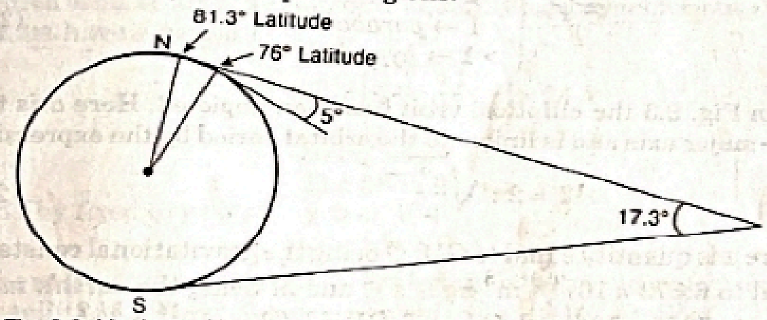

Fig. 2.2. Maximum Illumination of the Earth by a Geosynchronous Satellite.

2.3. Orbital Parameters

The satellite's movement in an orbit follows the three Kepler's laws. The first Kepler's law gives that the satellite moves along a conic (orbit) on the orbit plane, the equation for which is

$$r = p/(1 + e\cos\upsilon) \quad \text{...(2.4)}$$

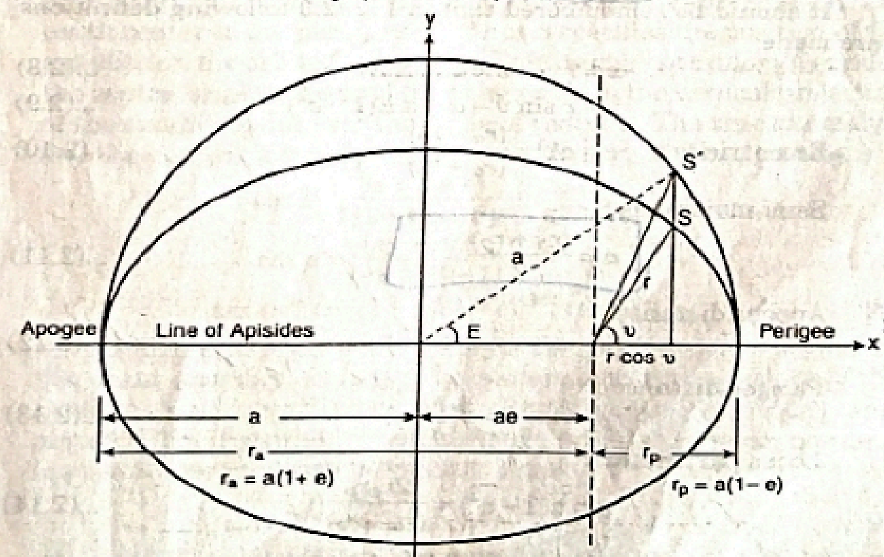

Fig. 2.3. Geometry of an Elliptical Orbit of Communication Satellite.

where p is a parameter, e is the eccentricity and υ is the central angle (also called the true anomaly). Fig. 2.3 represents the above orbit of satellite. In fact the value of the eccentricity e determines the type of the conic (orbit).

$$e = \begin{Bmatrix} 0 \rightarrow \textit{circle} \\ < 1 \rightarrow \textit{ellipse} \\ 1 \rightarrow \textit{parabola} \\ > 1 \rightarrow \textit{hyperbola} \end{Bmatrix} \qquad \text{...(2.5)}$$

In Fig. 9.3 the elliptical orbit has been depicted. Here a is the semi-major axis and is linked to the orbital period by the expression

$$T = 2\pi \sqrt{\frac{a^3}{\mu}} \qquad \text{...(2.6)}$$

where μ is quantity equal to GM, G being the gravitational constant, equal to 6.672×10^{-11} m^3 kg^{-1}s^{-2} and M being the earth's mass equal to 5974×10^{24} kg. Thus $\mu = GM = 3.986 \times 10^{14}$ m^3/s^2.

Satellite velocity for a circular orbit is $\sqrt{\mu/a}$ where a is the orbit radius. In fact the satellite velocity at an orbital point s (distance to the centre of the earth r) is given by

$$\boxed{V_s^2 = \frac{2\mu}{r} - \frac{\mu}{a}} \qquad \text{...(2.7)}$$

It should be remembered that in Fig. 2.3 following definitions are made

$$x = r \cos \upsilon = a\,(\cos E - e) \qquad \text{...(2.8)}$$

$$y = r \sin \upsilon = a \sin E(1 - e^2)^{1/2} \qquad \text{...(2.9)}$$

Eccentricity : $\quad e = c/a = \dfrac{(r_a - r_p)}{(r_a + r_p)}$ ...(2.10)

Semi major axis :

$$\boxed{a = \frac{(r_a + r_p)}{2}} \qquad \text{...(2.11)}$$

Apogee distance :

$$r_a = a + c = a(1 + e) \qquad \text{...(2.12)}$$

Perigee distance :

$$r_p = a - c = a(1 - e) \qquad \text{...(2.13)}$$

Locus parameter :

$$p = a(1 - e^2) = \frac{2r_a r_p}{(r_a + r_p)} \qquad \text{...(2.14)}$$

Semi minor axis $\quad b = a(1 - e^2)^{1/2}$

$$= (r_a r_p)^{1/2} \qquad \text{...(2.15)}$$

A synchronous satellite has T as the *sideral period* of rotation of its primary body, T_p (*i.e.* the period in any fixed reference coordinate system). It should be noted that for the earth T_p is not 24 hours (the so called *synodal period*) because in one day the earth both rotates once around its polar axis and also completes 1/36524 of the annual earth orbit around the sun. Consequently a geosynchronous satellite must have a period

$$T = T_p = \left(1 - \frac{1}{36524}\right) \times 24\ h$$

$$= 86163{,}44\ s$$

$$= 23\ h\ 56\ m\ 4\ s \qquad \text{...(2.16)}$$

in any fixed coordinate system. If

$$T = nT_p,\ n = 2, 3, 4, \qquad \text{...(2.17)}$$

the satellite is called *super-synchronous* whereas a *sub-synchronous* satellite has T and T_p interchanged in Eq. (2.17). In Eqs. (2.12) and (2.13), *perigee is* defined as a point in the orbit where the satellite is closest to the earth and similarly the point where the satellite is farthest from the earth is called the *apogee.*

