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SATELLITE COMMUNICATION-BEC515D

Prepared By

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COURSE LEARNING OUTCOMES(CLO)

Course objectives: This course will enable students to:

CLO1: Understand the basic principle of satellite orbits and trajectories.

CLO2: Study of electronic systems associated with a satellite and the earth station.

CLO3: Understand the various technologies associated with the satellite communication.

CLO4: Focus on a communication satellite and the national satellite system.

CLO5: Study of satellite applications focusing various domains services such as remote sensing, weather forecasting and navigation

COURSE OUTCOMES (CO)

CO1: Describe the satellite orbits and its trajectories with the definition of parameters associated with it	L2
CO2: Describe the electronic hardware systems associated with the satellite subsystem and earth station	L2
CO3: Describe the communication satellites with the focus on national satellite system	L3
CO4: Compute the satellite link parameters under various propagation conditions with the illustration of multiple access techniques	L3

SYLLABUS

Module:1:-Satellite Orbits and Trajectories

- Definition
- Basic Principles
- Orbital parameters
- Injection velocity and satellite trajectory
- Types of Satellite orbits
- Orbital perturbations
- Satellite stabilization
- Orbital effects on satellite's performance
- Eclipses
- Look angles: Azimuth angle, Elevation angle

DEFINITION

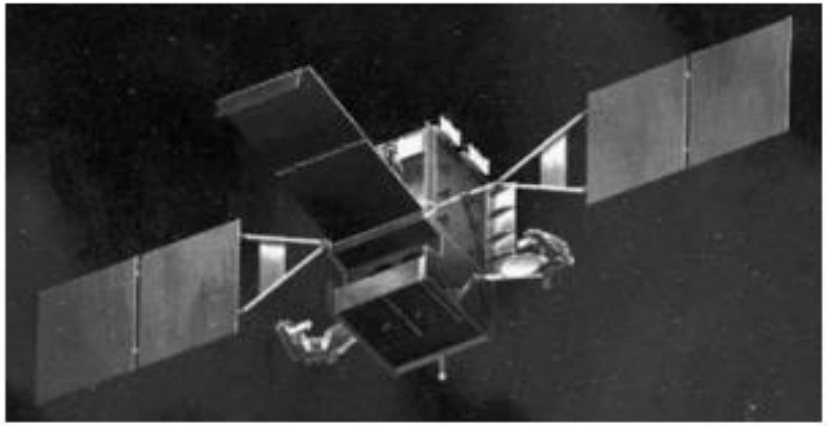
- A satellite in general is any **natural or artificial body** moving around a celestial body such as planets and stars.
- Here the **reference** is made only to **artificial satellites** orbiting the planet Earth.
- These satellites are put into the **desired orbit** and have **payloads** depending upon the **intended application.**

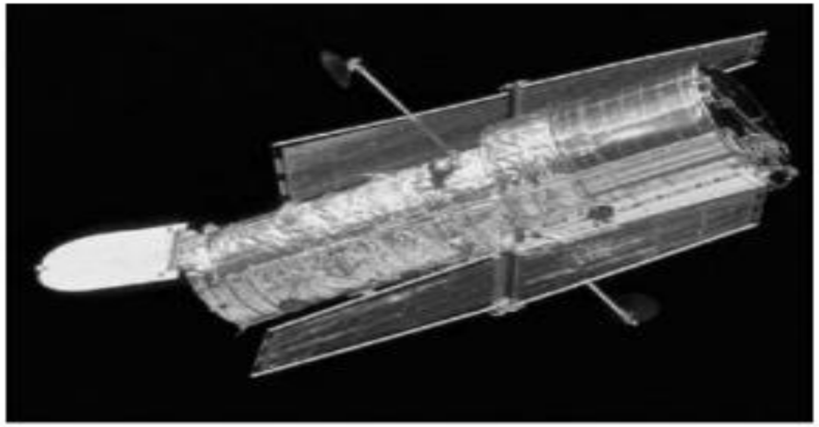
A **communication satellite** as shown in the figure is a kind of **repeater station** that receives signals from ground, processes them and then retransmits them back to Earth.

An **Earth observation satellite** as shown in the figure is a photographer that takes pictures of **regions of interest** during **its periodic motion.**

A weather forecasting satellite as shown in the figure takes photographs of clouds and monitors other **atmospheric parameters,** thus assisting the weatherman in making timely and accurate forecasts

A satellite could effectively do the job of a spy in the case of some **military satellites** (Figure 1.1) meant for the purpose or of an explorer when suitably equipped and launched for **astrophysical applications** (Figure 1.2).



