SATELLITE & OPTICAL COMMUNICATION-MOD 1 NOTES –BEC515D

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MODULE-1

SATELLITE ORBITS AND TRAJECTORIES

The study of orbits and trajectories of satellites and satellite launch vehicles is the most fundamental topic of the subject of satellite technology and perhaps also the most important one. It is important because it gives an insight into the operational aspects of this wonderful piece of technology. An understanding of the orbital dynamics would give a sound footing to address issues like types of orbits and their suitability for a given application, orbit stabilization, orbit correction and station keeping, launch requirements and typical launch trajectories for various orbits, Earth coverage and so on.

1.1 Definition

While a trajectory is a path traced by a moving body, an orbit is a trajectory that is periodically repeated. While the path followed by the motion of an artificial satellite around Earth is an orbit, the path followed by a launch vehicle is a trajectory called the launch trajectory.

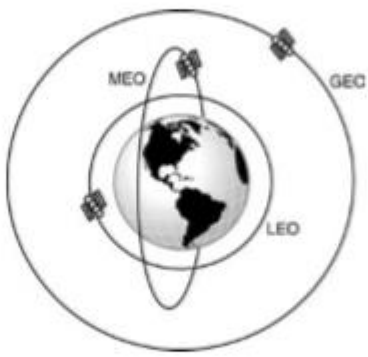


Fig.1.1 Example of orbital motion-satellite revolving around earth

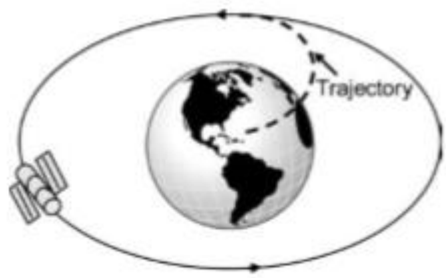


Fig.1.2. Example of trajectory-path followed by a rocket on its way during satellite launch

The motion of different planets of the solar system around the sun and the motion of artificial satellites around Earth (Fig.1.1) are examples of orbital motion. The term ‘trajectory’, on the other hand, is associated with a path that is not periodically revisited. The path followed by a rocket on its way to the right position for a satellite launch (Fig. 1.2).The path followed by orbiting satellites when they move from an intermediate orbit to their final destined orbit (Fig.1.3) is examples of trajectories.

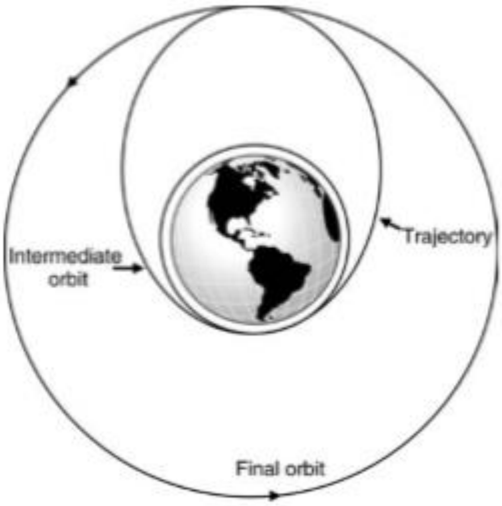


Fig.1.3 Example of trajectory-motion of a satellite from the intermediate orbit to the final orbit

1.2 Basic Principles

The motion of natural and artificial satellites around Earth is governed by two forces. One of them is the centripetal force directed towards the centre of the Earth due to the gravitational force of attraction of Earth and the other is the centrifugal force that acts outwards from the centre of the Earth (Fig.1.4).

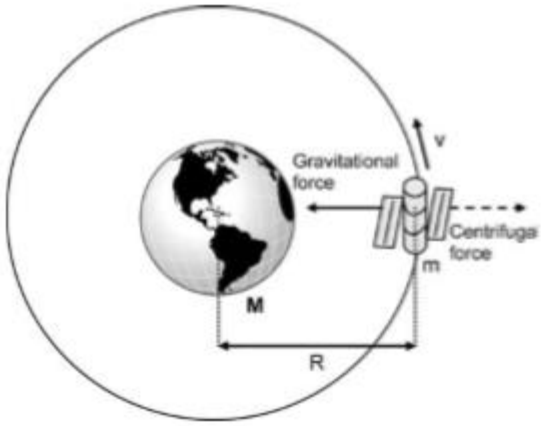


Fig.1.4 Gravitational force and the centrifugal force acting on the bodies orbiting earth.

It may be mentioned here that the centrifugal force is the force exerted during circular motion, by the moving object upon the other object around which it is moving. In the case of a satellite orbiting Earth, the satellite exerts a centrifugal force. However, the force that is causing the circular motion is the centripetal force. In the absence of this centripetal force, the satellite would have continued to move in a straight line at a constant speed after injection. The centripetal force directed at right angles to the satellite's velocity towards the centre of the Earth transforms the straight line motion to the circular or elliptical one, depending upon the satellite velocity

1.3 Orbital Parameters

The satellite orbit, which in general is elliptical, is characterized by a number of parameters. These not only include the geometrical parameters of the orbit but also parameters that define its orientation with respect to Earth.

1.3.1 Ascending and descending nodes

The satellite orbit cuts the equatorial plane at two points: the first, called the descending node (N1),where the satellite passes from the northern hemisphere to the southern hemisphere, and the second, called the ascending node(N2),where the satellite passes from the southern hemisphere to the northern hemisphere(Fig.1.5).

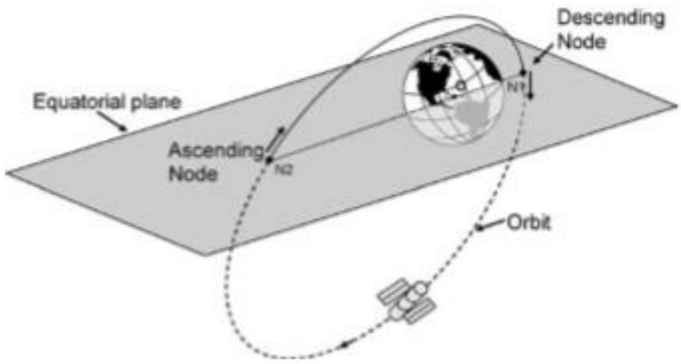


Fig.1.5 Ascending and Descending nodes

1.3.2 Equinoxes

The inclination of the equatorial plane of Earth with respect to the direction of the sun, defined by the angle formed by the line joining the centre of the Earth and thesun with the Earth’s equatorial plane follows a sinusoidal variation and completes one cycle of sinusoidal variation over a period of 365 days (Fig.1.6). The sinusoidal variation of the angle of inclination is defined by Inclination angle (in degrees)= 23.4 sin ($2\pi t/T$).