As already mentioned before υ is termed the *true anomaly* and is the positive angle oriented according to the velocity vector of the spacecraft, between the earth center to the satellite axis and the earth center to the perigee axis. Thus υ specifies the position of the satellite on the orbit. Angle E is called the *eccentric anomaly* and is the central angle measured from the x axis to the vertical projection of the satellite point over the circle of radius a. The true anomaly υ and the ecentric anomaly E are related by any of the expressions

$$\cos E = \frac{(e + \cos \upsilon)}{(1 + e \cos \upsilon)} \qquad \text{...(2.18)}$$

$$\cos \upsilon = \frac{(e - \cos E)}{(e \cos E - 1)} \qquad \text{...(2.19)}$$

$$\tan(\upsilon/2) = [(1 + e)/(1 - e)]^{1/2} \tan(E/2) \qquad \text{...(2.20)}$$

Another kind of anomaly usually defined is M which is also called the *mean anomaly*. It may be considered as a true anomaly the satellite would have if it proceeded along a circular orbit of the same period T. This hypothetical circle may be considered as circumscribing the ellipse as shown in Fig. 2.3. Thus

$$m = \frac{2\pi t}{T} \qquad \text{...(2.21)}$$

where $2\pi/T$ is the satellite mean motion. Angles M and E are related by the equation

$$M = E - e \sin E \qquad \text{...(2.22)}$$

2.4. Satellite Location with Respect to the Earth

For an earth satellite, the reference frame is taken as rectangular coordinate system (OXYZ) with origin O in the earth's center of gravity. The z axis coincides with the polar axis and is oriented towards the north. The X-Y plane is thus the equatorial plane of the earth. Its angle i with the satellite orbit plane (or the conic section) is called the *inclination* and is defined in the interval ($0° < i < 180°$). A satellite orbit is *equatorial* if the orbit plane coincides with the reference plane of the primary body ($i = 0°$). An orbit is *polar* if the orbit plane contains the polar axis of the primary body ($i = 90°$). An orbit is inclined if it is neither equatorial nor polar. Fig. 2.4 indicates these equatorial and inclined orbits of the satellite. Inclined or equatorial orbits in which the satellite's projection on the equatorial plane of the primary body revolves in the same direction as the primary body itself are known as *direct orbits*. They have inclinations less than 90°. Earth rotation is seen to assist the launching of satellites into direct orbits. Orbits for which $i > 90°$ are called *retrograde orbits*.

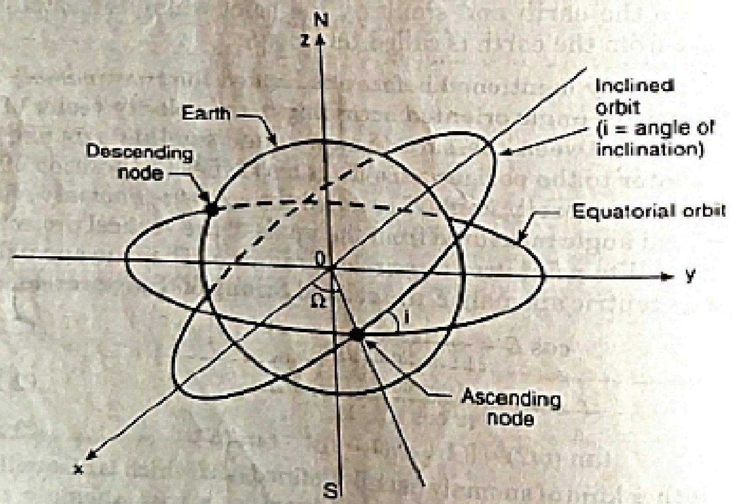

Fig. 2.4. Satellite Orbits.

There are two points at which the orbit of the satellite penetrates the equatorial plane. These points are called nodes. There are two such nodes, namely the *ascending node* and the *descending node* as shown in Fig. 2.4. The satellite moves upward through the equatorial plane at the ascending node and downward through the equatorial plane at the descending node. It should be noted that in the reference

frame OXYZ mentioned above the x axis (Fig. 2.4) points towards a fixed location in the free space called the *first point of Aries*. This is the direction of a line from the center of the earth through the center of the sun at the vernal equinox (about March 21), the instant when the subsolar point crosses the equator north to south. Angular distance measured eastward in the equatorial plane from the x axis is called the *right ascension* and is represented by the symbol RA. The right ascension of the ascending node is called Ω. Thus Ω is the longitude of the ascending node. Infact the variables Ω and i together locate the orbital plane with respect to the equatorial plane.

Another parameter of interest for the satellite orbit location with respect to earth is ω the *argument of perigee*. This is the angle measured along the orbit from the ascending node to the perigee. Thus the five parameters (i, a, e, Ω, ω) completely define the satellite orbit in the space. Inclusion of sixth parameter which may be either the true anomaly v or the mean anomaly or the eccentric anomaly will define the motion of the satellite on its orbit.

It would be of importance to metion here that the nonsphereeity of the earth, the nonuniform distribution of masses of its surface and the effects of lunar and solar gravitation all make the actual motion of satellites depart from that described above. These variation in the satellite orbit are called *secular variations* and are of two type known as the *nodal regression* and the *rotation of the lines of the apsides*. The nodal regression is a rotation of the orbit plane in the direction opposite to the satellite motion around the axis of rotation of the earth. The rate of this motion in degrees per day is given as

$$\Omega^0 = -[10/(1-e^2)^2]\,(R/a)^{7/2}\cos i \quad \text{...(2.24)}$$

The rotation of the line of the apsides is the rotation of the ellipse major axis around the center of the earth on a fixed orbit plane. Apogee and perigee move then with respect to the earth, and the rate of motion of the argument of perigee is also a function of orbit inclination angle i, or

$$\omega^0 = [5/(1-e^2)^2]\,(R/a)^{7/2}\,(5\cos^2_i - 1)°/\text{day} \quad \text{...(2.25)}$$

The critical value $i = \cos^{-1}(1/\sqrt{5}) = 63°24'$ stops this motion. Smaller values of i make the ellipse rotate in the same sense as the satellite motion along it. Higher values of i make the ellipse rotate in the opposite sense. The effect becomes smaller for higher orbits. For lower altitude orbits the earth's gravitational field affects the satellite orbit but for higher orbits the solar and lunar gravitational fields are of much concern. For geostationary orbit these effects cause a change in satellite inclination by 1° per year. It may be noted that the inclination causes the satellite to move in the form of figure of eight (Fig. 2.5) which gives the corresponding amount of drift in distance (km) both in longitude and latitude of satellite. An inclination of 1° causes drift of satellite around 735.9 km in latitude side and 3.23 km in longitude side.