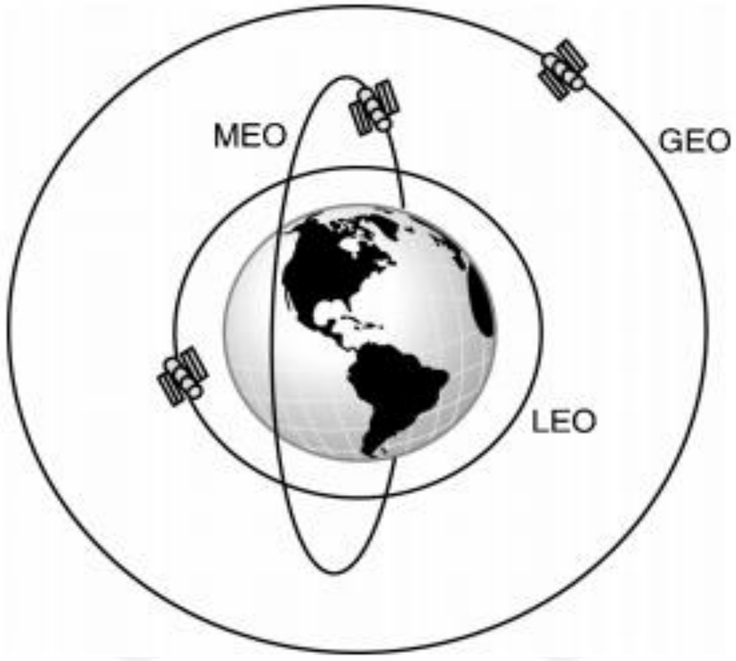


Definition of an Orbit and a Trajectory

- 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.**

Definition of an Orbit and a Trajectory

- The **motion of different planets** of the solar system around the sun
- And the **motion of artificial satellites** around Earth are examples of orbital motion.



Example of orbital motion – satellites revolving around Earth

- 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 (Figure 2.2) or the path followed by orbiting satellites when they move from an intermediate orbit to their final destined orbit (Figure 2.3) are examples of trajectories.

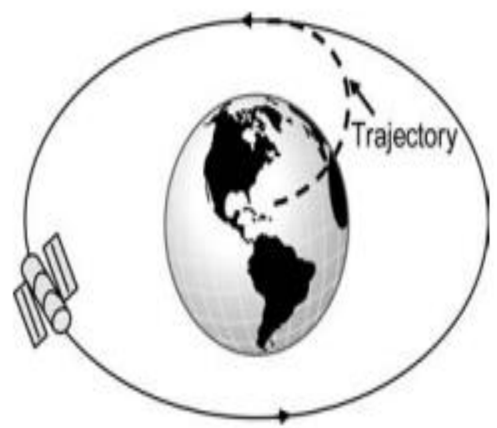


Figure 2.2 Example of trajectory – path followed by a rocket on its way during satellite launch

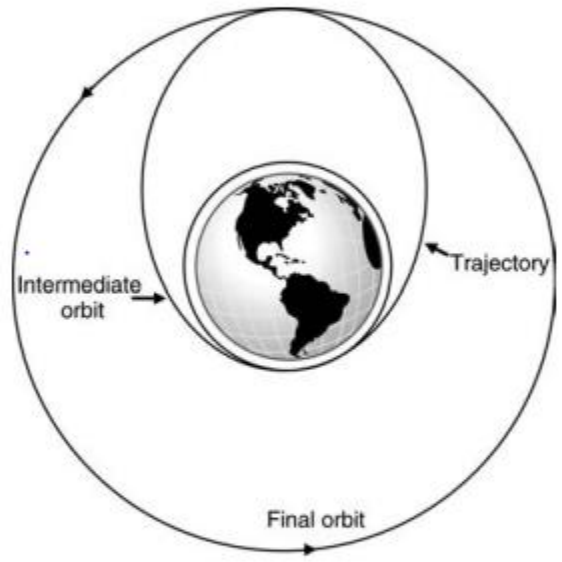


Figure 2.3 Example of trajectory – motion of a satellite from the intermediate orbit to the final orbit

Orbiting Satellites – Basic Principles

The motion of **natural** and **artificial satellites** around Earth is **governed by two forces**.

- **centripetal force** directed **towards** the center of the Earth due to the **gravitational force** of attraction of Earth
- **centrifugal force** that acts **outwards** from the center of the Earth