Where T is365days.Thisexpressionindicatesthattheinclinationangleiszerofort = T/2 and T. This is observed to occur on 20-21 March, called the spring equinox, and 22-23 September, called the autumn equinox. The two equinoxes are understandably spaced 6 months apart.

During the equinoxes, it can be seen that the equatorial plane of Earth will be aligned with the direction of the sun. Also, the line of intersection of the Earth’s equatorial plane and the Earth’s orbital plane that passes through the centre of the Earth is known as the line of equinoxes.

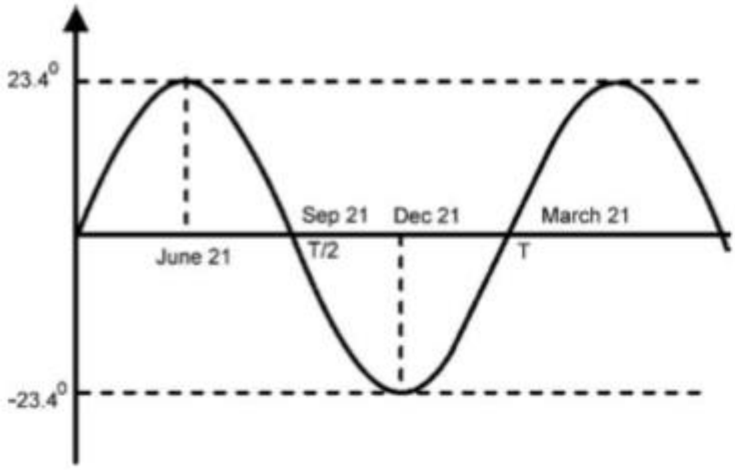


Fig.1.6 Yearly variation of angular inclination of earth with the sun.

The direction of this line with respect to the direction of the sun on 20-21 March determines a point at infinity called the vernal equinox (Y) (Figure 1.7).

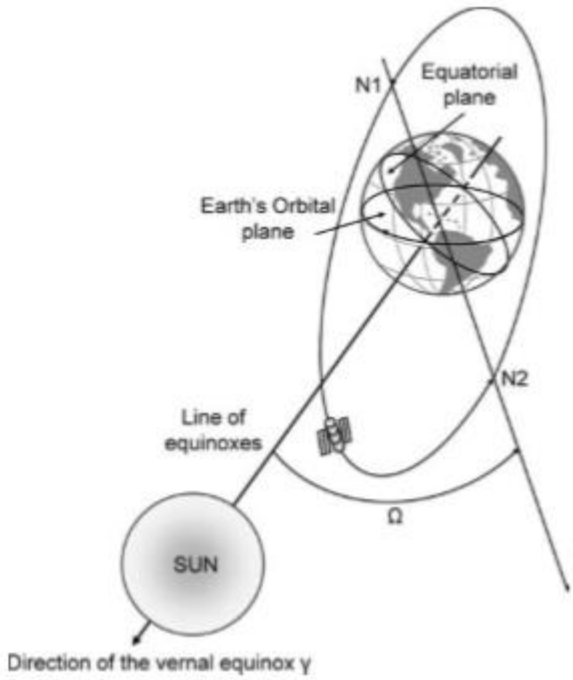


Fig.1.7 Vernol equinox

1.3.3 Solstices

Solstices are the times when the inclination angle is at its maximum, i.e. 23.4◦. Thesealsooccurtwiceduringayearon20-21June,calledthesummersolstice,and21-22 December, called the winter solstice.

1.3.4 Apogee

Apogee is the point on the satellite orbit that is at the farthest distance from the centre of the Earth (Fig.1.8).

The apogee distance can be computed from the known values of the orbit eccentricity e and the semi-major axis a from Apogee distance= a(1+e).

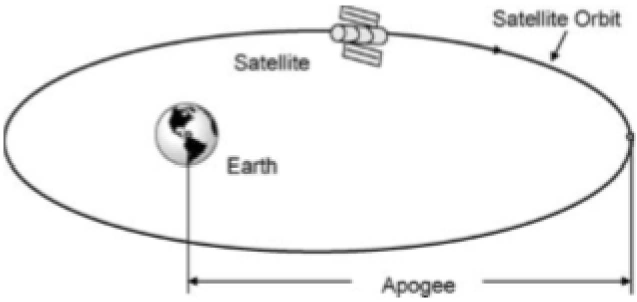


Fig.1.8 Apogee

The apogee distance can also be computed from the known values of the perigee distance and velocity at the perigee Vp from

Vp = √(2μ / Perigee distance − 2μ / Perigee distance+Apogee distance)

Where

Vp = V(d cosγ / Perigee distance)

With V being the velocity of the satellite at a distance d from the centre of the Earth.

1.3.5 Perigee

Perigee is the point on the orbit that is nearest to the centre of the Earth (Fig.1.9).

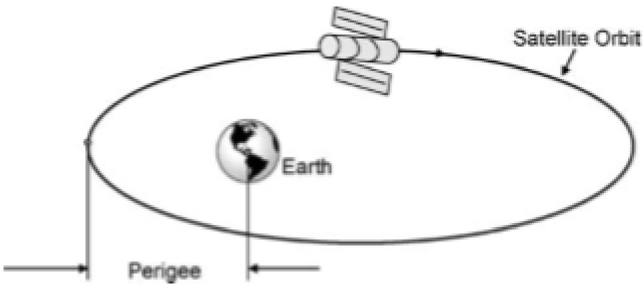


Fig.1.9 Perigee

The perigee distance can be computed from the known values of orbit eccentricity e and the semi-major axis a from

Perigee distance= a(1−e).

1.3.6 Eccentricity

The orbit eccentricity e is the ratio of the distance between the centre of the ellipseandthecentreoftheEarthtothesemi-majoraxisoftheellipse.Itcanbecomputed from any of the following expressions:

e = apogee−perigee / apogee + perigee

e = apogee−perigee / 2a.

Thus **e =$\sqrt{(a^2 - b^2)}$/a,** where a and b are semi-major and semi-minor axes respectively.

1.3.7 Semi-major axis

This is a geometrical parameter of an elliptical orbit. It can, however, be computed from known values of apogee and perigee distances as

a = apogee + perigee / 2.

1.3.8 Right ascension of the ascending node

The right ascension of the ascending node tells about the orientation of the line of nodes, which is the line joining the ascending and descending nodes, with respect to the direction of the vernal equinox. It is expressed as an angle measured from the vernal equinox towards the line of nodes in the direction of rotation of Earth (Fig.1.10). The angle could be anywhere from 0◦ to 360◦.