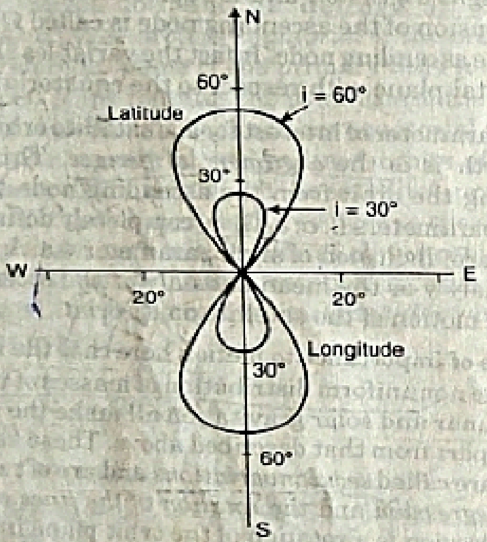

Fig. 2.5. Satellite's Apparent Movement in an Inclined and Synchronous Orbit with respect to the Ascending Node.

2.5. Look Angles

The look angles are the angles (coordinates) to which an earth station antenna must be pointed to communicate with the geosynchronous satellite. These angles are the azimuth (A) and elevation angle (E). These are calculated on the basis of a knowledge of latitude (ϕ) and relative longitude (θ) both in degrees of the earth station. Infact θ is the absolute value of the difference between the geostationary satellite longitude and that of the earth station. Azimuth angle is usually defined as the angle by which the antenna, pointing at the horizon must be rotated clockwise around its vertical axis, from the geographical north, to bring the antenna boresight into the vertical plane containing the satellite direction. Value of the azimuth angle is between 0° and 360°. Its value is calculated from Fig. 2.6 by determining the value of angle γ from the chart and then deriving A from angle γ using the insert table. Angle γ may also be calculated from the geometry of satellite orbit in the reference

system shown in Fig. 2.7. Angle γ can be calculated from the formula

$$\gamma = \text{arc}\, t_g \,(t_g\, \theta / \sin \phi) \qquad ...(2.26)$$

True Azimuth	Earth station Quadrant
A = 180−γ	NW NE
A = 180+γ	NW
A = γ	SW
A = 360−γ	SE

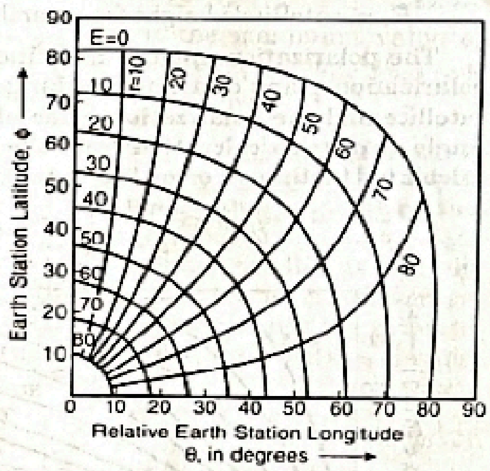

Fig. 2.6. Azimuth Angle γ and Elevation Angle E for Earth's Station Antennas.

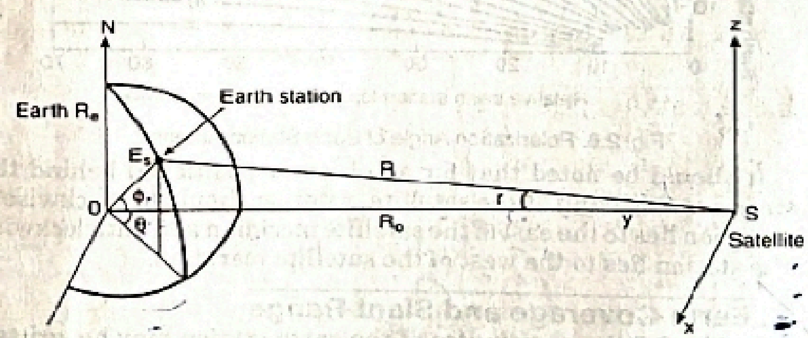

Fig. 2.7. Calculation of Angle γ from the Geometry of Satellite Orbit in the Reference System.

The *elevation angle* is defined as the angle by which the antenna boresight should be rotated in the vertical plane that contains the satellite direction from the horizontal to the satellite direction. Elevation angle E is also calculated from the chart given in Fig. 2.6. It is also calculated from the formula given below :

$$E = \frac{\text{Arc } t_g\left(\cos\theta\cos\phi - \dfrac{R_e}{R_e + R_o}\right)}{[1 - (\cos\theta\cos\phi)^2)]^{1/2}} \quad \text{...(2.27)}$$

where R_e is the earth radius $= 6378$ km and

$R_o =$ satellite's height from earth's centre = 35786 km.

The polarization angle ψ is defined as the angle between the polarization plane of a linear polarized wave transmitted by the satellite and the polarization plane of the earth station antenna. Angle ψ may be calculated from the chart of Fig. 2.8. It is also calculated (within an error less than 0.1° by the formula

$$t_g\,\psi = \frac{\sin\theta}{t_g\,\phi} \quad \text{...(2.28)}$$

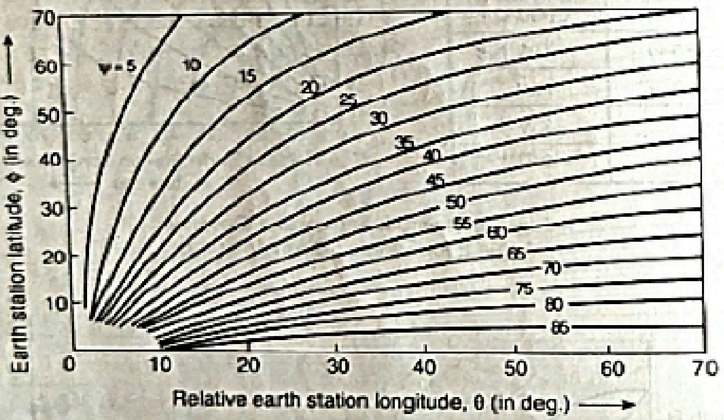

Fig. 2.8. Polarization Angle of Earth Station Antenna.