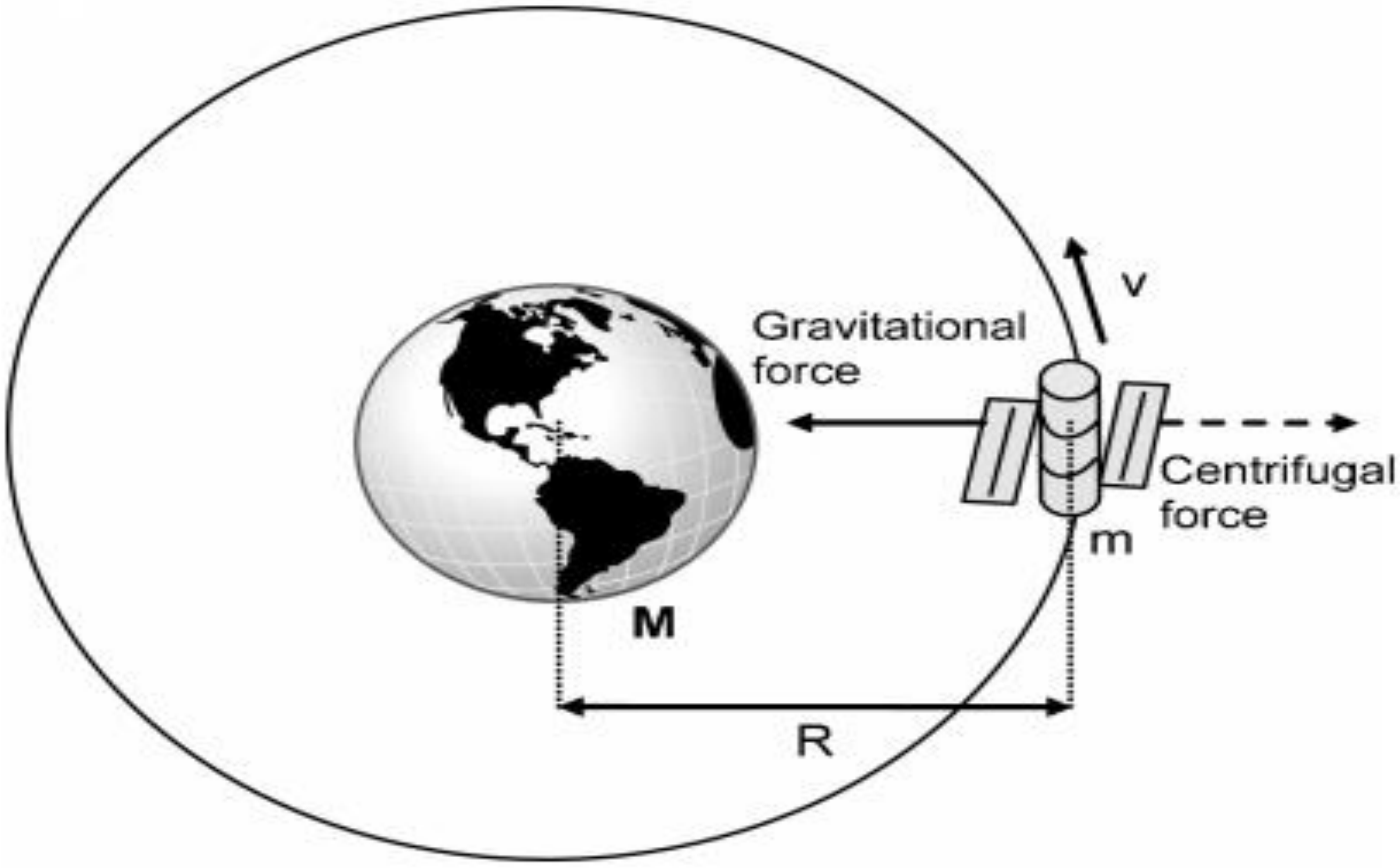


Fig: Gravitational force and the centrifugal force acting on bodies orbiting Earth

- 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 two forces can be explained from **Newton's law of gravitation** and **Newton's second law of motion.**

Newton’s Law of Gravitation

- Newton’s law of gravitation, every particle irrespective of its **mass** attracts every other particle with a **gravitational force** whose **magnitude** is **directly proportional** to the **product of the masses of** the two particles and **inversely proportional** to the square of the distance between them.

$$F = \frac{G m_1 m_2}{r^2}$$

Where,

$m1$, $m2$ = masses of the two particles
r = distance between the two particles
G = gravitational constant = 6.67 × 10-11 m3/kg s2

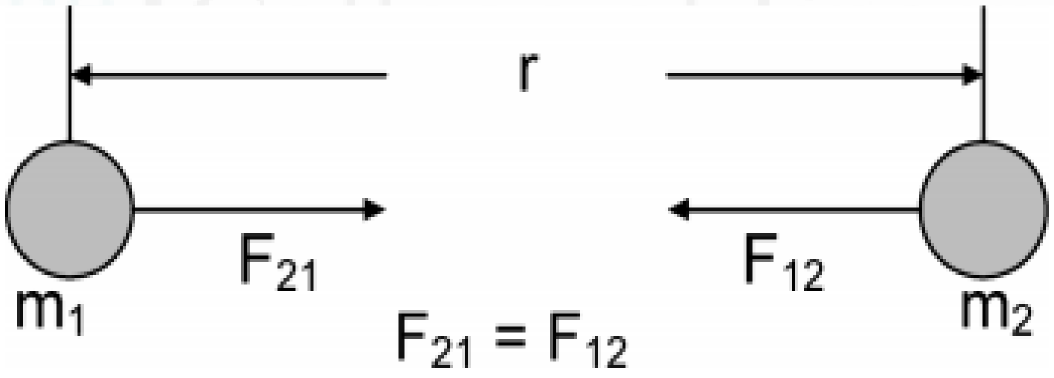


Fig: Newton’s law of gravitation

Newton's Second Law of Motion

- In the case of a satellite orbiting Earth, if the **orbiting velocity** is *v*, then the acceleration, called **centripetal acceleration**, experienced by the **satellite** at a distance *r* from the centre of the Earth would be

$$v^2/r$$

- If the mass of satellite is *m*, it would experience a reaction force of

$$mv^2/r$$

•Expression for the orbital velocity v as follows:

$$\frac{Gm_1m_2}{r^2} = \frac{m_2v^2}{r}$$

$$v = \sqrt{\left(\frac{Gm_1}{r}\right)} = \sqrt{\left(\frac{\mu}{r}\right)}$$

Where,

$m1$ = mass of Earth
$m2$ = mass of the satellite
$\mu = Gm1$ = 3.986 013 × 105 km3/s2 = 3.986 013 × 1014 N m2/kg