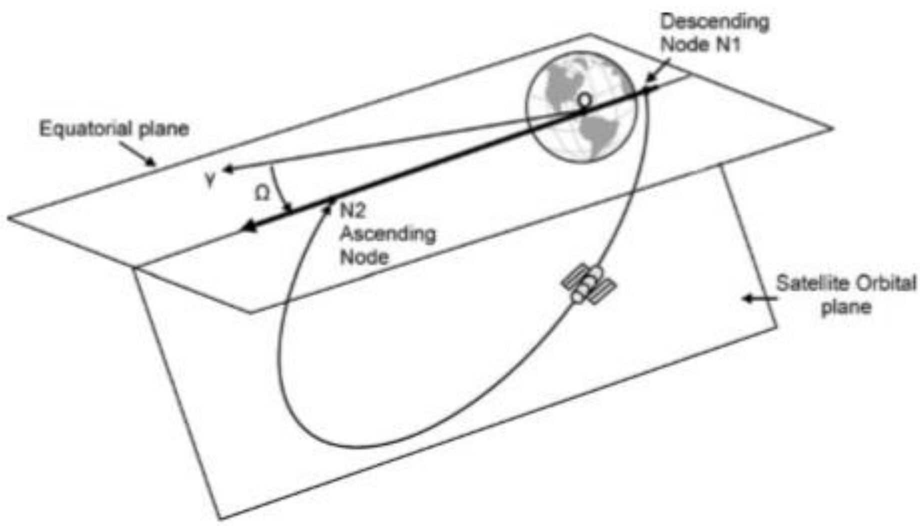


Fig.1.10 Right ascension of the ascending node

Acquisition of the correct angle of right ascension of the ascending node (**Ω**)is important to ensure that the satellite orbits in the given plane. This can be achieved by choosing an appropriate injection time depending upon the longitude.

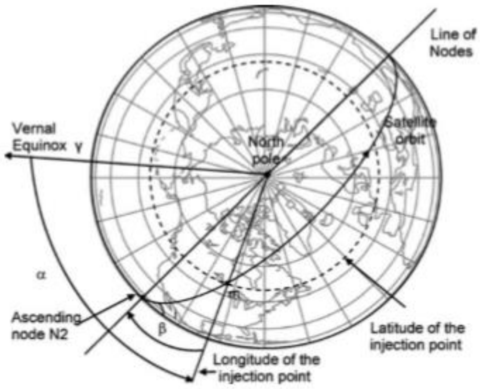


Fig.1.11 Computation of the right ascension of the ascending node.

Angle Ω can be computed as the difference between two angles. One is the angle α between the direction of the vernal equinox and the longitude of the injection point and the other is the angle β between the line of nodes and the longitude of the injection point, as shown in Fig.1.11 Angle β can be computed from

$$\textbf{Sin } \beta = \cos i \sin l / \cos l \sin i.$$

Where ∠i is the orbit inclination and l is the latitude at the injection point.

1.3.9 Inclination

Inclination is the angle that the orbital plane of the satellite makes with the Earth's equatorial plane. It is measured as follows. The line of nodes divides both the Earth's equatorial plane as well as the satellite's orbital plane into two halves. Inclination is measured as the angle between that half of the satellite's orbital plane containing the trajectory of the satellite from the descending node to the ascending node to that half of the Earth's equatorial plane containing the trajectory of a point on the equator from n1 to n2, where n1 and n2 are respectively the points vertically below the descending and ascending nodes (Fig.1.12). The inclination angle can be determined from the latitude l at the injection point and the angle A_Z between the projection of the satellite's velocity vector on the local horizontal and north. It is given by

$$\textbf{Cos } i = \sin A_Z . \cos l.$$

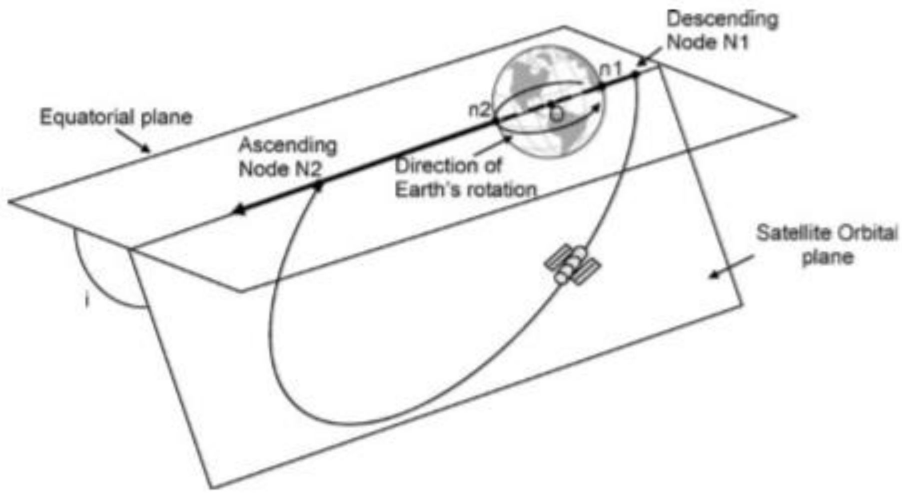


Fig.1.12 Angle of inclination

1.3.10 Argument of the perigee

This parameter defines the location of the major axis of the satellite orbit. It is measured as the angle ω between the line joining the perigee and the centre of the Earth and the line of nodes from the ascending node to the descending node in the same direction as that of the satellite orbit (Fig.1.13).

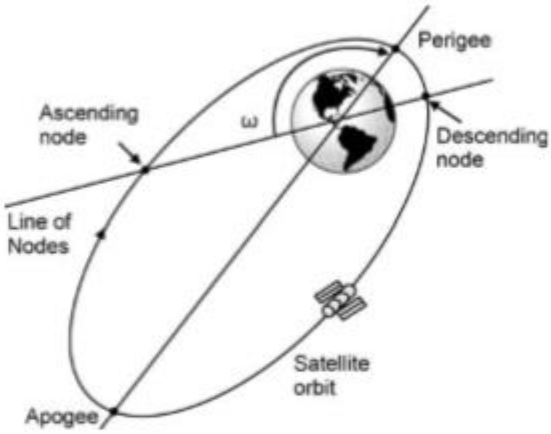


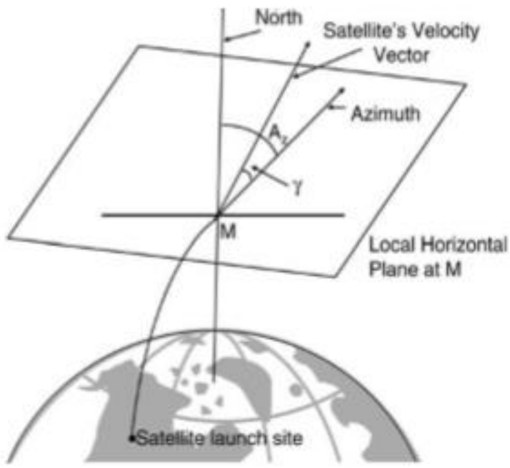
Fig.1.13 Argument of perigee.