It should be noted that for an observer positioned behind the antenna and looking at the satellite, rotation should be clockwise if the station lies to the east of the satellite meridian and anticlockwise if the station lies to the west of the satellite meridian.

2.6. Earth Coverage and Slant Range

In Fig. 2.7 the coordinates of the earth station may be written as

$$x = R_e \cos\phi \sin\theta$$

$$y = R_0 + R_e (1 - \cos\phi \cos\theta)$$

$$z = R_e \sin\phi$$

The range R (also called *slant range*) from satellite to earth station then would be given as

$$R^2 = R_0^2 + 2R_e (R_0 + R_e)(1 - \cos\phi\cos\theta) \quad \text{...(2.30)}$$

For $R_e/R_o \approx 0.178$,

$$(R/R_0)^2 \approx 1 + 0.42\,(1 - \cos\phi\cos\theta) \quad \text{...(2.31)}$$

It can be seen from Eq. (2.31) that the maximum value of $(R/R_0)^2$ is 1.356 and if R^2 is approximated by R_0^2 it would lead to a maximum error of 1.3 dB in the power link budget.

Regarding the coverage of the earth by a single beam communication satellite there are two things to be considered. Firstly it is desired that it should atleast cover the country and then the coverage should be as maximum as possible. The maximum coverage (illumination) of the earth by a geostationary satellite is already shown in Fig. 2.2. In other words the maximum geometrical coverage is given by the portion of the earth within a cone with the satellite at its apex and tangent of the earth's surface. The apex angle (coverage angle) 2 α_{MAX} of this cone is

$$\boxed{2 \text{ Arc sin } \frac{R_e}{R_o + R_e} = 17.4^\circ} \quad \text{...(2.32)}$$

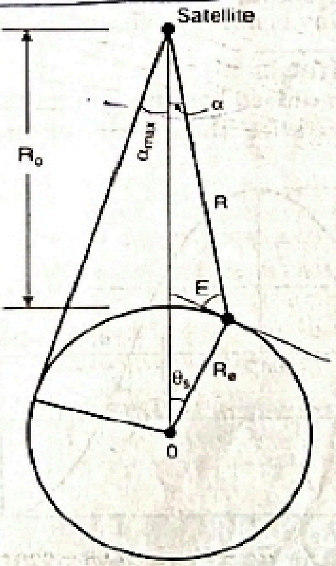

Fig. 2.9. Elevation Angle and Coverage Angle with Reference to Synchronous Satellite.

This angle is also shown in Fig. 2.2 and is redrawn in Fig. 2.9. In Fig. 2.9, θ_s is the central angle that represents the angular radius of the satellite footprint. It is given by

$$\theta_s = 180° - (90° + E + \alpha) = 90° - E - \alpha \qquad ...(2.33)$$

For a geostationary orbit, corresponding to coverage angle α_{max}, the central angle θ_s, is obtained from Eq. (2.33) by substituting $E = 0$ and $\alpha = \alpha_{max}$, this gives $\theta_s = 81.3°$. But as mentioned in Section 2.2 a minimum elevation angle of 5° is required then $\theta_s = 76.3°$. Thus not more than 76.3° of northern or southern latitudes will be covered by the geosynchronous satellite.

The slant range R may also be calculated in terms of elevation angle E as shown in Fig. 2.9. Its value would be

$$R^2 = (R_e + R_o)^2 + R_e^2 - 2R_e\,(R_e - R_o) \times \sin\left[E + \sin^{-1}\left(\frac{R_e}{R_e + R_o}\cos E\right)\right] \qquad ...(2.34)$$

which for $E_{min} = 5°$ would give the maximum slant range R = 41,127 km which gives a satellite round trip delay of $2d/c = 0.274$ s where c is the velocity of light. To this delay, delays caused by terrestrial extensions (which are of the order of 10-50 ms) are also added. However, because of the use of echo suppressor or echo cancellers the delay is not troublesome in telephone conversations.

2.7. Eclipse Effects

Solar eclipses caused by the earth and moon to the satellite are important as these affect the working of satellites and in particular

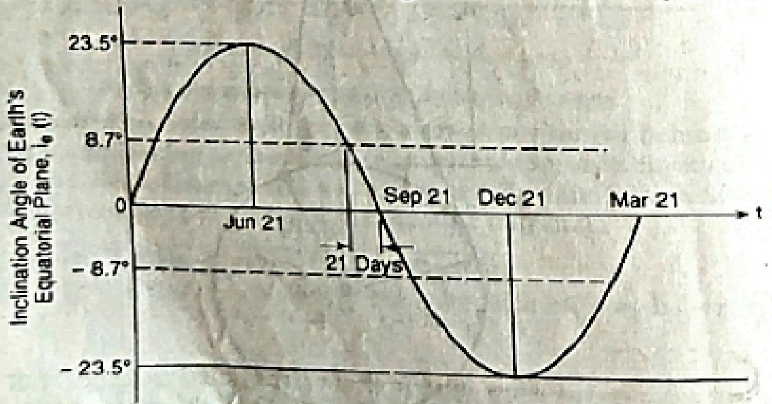

Fig. 2.10. (*a*) Variation of the Earth's Equatorial Plane Inclination Angle i_e (*t*).

the energy generated by solar cells is greatly affected then. Therefore the periodicity and duration of these solar eclipses are important. It should be noted that solar eclipse due to earth is of more importance than due to moon on the communication satellite as the former lasts for several days. In this section firstly the solar eclipse that due to earth would be discussed and then that due to moon would be discussed.