The orbital period in such a case can be computed from

$$T = \frac{2\pi r^{3/2}}{\sqrt{\mu}}$$

- In the case of an **elliptical orbit**, the forces governing the motion of the satellite are the same.
- The velocity at any point on an elliptical orbit at a distance *d* from the centre of the Earth is given by the formula.

$$v = \sqrt{\left[\mu\left(\frac{2}{d} - \frac{1}{a}\right)\right]}$$

where

a= semi-major axis of the elliptical orbit

- The **orbital period in the case of an elliptical orbit** is given by

$$T = \frac{2\pi a^{3/2}}{\sqrt{\mu}}$$

The **movement of a satellite** in an orbit is governed by **three Kepler's laws**, explained below.

Kepler's First Law

- The **orbit** of a satellite around Earth is **elliptical** with the centre of the **Earth** lying at one of the **foci of the ellipse**.
- The elliptical orbit is characterized by its **semi-major axis a** and **eccentricity e.**
- Eccentricity *(e)* is the ratio of the distance between the centre of the ellipse and either of its foci (= *ae*) to the semi-major axis of the ellipse *a*.

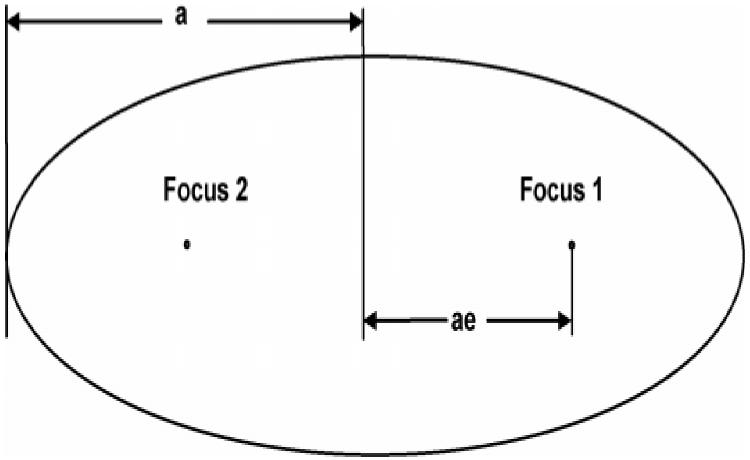


Fig: Kepler's first law

A **circular orbit** is a special case of an elliptical orbit where the **foci merge** together to give a **single central point** and the **eccentricity becomes zero**.

Other important parameters of an **elliptical satellite orbit include,**

- Apogee (farthest point of the orbit from the Earth's Centre)
- Perigee (nearest point of the orbit from the Earth's Centre) distances.

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- The law of conservation of energy is valid at all points on the orbit.

- In the context of satellites, it means that the sum of the kinetic and the potential energy of a satellite always remain constant. **The value of this constant is equal to -*Gm*1*m*2/(2*a*), where**

$m1$ = mass of Earth
$m2$ = mass of the satellite
a = semi-major axis of the orbit

$$\text{Kinetic energy} = \frac{1}{2}(m_2 v^2)$$

$$\text{Potential energy} = -\frac{G m_1 m_2}{r}$$

$$\frac{1}{2}(m_2 v^2) - \frac{G m_1 m_2}{r} = -\frac{G m_1 m_2}{2a}$$

$$v^2 = G m_1 \left(\frac{2}{r} - \frac{1}{a}\right)$$

$$v = \sqrt{\left[\mu\left(\frac{2}{r} - \frac{1}{a}\right)\right]}$$

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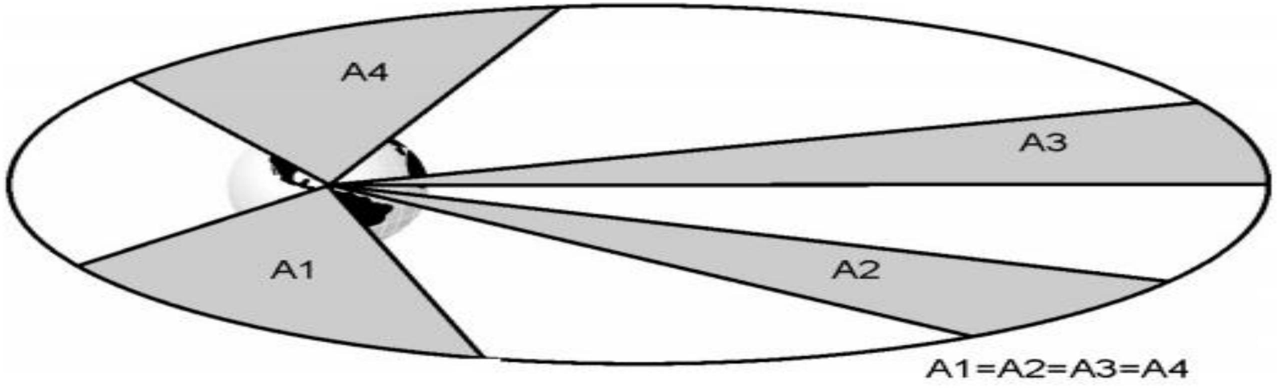
Kepler's Second Law

The line joining the satellite and the centre of the Earth sweeps out equal areas in the plane of the orbit in equal time intervals ie the rate (dA/dt) at which it sweeps area *A* is constant.