1.3.11 True anomaly of the satellite

This parameter is used to indicate the position of the satellite in its orbit. This is done by defining an angle θ, called the true anomaly of the satellite, formed by the line joining the perigee and the centre of the Earth with the line joining the satellite and the centre of the Earth (Fig.1.14).

1.3.12 Angles defining the direction of the satellite.

The direction of the satellite is defined by two angles, the first by angle γ between the direction of the satellite's velocity vector and its projection in the local horizontal and the second by angle Az between the north and the projection of the satellite's velocity vector on the local horizontal (Fig1.15).



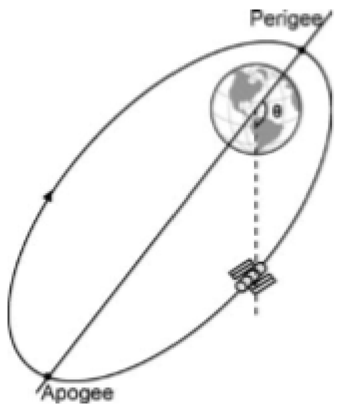


Fig.1.14 True anomaly of a satellite

Fig.1.15 Angles deflecting the direction of the satellite.

1.4 Injection Velocity and Satellite Trajectory

The horizontal velocity with which a satellite is injected into space by the launch vehicle With the intention of imparting a specific trajectory to the satellite has a direct bearing on theSatellite trajectory. The phenomenon is best explained in terms of the three cosmic velocities.The general expression for the velocity of a satellite at the perigee point (VP), assuming an elliptical orbit, is given by

$$\mathbf{VP = \sqrt{[(2\mu / r) - (2\mu / R + r)]}.}$$

Where

R =apogee distance

r =perigee distance

μ = GM =constant

The first cosmic velocity V1 is the one at which apogee and perigee distances are equal, i.e. R = r, and the orbit is circular. The above expression then reduces to

$$\mathbf{V1 = \sqrt{(\mu / r)}.}$$

1.5 Types of satellite Orbits

As described at length in the earlier pages of this chapter, satellites travel around Earth along predetermined repetitive paths called orbits. The orbit is characterized by its elements or parameters, which have been covered at length in earlier sections. The orbital elements of a particular satellite depend upon its intended application. The satellite orbits can be classifiedon the basis of:

1.5.1 Orientation of the orbital plane

The orbital plane of the satellite can have various orientations with respect to the equatorial plane of Earth.Theanglebetweenthetwoplanesiscalledtheangleofinclinationofthesatellite. On this basis, the orbits can be classified as equatorial orbits, polar orbits and inclined orbits. In the case of an equatorial orbit, the angle of inclination is zero, i.e. the orbital plane of the satellite coincides with the Earth's equatorial plane (Fig.1.16).

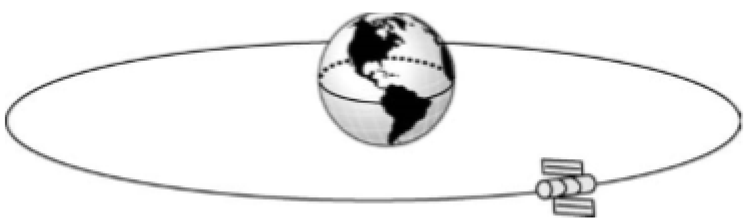


Fig.1.16 Equatorial orbit

A satellite in the equatorial orbit has latitude of 0°. For an angle of inclination equal to 90°, the satellite is said to be in the polar orbit (Fig.1.17). For an angle of inclination between 0° and 180°, the orbit is said to be an inclined orbit.

Fig.1.17 Polar orbit

Forinclinationsbetween0°and90°,thesatellitetravelsinthesamedirectionasthedirection of rotation of the Earth. The orbit in this case is referred to as a direct or prograde orbit(Fig.1.18).

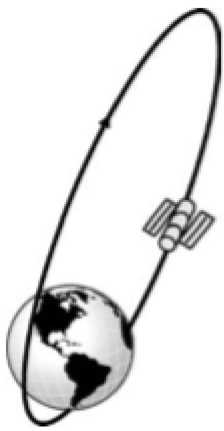


Fig.1.18 Prograde orbit

Fig.1.19 Retrograde orbit

Forinclinationsbetween90°and180°,thesatelliteorbitsinadirectionopposite to the direction of rotation of the Earth and the orbit in this case is called a retrograde orbit (Fig.1.19).

1.5.2 Eccentricity of the Orbit

On the basis of eccentricity, the orbits are classified as elliptical (Fig.1.20 (a)) and circular (Fig.1.20 (b)) orbits. Needless to say, when the orbit eccentricity lies between 0 and 1, the orbit is elliptical with the centre of the Earth lying at one of the foci of the ellipse. When the eccentricity is zero, the orbit becomes circular. It may be mentioned here that all circular orbits are eccentric to some extent. As an example, the eccentricity of orbit of geostationary

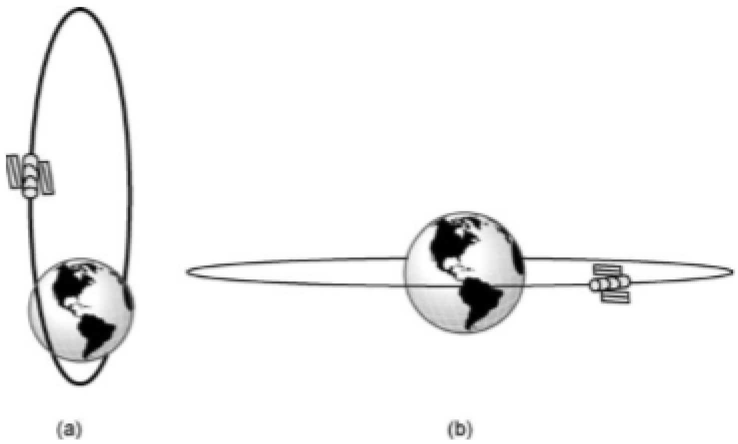


Fig.1.20 (a) elliptical orbit and (b) circular orbit

Satellite INSAT-3B, an Indian satellite in the INSAT series providing communication and meteorological services, is 0.0002526. Eccentricity figures for orbits of GOES-9 and Meteosat-7geostationarysatellites, bothofferingweatherforecastingservices,are0.0004233 and 0.0002526 respectively.