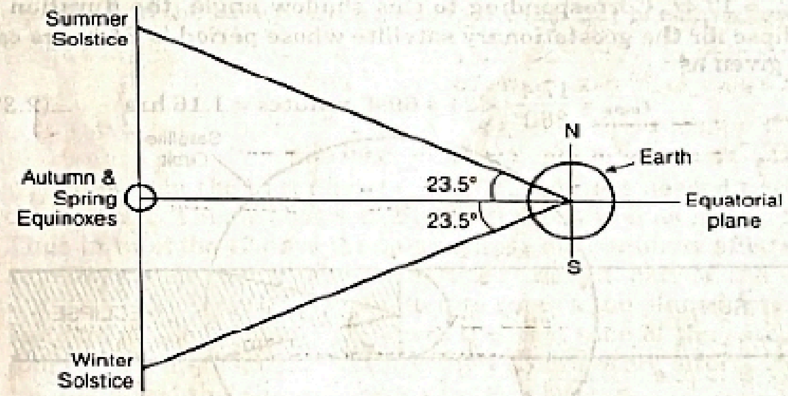

Fig. 2.10. (*b*) Apparent Movement of the Sun Relative to the Earth.

The inclination angle of the earth's equatorial plane $i_e\,(t)$ with respect to the sun's direction is as that shown in Fig. 2.10. Here angle $i_e\,(t)$ is in degrees. The annual variation $i_e\,(t)$ is sinusoidal in nature. Fig. 2.10 (*b*) shows the apparent movement of the sun relative to the earth. It is apparent that summer solstice occurs on 21st June and winter solstice is on 21st Dec. Similarly Spring and Autumn equinoxes occur on 21st March and 21st September respectively. The variation of $i_e\,(t)$ as that shown in Fig. 2.10 (*a*) can be written mathematically as

$$i_e\,(t) = 23.5 \sin \frac{2\pi t}{T} \qquad ...(2.35)$$

where T is the annual period equal to 365 days and the maximum inclination angle $i_e\,(t)$ is 23.5°. At the spring and autumn equinoxes the inclination angle is zero whereas it is at its maximum on summer and winter solstices.

The satellite is always illuminated at the solstice. It goes the maximum duration of eclipse during equinoxes. The solar eclipse during equinox is as that shown in Fig. 2.11. Here the earth's shadow is considered to be a cylinder of constant diameter. The maximum shadow angle occurs at equinoxes and is given as

$$\phi_{max} = 180° - 2\cos^{-1}\frac{R_e}{R_o + R_e} \quad ...(2.36)$$

which is same as that of coverage angle given in Eq. (2.32). Substituting value of $R_e = 6378$ km and $R_0 = 35786$ km (satellite altitude), $\phi_{max} = 17.4°$. Corresponding to this shadow angle, the duration of eclipse for the geostationary satellite whose period is 24 hours can be given as

$$t_{max} = \frac{17.4°}{360°} \times 24 = 69.4 \text{ minutes} = 1.16 \text{ hrs} \quad ...(2.37)$$

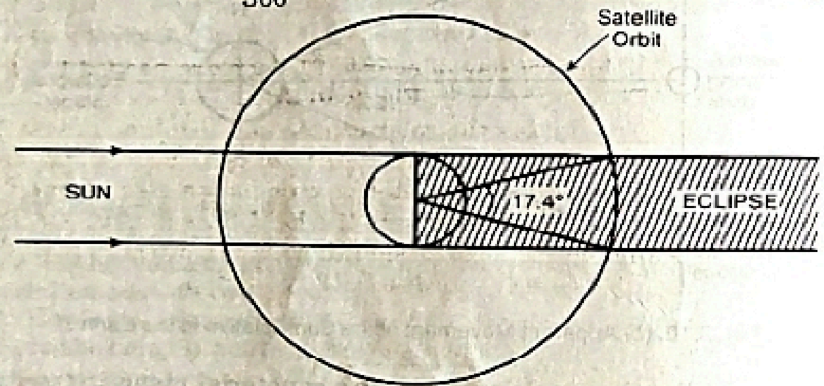

Fig. 2.11. Eclipse Due to Earth when Sun is at Equinox.

The first eclipse day before an equinox and the last day of eclipse after an equinox correspond to the relative position of the sun such that the sun rays tangent to the earth pass through the satellite orbit. Such a situation is shown in Fig. 2.12. In these cases the

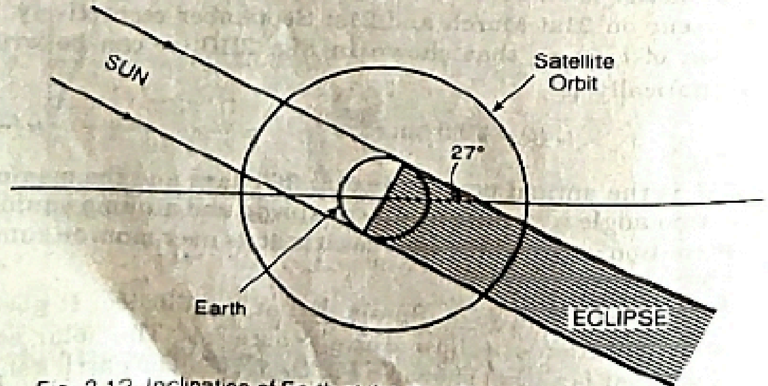

Fig. 2.12. Inclination of Earth at the First Day of Eclipse Before Equinox for a Geostationary Satellite.

inclination angle of the equatorial plane with respect to the direction of the sun is

$$i_e = \frac{1}{2}\phi_{max} = 8.7° \quad ...(2.38)$$

Corresponding to this inclination angle of the earth's equatorial plane, from Eq. (2.35), the time t from the first day of eclipse to the equinox and the time from equinox to the last day of eclipse may be derived as

$$t = \frac{365}{\pi}\sin^{-1}\left(\frac{8.7}{23.5}\right)$$
$$\approx 21 \text{ days} \quad ...(2.39)$$

Thus from the first day of eclipse 21 days are needed to reach the equinox. This is shown in Fig. 2.10 (*b*) as well as in Fig. 2.13. Thus in total for 42 days the solar eclipse near equinox affects the satellite's working. It should be noted that at half of the daily duration period of eclipse the satellite crosses the plane formed by the sun and the earth's axis where the local time at the satellite's longitude is mid night. Thus the satellite services are interrupted at midnight and the satellite needs a battery backup.