The rate of change of the swept-out area is given by

$$\frac{dA}{dt} = \frac{\text{angular momentum of the satellite}}{2m}$$



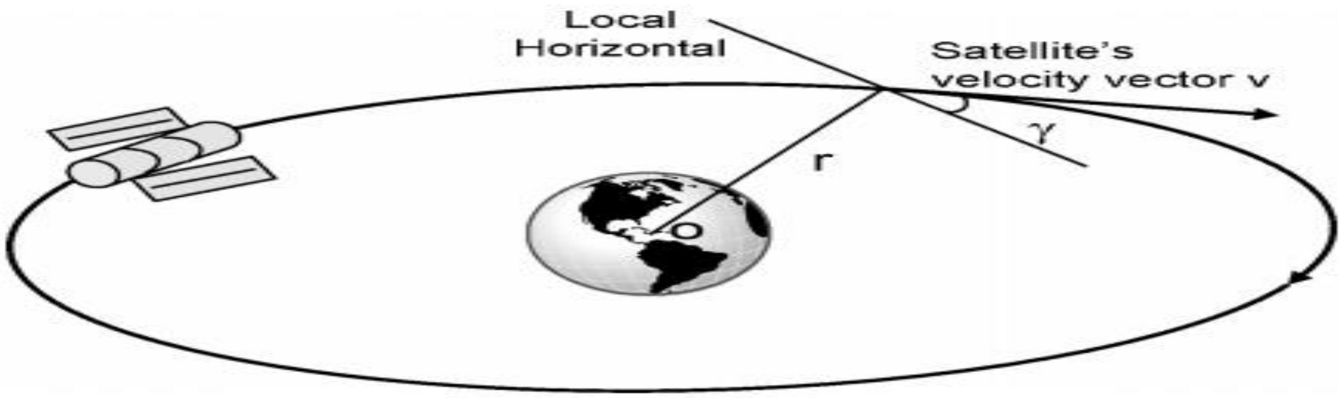


Fig: Satellite's position at any given time

Kepler's second law is also equivalent to the law of conservation of momentum, which implies that the angular momentum of the orbiting satellite given by the product of the radius vector and the component of linear momentum perpendicular to the radius vector is constant at all points on the orbit.

$$v_p r_p = v_a r_a = v r \cos \gamma$$

where

v_p = velocity at the perigee point

r_p = perigee distance

v_a = velocity at the apogee point

r_a = apogee distance

v = satellite velocity at any point in the orbit

r = distance of the point

γ = angle between the direction of motion of the satellite and the local horizontal

Kepler's Third Law

- The square of the time period of any satellite is proportional to the cube of the semi-major axis of its elliptical orbit.
- A circular orbit with radius *r* is assumed.
- A circular orbit is only a special case of an elliptical orbit with both the semi-major axis and semi-minor axis equal to the radius.

Equating the gravitational force with the centrifugal force gives

$$\frac{Gm_1m_2}{r^2} = \frac{m_2v^2}{r}$$

Replacing v by ωr in the above equation gives

$$\frac{Gm_1m_2}{r^2} = \frac{m_2\omega^2r^2}{r} = m_2\omega^2r$$

which gives $\omega^2 = Gm_1/r^3$. Substituting $\omega = 2\pi/T$ gives

$$T^2 = \left(\frac{4\pi^2}{Gm_1}\right)r^3$$

This can also be written as

$$T = \left(\frac{2\pi}{\sqrt{\mu}}\right)r^{3/2}$$

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The above equation holds good for elliptical orbits provided r is replaced by the semi-major axis a. This gives the expression for the time period of an elliptical orbit as