1.5.3 Distance from Earth

Again depending upon the intended mission, satellites may be placed in orbits at varying distances from the surface of the Earth. Depending upon the distance, these are classified as lowEarthorbits(LEOs),mediumEarthorbits(MEOs)andgeostationaryEarthorbits(GEOs), as shown in Fig.1.21.

Satellites in the low Earth orbit (LEO) circle Earth at a height of around 160 to 500km above the surface of the Earth. These satellites, being closer to the surface of the Earth, have much shorter orbital periods and smaller signal propagation delays.A lower propagation delay makes them highly suitable for communication applications.

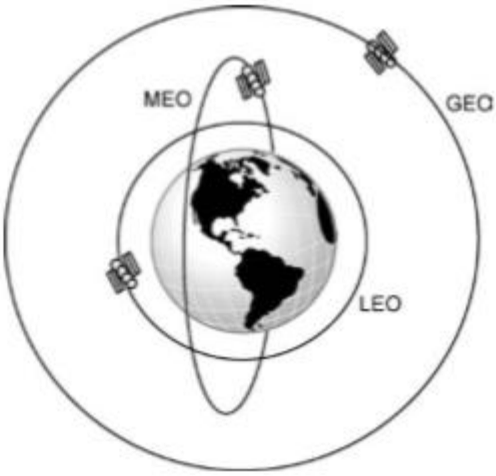


Fig.1.21 LEO, MEO and GEO

Due to lower propagation paths, the power required for signal transmission is also less, with the result that the satellites are of small physical size and are inexpensive to build. However, due to a shorter orbital period, of the order of an hour and a half or so, these satellites remain over a particular ground station for a short time. Hence, several of these satellites are needed for 24 hour coverage. One important application of LEO satellites for communication is the project Iridium, which is a global communication system conceived by Motorola (Fig.1.22). A total of 66 satellites are arranged in a distributed architecture, with each satellite carrying 1/66 of the total system capacity. The system is intended to provide a variety of telecommunicationservicesatthegloballevel.Theprojectisnamed 'Iridium'asearliertheconst elation wasproposed to have 77 satellites and the atomic number of iridium is 77. Other applications where LEO satellites can be put to use are surveillance, weather forecasting, remote sensing and scientific studies.

Fig.1.21 Iridium constellation of satellites.

Medium Earth orbit (MEO) satellites orbit at a distance of approximately 10000 to 20000km above the surface of the Earth. They have an orbital period of 6 to 12 hours. These satellites stay in sight over a particular region of Earth for a longer time. The transmission distance and propagation delays are greater than those for LEO satellites. These orbits are generally polar in nature and are mainly used for communication and navigation applications.

A geosynchronous Earth orbit is a prograde orbit whose orbital period is equal to Earth's rotational period. If such an orbit were in the plane of the equator and circular, it would remain stationary with respect to a given point on the Earth. These orbits are referred to as the geostationary Earth orbits (GEOs). For the satellite to have such an orbital velocity, it needs to be at a height of about 36000km, 35786km to be precise, above the surface of the Earth. To be more precise and technical, in order to remain above the same point on the Earth's surface, a satellite must fulfil the following conditions:

1. It must have constant latitude, which is possible only at 0◦ latitude.
2. The orbit inclination should be zero.
3. It should have a constant longitude and thus have a uniform angular velocity, which is possible when the orbit is circular.
4. The orbital period should be equal to 23 hours 56 minutes, which implies that the satellite must orbit at a height of 35 786km above the surface of the Earth.
5. The satellite motion must be from west to east.

In the case where these conditions are fulfilled, then as the satellite moves from a position O1 toO2 in its orbit, appoint vertically below on the equator moves with the same angular velocity and moves from E1 to E2, as shown in Fig.1.22 Satellites in geostationary orbits play an essential role in relaying communication.

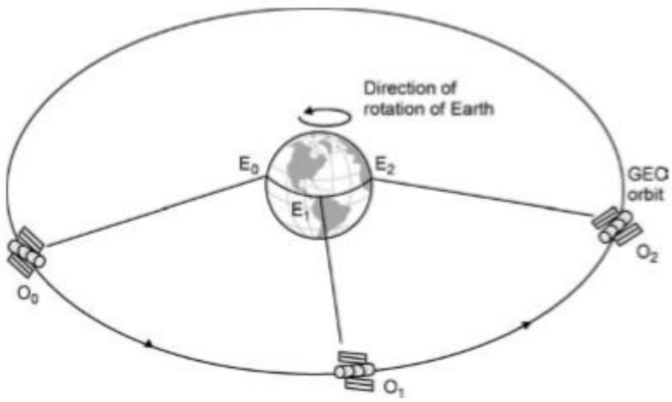


Fig.1.22 GEO satellites appear stationary wrt to a point on Earth

1.6 Orbital Perturbations

The satellite once placed in its orbit, experiences various perturbing torques that cause variations in its orbital parameters with time. These include gravitational forces from other bodieslike solar and lunar attraction, magnetic field interaction, solar radiation pressure, asymmetry of Earth's gravitational field etc. Due to these factors, the satellite drift and its orientation also changes and hence the true orbit of the satellite is different from that defined using Kepler's laws. The satellite's position thus needs to be controlled both in the east–west as well as the north – south directions. The east–west location needs to be maintained to prevent radio frequency (RF) interference from neighbouring satellites. It may be mentioned here that in the case of a geostationary satellite, a 1◦ drift in the east or west direction is equivalent to a drift of about 735 km along the orbit (Fig.1.23). The north–south orientation has to be maintained to have proper satellite inclination. The Earth is not a perfect sphere and is flattened at the poles. The equatorial diameter is about 20–40 km more than the average polar diameter. Also, the equatorial radius of the Earth is not constant. In addition, the average density of Earth is not uniform. All of this results in a non-uniform gravitational field around the Earth which in turn results in variation in gravitational force acting on the satellite due to the Earth. The effect of variation in the gravitational field of the Earth on the satellite is more predominant for geostationary satellites than for satellites orbiting in low Earth orbits as in the case of these satellites the rapid change in the position of the satellite with respect to the Earth's surface will lead to the averaging out of the perturbing forces. In the case of a geostationary satellite, these forces result in an acceleration or deceleration component that varies with the longitudinal location of the satellite.