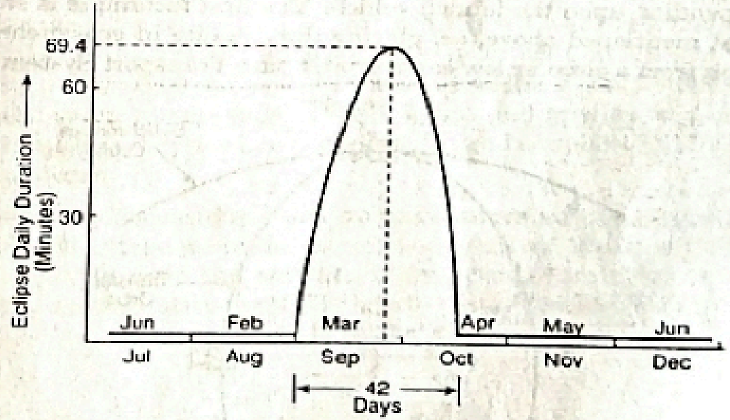

Fig. 2.13. Daily Duration of Solar Eclipse Caused by Earth.

The solar eclipse caused by moon to the geostationary satellite occurs when the moon passes in front of the sun. The eclipse occurs irregularly in time of duration and depth. Oftenly eclipses may occur twice within a 24 hour period. Eclipse duration may range from a few minutes to over two hours with an average duration of about 40 minutes. Compared to earth-solar eclipse, the number of moon solar eclipse range from zero to four with an average of two per year. It is worthwhile to note that if the moon solar eclipse of long duration occurs just before or just after the earth-solar eclipse, the satellite has to face special problems in connection with battery recharging and spacecraft thermal reliability. In order to cope up with the solar battery problems during eclipses an energy reserve is provided with the satellite.

2.8. Satellite Placement in Geostationary Orbit

The placement of satellite in the geostationary orbit is carried out on the principle of *Hohmann transfer* and is as that shown in Fig. 2.14. Here the satellite firstly has the lower circular earth orbit at an altitude of around 300 km or so. Then a velocity increment changes the satellite's lower circular earth orbit into an elliptical transfer orbit with the perigee at initial altitude (300 km) and with apogee at the altitude of final circular orbit (radius of the geosynchronous orbit is equal to 42,164.2 km). A second velocity increment then finally places the satellite into this desired orbit. The velocity increments are carried out by various auxiliary propulsion stages. There are three ways to carry out above procedure, of course depending upon the launch vehicle. The first technique is same as that mentioned above, *i.e.* placing the satellite in geosynchronous orbit from a circular low earth orbit. Space Transport System (STS)

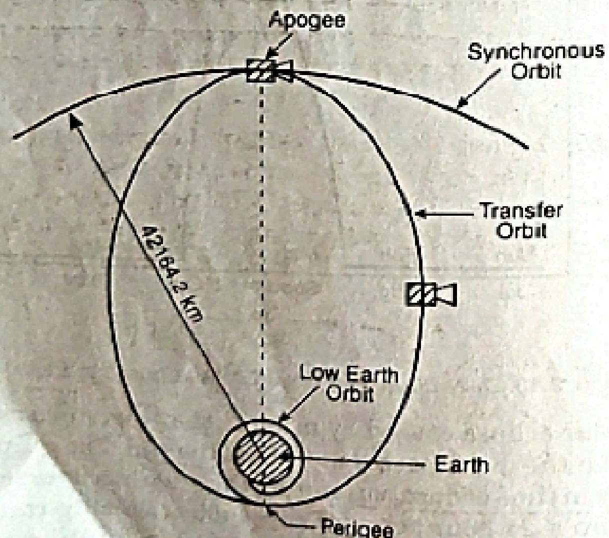

Fig. 2.14. Satellite Placement in Synchronous (Geostationary) Orbit from a low Earth Orbit.

follows this technique. In the second technique, there is no initial circular orbit and the vehicle provides the necessary velocity at the perigee of the elliptical transfer orbit. Thus here only one velocity increment is required from the satellite at the apogee. This technique has been used by expendable launch vehicles such as Ariane, Delta or Atlas-Centaur launchers. In the third technique, satellites are directly placed into geostationary (geosynchronous) orbit. This is achieved by special expendable launch vehicles such as US Titan III *C* and the USSR Proton launchers.

The velocity of satellite (V_s) at the perigee and apogee of the elliptical transfer orbit can be calculated from Eq. (2.7), namely

$$V_s^2 = \frac{2\mu}{r} - \frac{\mu}{a} \qquad \ldots(2.40)$$

where a is the semi-major axis of the ellipse, μ is gravitational constant of the earth (μ = 398600 km^3/s^2). r is the distance from the center of the earth to the point at which the velocity V is required to maintain the ellipse.

Thus at perigee (of 300 km), r = 6678.2 km, a = (6678.2 + 42164.2)/2 = 24421.2 km. The velocity is then V_{ps} = 10.15 km/s. At the Apogee, r = 42164.2 km and there the velocity V_{AS} = 1.61 km/s. Since the velocity in a synchronous orbit ($r = a$ = 42164.2 km) is V_{cs} = 3.07 km/s so the required velocity increment at apogee is $\Delta V_{cs} = V_{cs} - V_{as}$ = 3.07 km/s – 1.61 km/s = 1.46 km/s. This velocity increment value corresponds to a minimum and applies to a circularization manoeuvre when transfer orbit and circular orbit lie in the same plane.