$$T = \left(\frac{2\pi}{\sqrt{\mu}} \right) a^{3/2}$$

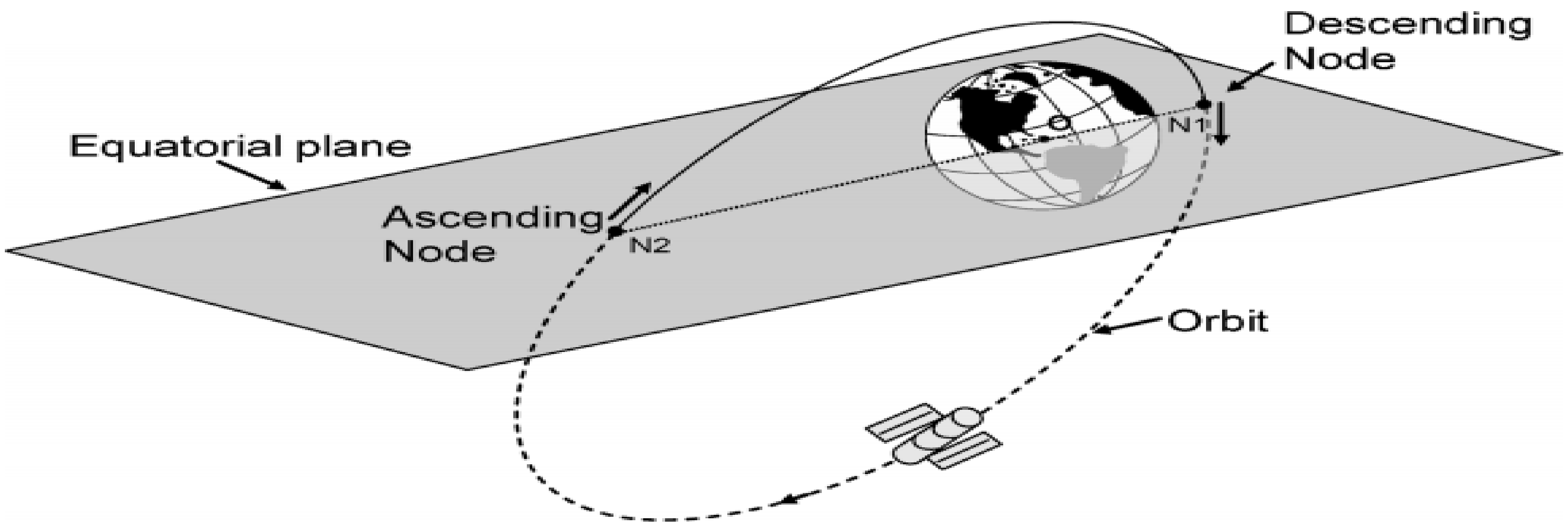
ORBITAL PARAMETERS

- Ascending and descending nodes
- Equinoxes
- Solstices
- Apogee
- Perigee
- Eccentricity
- Semi-major axis
- Inclination
- Argument of the perigee
- True anomaly of the satellite
- Angles defining the direction of the satellite

1. Ascending and Descending nodes

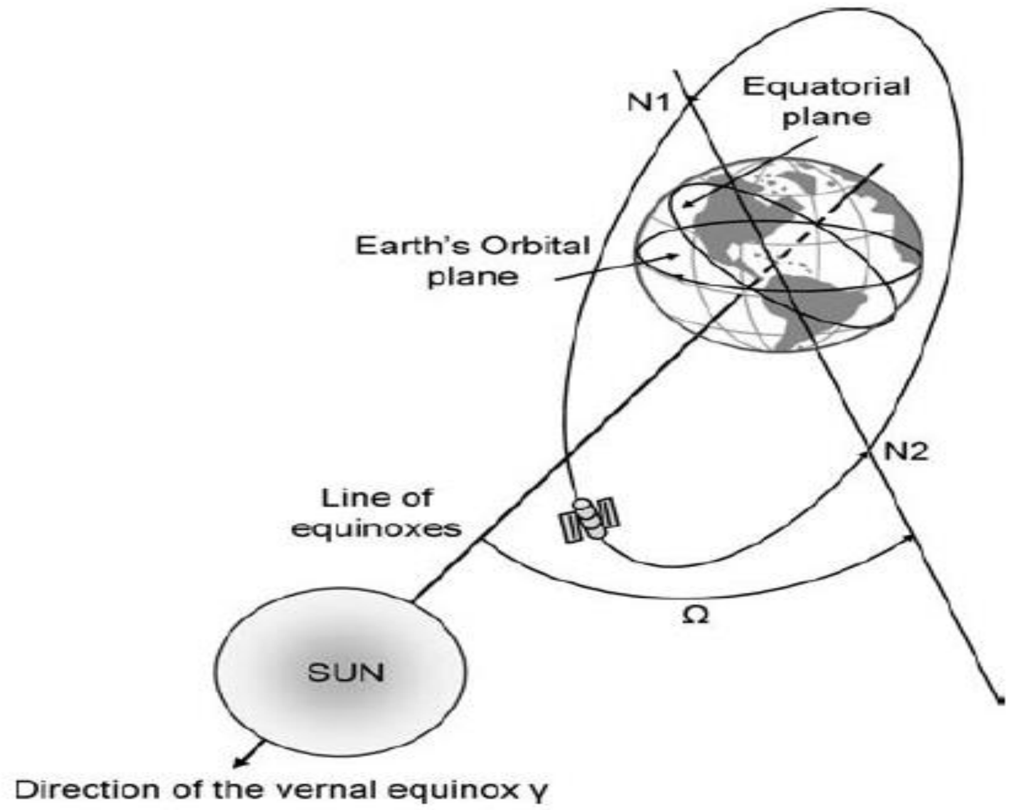
The satellite orbit cuts the equatorial plane at two points:

1. **Descending node (N1),** where the satellite passes from the **northern hemisphere** to the **southern hemisphere.**
2. **Ascending node (N2),** where the satellite passes from the **southern hemisphere** to the **northern hemisphere**.



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 center of the Earth and the sun with the Earth's equatorial plane follows a sinusoidal variation and completes one cycle of sinusoidal variation over a period of 365 days.