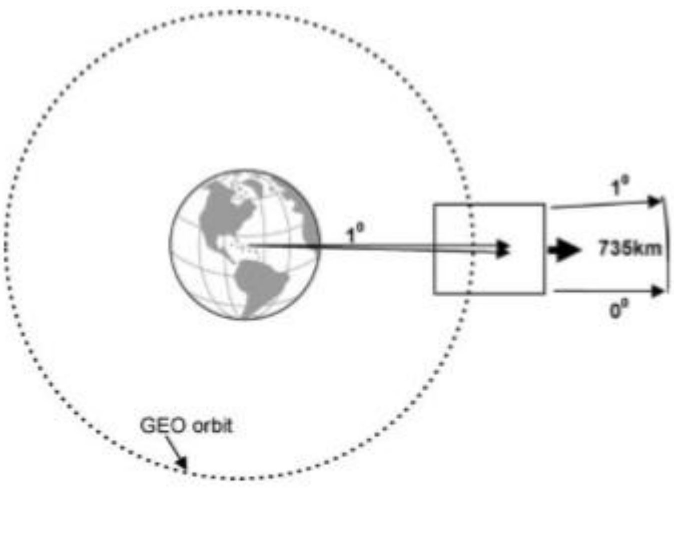


Fig.1.23 Drift of a geostationary satellite.

As the perturbed orbit is not an ellipse anymore, the satellite does not return to the same point in space after one revolution. The time elapsed between the successive perigee passages is referred to as anomalistic period. The anomalistic period (TA) is given by

$$t_A = 2\pi / \omega_{mod.}$$

Where,

$$\omega_{mod} = \omega 0[1+ K(1-1.5\sin^2 i) / a^2(1-e^2)^{3/2}].$$

ω0 is the angular velocity for spherical Earth, K =66063.1704km2,aisthesemi-majoraxis, e is the eccentricity and i =cos−1 WZ, WZ is the Z axis component of the orbit normal. The attitude and orbit controlsystemmaintainsthesatellite'spositionanditsorientationand keeps the antenna pointed correctly in the desired direction (bore-sighted to the center of the coverage area of the satellite).The orbit control is performed by firing thrusters in the desired directionorbyreleasingjetsofgas.Itisalsoreferredtoasstationkeeping.Thrustersandgasjets areusedtocorrectthelongitudinaldrifts(in-planechanges)andtheinclinationchanges(out-ofplane changes).

1.7 Satellite Stabilization

Commonly employed techniques for satellite attitude control include:

1.7.1 Spin stabilization

In a spin-stabilized satellite, the satellite body is spun at a rate between 30 and 100 rpm about an axis perpendicular to the orbital plane (Fig.1.24). Like a spinning top, the rotating body offers inertial stiffness, which prevents the satellite from drifting from its desired orientation. Spin-stabilized satellites are generally cylindrical in shape. For stability, the satellite should be spun about its major axis, having a maximum moment of inertia. To maintain stability, the moment of inertia about the desired spin axis should at least be 10% greater than the moment of inertia about the transverse axis

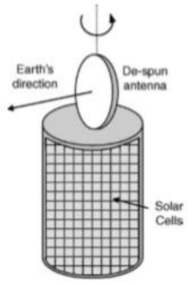


Fig.1.24 Spin stabilized satellite

Therearetwotypesofspinningconfigurationsemployedinspin-stabilizedsatellites.These includethesimplespinnerconfigurationandthedualspinnerconfiguration.Inthesimplespinner configuration,thesatellitepayloadandothersubsystemsareplacedinthespinningsection, whiletheantennaandthefeedareplacedinthede-spunplatform.Thede-spunplatformisspun in a direction opposite to that of the spinning satellite body. In the dual spinner configuration, the entire payload along with the antenna and the feed is placed on the de-spun platform and the other subsystems are located on the spinning body.

1.7.2 Three-axis or body stabilization.

In the case of three-axis stabilization, also known as body stabilization, the stabilization is achieved by controlling the movement of the satellite along the three axes, i.e. yaw, pitch and roll, with respect to a reference (Fig.1.25). The system uses reaction wheels or momentum

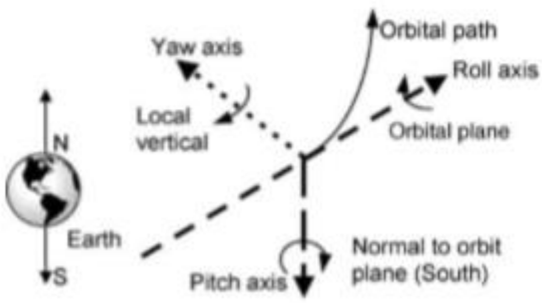


Fig.1.25 Three-axis stabilization.

Wheels to correct orbit perturbations. The stability of the three-axis system is provided by the active control system, which applies small corrective forces on the wheels to correct the undesirable changes in the satellite orbit.Most three-axis stabilized satellites use momentum wheels. The basic control technique used here is to speed up or slow down the momentum wheel depending upon the direction in which the satellite is perturbed. The satellite rotates in a direction opposite to that of speed changeofthewheel.Forexample,anincreaseinspeedofthewheelintheclockwisedirection will make the satellite to rotate in a counterclockwise direction. The momentum wheels rotate in one direction and can be twisted by a gimbal motor to provide the required dynamic force on the satellite.

1.8 Orbital effects on satellite’s Performance

AsweknowthesatelliteisrevolvingconstantlyaroundtheEarth.Themotionofthesatellitehas significant effects on its performance.

These include the Doppler shift, effect due to variation in the orbital distance, effect of solar eclipse and sun's transit outrage.

1.8.1 Doppler Shift

Thegeostationarysatellitesappearstationarywithrespecttoan Earthstationterminalwhereas in the case of satellites orbiting in low Earth orbits, the satellite is in relative motion with respect to the terminal. However, in the case of geostationary satellites also there are some variations between the satellite and the Earth station terminal. As the satellite is moving with respect to the Earth station terminal, the frequency of the satellite transmitter also varies with respecttothereceiverontheEarthstationterminal.Ifthefrequencytransmittedbythesatellite is f_T, then the received frequency f_R is given by

$$\mathbf{(f_R - f_T / f_T) = (\Delta f / f_T) = (V_T \; V_P).}$$

Where,

V_T is the component of the satellite transmitter velocity vector directed towards the Earth station receiver vP is the phase velocity of light in free space (3×108 m/s).

1.8.2 Variation in the Orbital Distance

Variation in the orbital distance results in variation in the range between the satellite and the Earth station terminal. If a Time Division Multiple Access (TDMA) scheme is employed by the satellite, the timing of the frames within the TDMA bursts should be worked out carefully so that the user terminals receive the correct data at the correct time. Range variations are more predominant in low and medium Earth orbiting satellites as compared to the geostationary satellites.

1.8.3 Solar Eclipse

There are times when the satellites do not receive solar radiation due to obstruction from a celestialbody.Duringtheseperiodsthesatellitesoperateusingonboardbatteries.Thedesignof the battery is such so as to provide continuous power during the period of the eclipse. Ground control stations perform battery conditioning routines prior to the occurrence of an eclipse to ensure best performance during the eclipse. These include discharging the batteries close to their maximum depth of discharge and then fully recharging them just before the eclipse occurs. Also, the rapidity with which the satellite enters and exits the shadow of the celestial body creates sudden temperature stress situations.