The above discussions about velocity increment at apogee of transfer orbit is true when the launch is carried out at the equator (0° latitude). In case the satellite is launched from other sites (launching station latitudes other than 0°) then the satellite's final

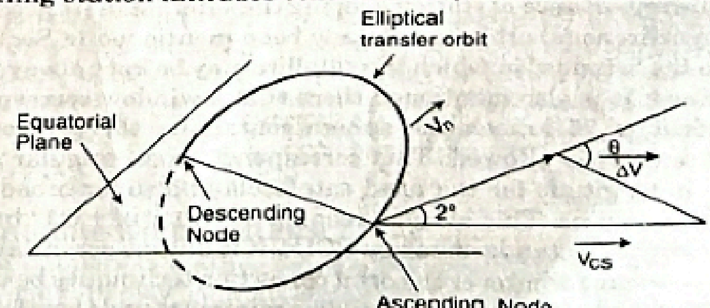

Fig. 2.15. Inclination Correction and Orbit Circularization.

orbit would be a synchronous orbit with an inclination i greater than or equal to the latitude of the launch point. It is therefore then essential to have an *orbit inclination correction manoeuvre*. Since the inclination of transfer orbit is the angle between the plane of the transfer orbit and the equatorial plane, correction of this inclination will help in converting the transfer orbit plane into the equatorial plane. Fig. 2.15 shows this orbit inclination correction. This orbit inclination correction is obtained by applying a velocity increment at one of nodes of the orbit which will cause a resultant velocity vector in the equatorial plane. Thus the magnitude of the incremental velocity vector ΔV required to correct the inclination is

$$\Delta V = \sqrt{V_n^2 + V_{cs}^2 - 2V_n\, V_{cs} \cos (i)} \qquad \text{...(2.41)}$$

where V_n is the velocity magnitude at node and V_{cs} is the satellite velocity in geostationary orbit in the equatorial plane. If the line connecting the apogee and perigee is the node line and the inclination correction is made at the apogee in conjunction with orbit circularization, then

$$\Delta V = \sqrt{V_{as}^2 + V_{cs}^2 - 2V_{as}\, V_{cs} \cos (i)} \qquad \text{...(2.42)}$$

Thus for $i = 28$, $V_{AS} = 1.61$ km/s and $V_{CS} = 3.07$ km/s, the incremental velocity required to correct the orbit inclination and to achieve orbit circularization obtained from Eq. (2.42) is $\Delta V = 1.81$ km/s. It is evident from Fig. 2.15 that the angle θ at which the main axis of satellite is correctly oriented relative to the satellite velocity in a plane perpendicular to the line of nodes is derived from expression given by

$$\theta = Arc \sin\left(\frac{V_{cs} \sin (i)}{\Delta V}\right) \qquad \text{...(2.43)}$$

Since V_{cs} is nearly equal to V_{as} so θ is nearly equal to $2i$.

2.9. Station keeping

The importance of station keeping the satellite in its geostationary (synchronous) orbit has already been mentioned in Section 2.1 where the 'window' in which the satellite may be kept always is also explained. It is also mentioned there that a window corresponding to a drift of 75 km on the sphere containing the geostationary satellite orbit is allowed. This corresponds to an angular shift of $\pm 0.1°$ in longitude for the fixed satellite and also for broadcasting satellite services. This station keeping accuracy of $\pm 0.1°$ has been internationally standardized by CCIR. It is always essential that in station keeping whatever the orbit correction techniques be utilized, the propellent consumption should be minimum and therefore some kind of strategy is followed. The strategy usually has several steps as given below. Firstly the direction and speed of drift of the satellite is determined. Then by extrapolation prediction is made as to which day the satellite will escape the window and a few days before this date the true orbit is determined using a new series of measurement. After that calculations are carried out regarding the date, the amplitude and duration of the velocity increments required to modify the orbital parameters. Appropriate thrusters are then fired and then the effects of correction are monitored.

Compensation of the inclination increments correspond to *North-South (N-S) station keeping* and requires a thrust impulse perpendicular to the orbital plane. *East-West (E-W) station keeping* (as well as quick relocation to a different satellite position in the geostationary orbit) are obtained by applying thrusts in the orbital plane. The above mentioned station keeping accuracy of $\pm 0.1°$ actually corresponds to *E-W* accuracy. Internationally there is no such demand on *N-S* station keeping accuracy. However, it should be noted that *NS* corrections are considerably more costly in terms of fuel expenditure for the reaction equipment (thrusters) and therefore for moderate *NS* movements correction are not carried out at all. For this reason, the satellite is launched into a slightly inclined orbit such that inclination decreases from its maximum value (i_{max}) to zero and then again increases to i_{max}. Then if the planned lifetime of the satellite is T_L years, i_{max} would be approximately

$$i_{max} = \frac{1}{2}\, i_{av}\, T_L \qquad \text{...(2.44)}$$

were i_{av} is the natural annual change of inclination averaged over the lifetime T_L. A typical value for long lived satellite is $i_{av} = 0.86°$ which corresponds to the velocity increment correction of $\Delta V = 460$ m/s. Compared to this, the required velocity increment correction for east-west station keeping of $\pm 0.1°$ is around 27 m/s which is much less than required for *NS* station keeping. Thus if the ground network could tolerate complete absence of *NS* corrections of the orbit of a nominally geostationary satellite, more than 90% of the thruster fuel for station keeping may be saved. This allows for an extra useful payload on board the satellite. Though this is an attractive tradeoff but involves careful consideration of the wider pointing or beamwidth margins of the antennas required in the absence of *N-S* station keeping.

2.10. Satellite Stabilization

Even if a perfect station keeping manoeuvre is carried out, the satellite may display motions about its center of mass. This will disturb the pointing of satellite antennas narrow beam towards the earth. It is therefore essential that the satellite be perfectly balanced. It may be noted that the disbalancing of satellite is caused by several reasons such as gravitational force from the sun, moon, and planets, solar pressure acting on the antennas, spacecraft body and solar cells, and earth's magnetic field etc. These forces vary cyclically through a 24th period and cause wobbling of the spacecraft which requires to be damped out mechanically. This is achieved by the techniques called *Satellite stabilization or attitude control*.

The disbalanced satellite may display its motion on either of the three axes as shown in Fig. 2.16. These axes are termed yaw, roll and pitch axes. If the satellite is stabilized about these axes, the stabilization is termed as *three axis body stabilization* and the corresponding satellite is calld *three axis body stabilized satellite*. Since a rotating body offers an inherent gyroscopic stiffness to torques tending to disturb the orientation of the rotational axis, so this technique may be used to stabilize the satellite. This kind of satellite stabilization is called *spin stabilization*. Fig. 2.17 shows the spin stabilized satellite. Here the spin stabilization is achieved by rotation of the geostationary satellite body between 30 and 120 rpm creating an inertial stiffness which maintains the satellite spin axis perpendicular to the equatorial plane. The satellite spin axis is thus North/South oriented. The spin stabilization has the disadvantage that it requires the use of a de-spun antenna. Thus here the satellite is cylindrical spinning about the axis of the cylinder. Solar cells cover the cylindrical surface and antennas are mounted on a de-spun plateform. The despun section is kept stationary by counter rotation so that the antennas mounted on it are constantly pointing toward the earth. In the three axis body stabilized satellite as shown in Fig. 2.18 a spinning momentum wheel within the satellite establishes