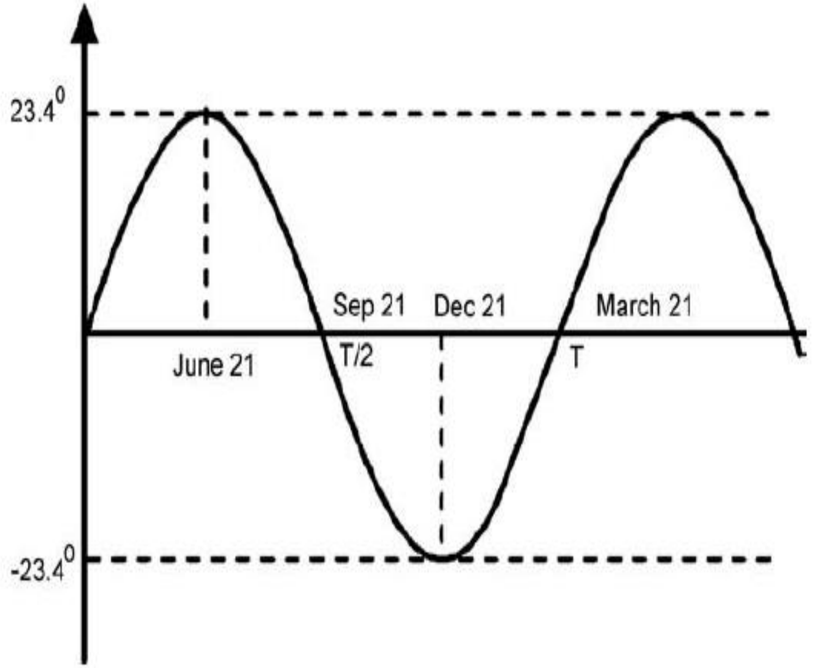
2. Equinoxes cont

The sinusoidal variation of the angle of inclination is defined by

$$\text{Inclination angle (in degrees)} = 23.4 \ \sin\left(\frac{2\pi t}{T}\right)$$

where T is 365 days.

- This expression indicates that the **inclination angle is zero** for $t = T/2$ 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.**



3. Solstices (Sun to stand still)

- Solstices are the times when the inclination angle is at its maximum, i.e. 23.4 ∘ .

These also occur twice during a year on

- 20-21 June, called the **summer solstice**, and
- 21-22 December called the **winter Solstice**

4. Apogee

Apogee is the point on the **satellite orbit** that is at the **farthest distance** from the **centre of the Earth.**

Apogee distance , A = a (1 + e)

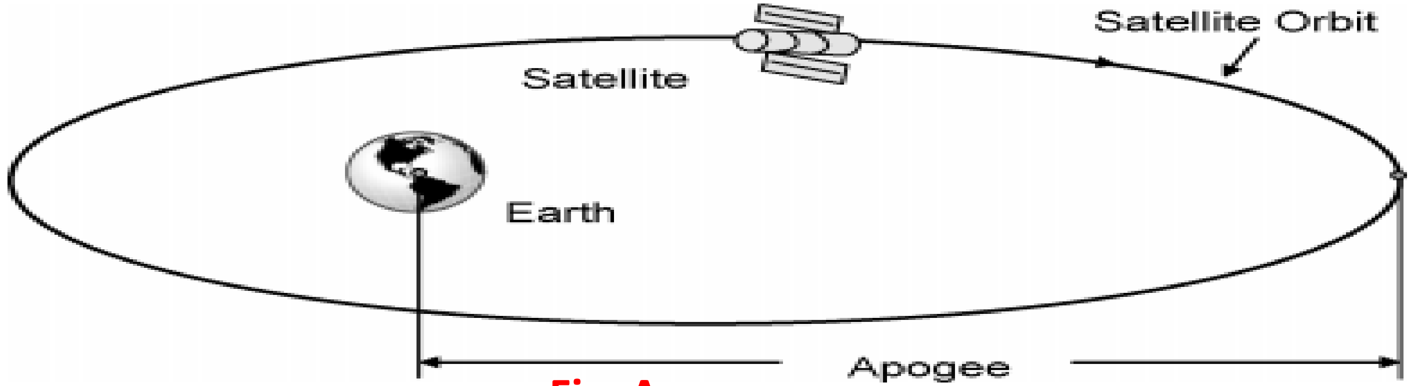


Fig: Apogee

5. Perigee

Perigee is the **point on the orbit** that is **nearest** to the **centre of the Earth .**