1.8.4 Sun Transit Outrage

There are times when the satellite passes directly between the sun and the Earth as shown in Fig.1.26 The Earth station antenna will receive signals from the satellite as well as the microwave radiation emitted by the sun (the sun is a source of radiation with an equivalent temperature varying between 6000K to 11000K depending upon the time of the 11-year sunspot cycle).

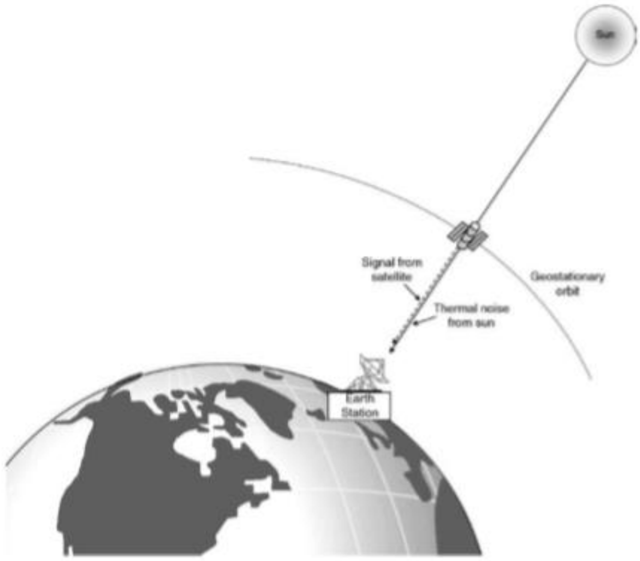


Fig.1.26 Sun outrage conditions

This might cause temporary outrage if the magnitude of the solar radiation exceeds the fade margin of the receiver. The traffic of the satellite may be shifted to other satellites during such periods.

1.9 Eclipses

With reference to satellites, an eclipse is said to occur when the sunlight fails to reach the satellite'ssolarpanelduetoanobstructionfromacelestialbody.Themajorandmostfrequent sourceofaneclipseisduetothesatellitecomingintheshadowoftheEarth(Fig.1.27).Thisis knownasasolareclipse.Theeclipseistotal;i.e.thesatellitefailstoreceiveanylight whatsoever ifitpassesthroughtheumbra,whichisthedarkcentralregionoftheshadow,andreceivesvery littlelightifitpassesthroughthepenumbra,whichisthelessdarkregionsurroundingtheumbra (Fig.1.28).TheeclipseoccursastheEarth'sequatorialplaneisinclinedataconstantangleof about23.5◦ toitseclipticplane,whichistheplaneoftheEarth'sorbitextendedtoinfinity.The eclipseisseenon42nightsduringthespringandanequalnumberofnightsduringtheautumn bythegeostationarysatellite.Theeffectistheworstduringtheequinoxesandlastsforabout72 minutes.Theequinox,asexplainedearlier,isthepointintimewhenthesuncrossestheequator, makingthedayandnightequalinlength.The spring andautumnequinoxesrespectivelyoccur on 20–21 March and 22–23 September. During the equinoxes in March and September, the satellite,theEarthandthesunarealignedatmidnightlocaltimeandthe satellite spendsabout 72minutesintotaldarkness.From21daysbeforeand21daysaftertheequinoxes,thesatellite crosses the umbral cone each day for some time, thereby receiving only a part of solar light for that time. During the rest of the year, the geostationary satellite orbit passes either above or below the umbral cone. It is at the maximum distance at the time of the solstices, above the umbral cone at the time of the summer solstice (20–21 June) and below it at the time of the winter solstice (21–22 December). Fig.1.29 further illustrates the phenomenon

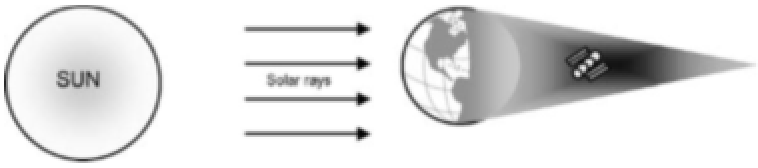


Fig.1.27 Solar eclipse

Fig.1.28 Umbra and penumbra

Hence, the duration of an eclipse increases from zero to about 72 minutes starting 21 days before the equinox and then decreases from 72 minutes to zero during 21 days following the equinox. The duration of an eclipse on a given day around the equinox can be seen from the graph in Fig.1.30. Another type of eclipse known as the lunar eclipse occurs whenthe moon’s shadow passes across the satellite (Fig.1.31). This is much less common and occurs once in 29 years. In fact, for all practical purposes, when an eclipse is mentioned with respect to satellites, it is a solar eclipse that is referred to.

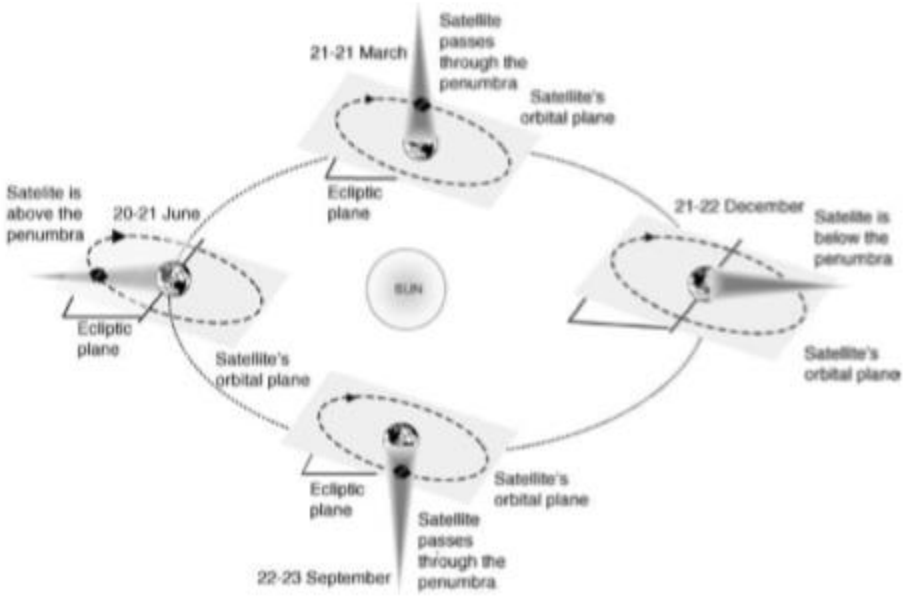


Fig.1.29 Positions of the geostationary satellite during the equinoxes and solstices