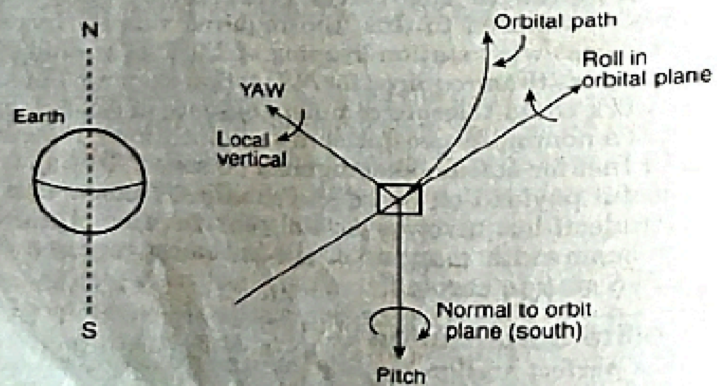

Fig. 2.16. Spin Axes of Geosynchronous Satellite.

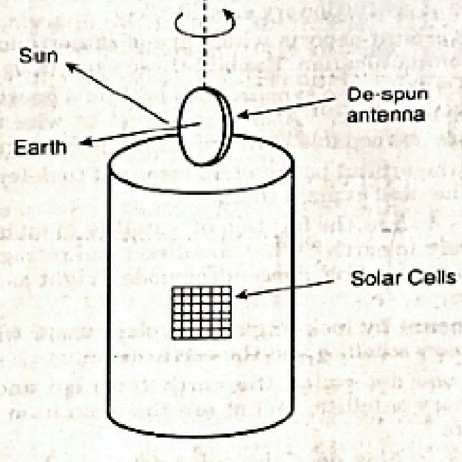

Fig. 2.17. Spin Stabilized Satellite.

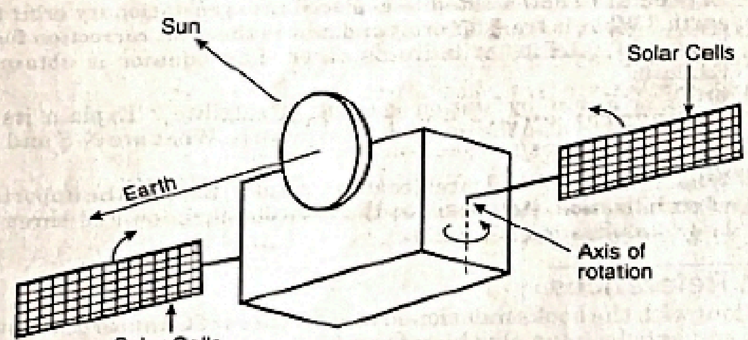

Fig. 2.18. Three Axis (body) Stabilized Satellite.

the same frame of reference as the spinning cylinder of the spin stabilized satellite. In this case, however the entire spacecraft is effectively the despun plateform which maintains its orientation in space relative to the momentum wheel axis. The solar cells in such kind of stabilized satellite are mounted on panels external to the spacecraft body, allowing them to be oriented toward the sun. The control system is therefore more complex for a three axis body stabilized satellite.

2.11. Review Questions

1. Explain Kepler's laws of planetary rotation. How are these applied to the case of geostationary satellite ?
2. Name the orbital aspects which are of importance in synchronous satellite communication. Explain these aspects in brief.
3. Explain as to how the synchronous orbit of a geostationary satellite is being determined ? Also explain as to why the geostationary satellite are not capable of illuminating polar regions ?
4. What are the orbital parameters required to determine a satellite's orbit ? Name and explain them.
5. Explain as to how the location of satellite in an orbit is carried out with respect to earth ? What are direct and retrograde orbits ? Also explain the ascending, descending nodes, right ascension and nodal regression.
6. What is meant by look angles ? Explain them with reference to a geostationary satellite and the earth station.
7. How can one determine the earth coverage and slant range for geostationary satellite ? What are the maximum values of the two parameters ?
8. Explain as to how does the solar eclipse affect the working of a communication satellite ? Mention the duration and the month when the eclipse effects are maximum ?
9. Explain as to how a satellite is placed into geostationary orbit from earth ? What is transfer orbit and how is the orbit correction for the launch of satellite at latitutde other than equator is obtained ? Explain.
10. What is meant by station keeping of satellite ? Explain its significance and also the methods to achieve it. What are N-S and E-W station keeping ?
11. What is meant by the satellite stabilization ? Explain the importance of stabilization. Also explain the spin stabilization and three axis body stabilization of satellite.

2.12. References

Alongwith the books mentioned in references of Chapter 1, following books and articles may also be referred.

1. Ash, M.E., Determination of Earth satellite orbits, Tech Note 1972-5. Lincoln Laboratory, MIT, April, 1972.
2. Ash, M.E., Communications Terminal Orbit Calculations. Tech Note 1973-74, Lincoln Lab, MIT, May 24, 1973.
3. Kaplan, M.H., Modern Spacecraft Dynamics and Control, New York, Wiley, 1976.
4. Miya, K., Satellite Communications Engineering, Tokyo, Lattice, 1975.
5. Martin, J., Communications Satellite Systems, Englewood Cliffs, NJ, Prentice Hall, 1979.
6. Fithenakis, E., 1984, Manuals of Satellite Communications, NY. McGraw Hill.

7. Van Trees, H.L., Ed, 1979, Satellite Communications, NY, IEEE Press.
8. Mittra, R., Imbriade, W.A., and Maariders, E.J., Satellite Communication Antenna Technology, North Holland, Amsterdam, 1983.
9. Avdnyevsky, V.S., Uspensky, G.R., Scientific and Economy Oriented Space Systems, MIR Publishers, 1988, Moscow.
10. Smith, F.L., 1972, A Nomogram for look to geostationary satellites, IEEE Trans Aerospace and Electtron Sys., Vol AES No. 5 (May), p. 394.
11. CCIR, 1982, Report 843, Station Keeping Techniques for Geostationary Satellites, Vol II, Geneva.