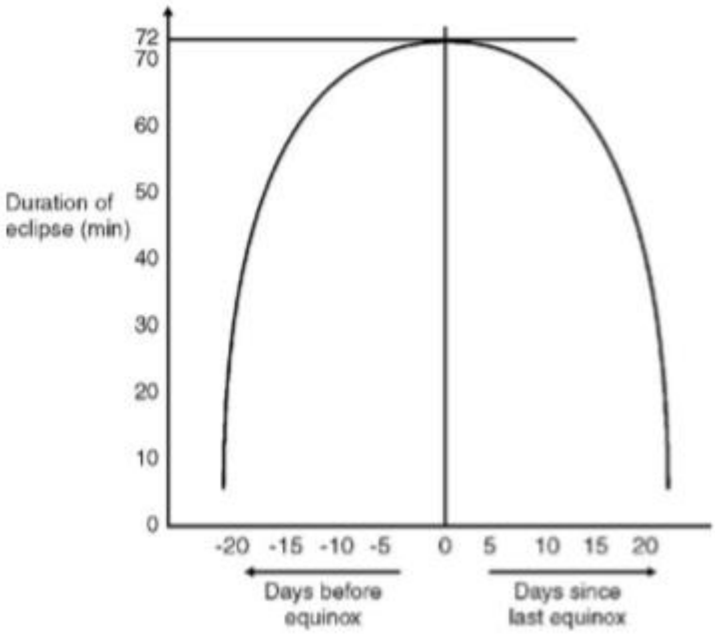


Fig.1.30 Duration of the eclipse before and after the equinox

While a solar eclipse takes place, the failure of sunlight to reach the satellite interrupts the battery recharging process. The satellite is depleted of its electrical power capacity.

It does not significantly affect low power satellites, which can usually continue their operation with back-up power. The high power satellites, however, shut down for all but essential services.

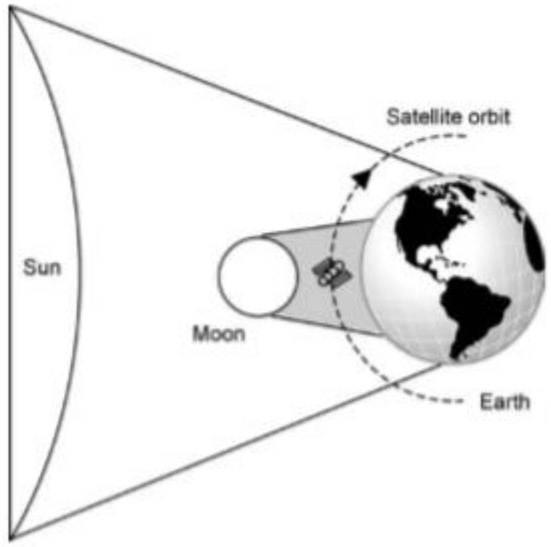


Fig.1.31 Lunar eclipse

1.10 Look angles

The look angles of a satellite refer to the coordinates to which an Earth station must be pointed in order to communicate with the satellite and are expressed in terms of azimuth and elevation angles.

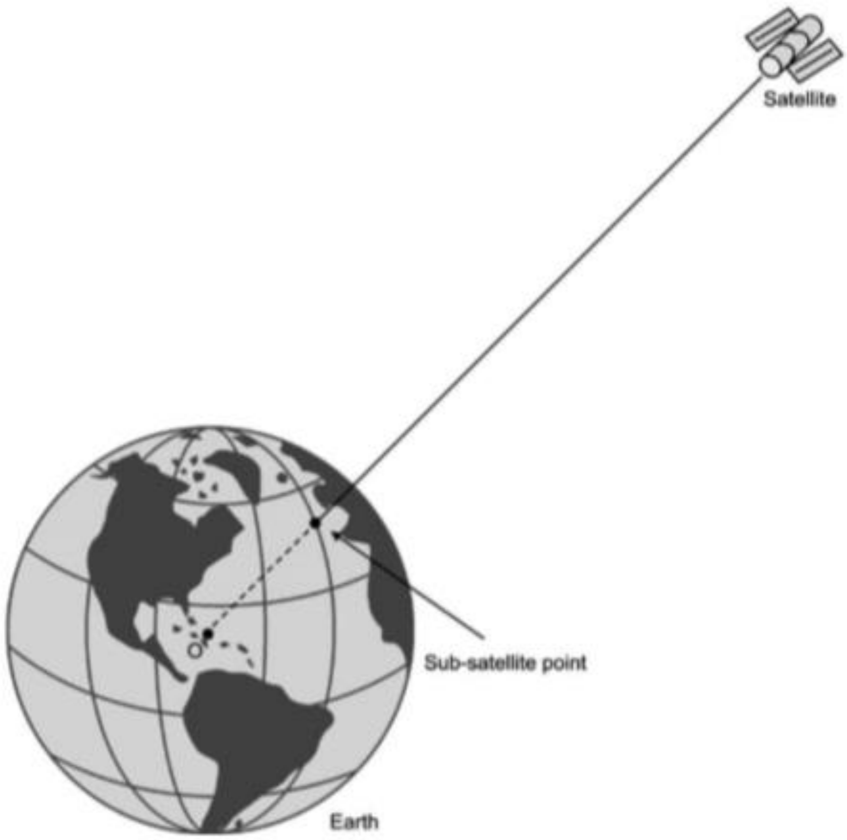


Fig.1.32 Sub-satellite point

In the case where an Earth station is within the footprint or coverage area of a geostationary satellite, it can communicate with the satellite by simply pointing its antenna towards it. The process of pointing the Earth station antenna accurately towards the satellite can be accomplished if the azimuth and elevation angles of the Earth station location are known. Also, the elevation angle, as we shall see in the following paragraphs, affects the slant range, i.e. line of sight distance between the Earth station and the satellite. In order to determine the look angles of a satellite, its precise location should be known.

The location of a satellite is very often determined by the position of the sub-satellite point. The sub-satellite point is the location on the surface of the Earth that lies directly between the satellite and the centre of the Earth. To an observer on the sub-satellite point, the satellite will appear to be directly overhead (Fig.1.32).

1.10.1 Azimuth angle

The azimuth angle A of an Earth station is defined as the angle produced by the line of intersection of the local horizontal plane and the plane passing through the Earth station, the satellite and the centre of the Earth with the true north. We can visualize that this line of intersection between the two above-mentioned planes would be one of the many possible tangents that can be drawn at the point of location of the Earth station. Depending upon the location of the Earth station and the sub-satellite point.

1.10.2 Elevation angle

The Earth station elevation angle E is the angle between the line of intersection of the local horizontal plane and the plane passing through the Earth station, the satellite and the centre of the Earth with the line joining the Earth station and the satellite.