Perigee distance, $P = a\,(1 - e)$

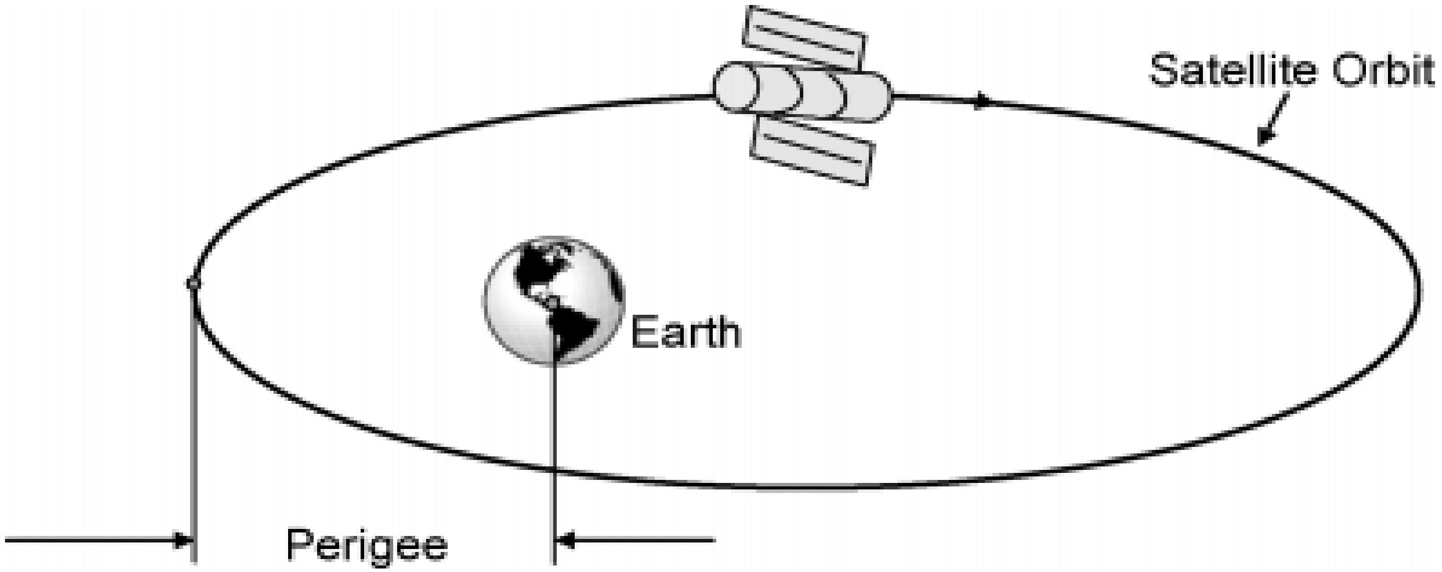


Fig: Perigee

6. Eccentricity

The orbit **eccentricity *e*** is the ratio of the distance between the **centre of the ellipse** and the **centre of the Earth** to the semi-major axis of the ellipse. It can be computed from any of the following expressions:

$$e = \frac{\text{apogee} - \text{perigee}}{\text{apogee} + \text{perigee}}$$

$$e = \frac{\text{apogee} - \text{perigee}}{2a}$$

$$e = \sqrt{(a^2 - b^2)}/a$$

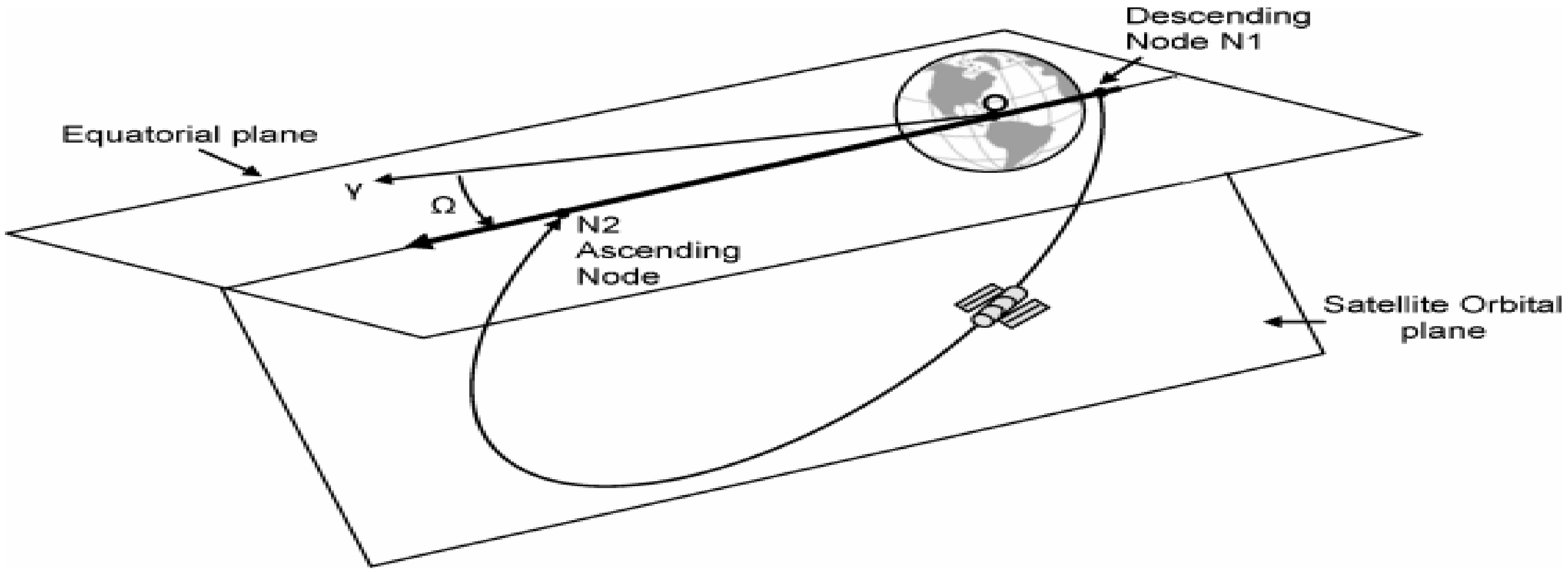
7. Semi-major axis

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

$$a = \frac{\text{apogee} + \text{perigee}}{2}$$

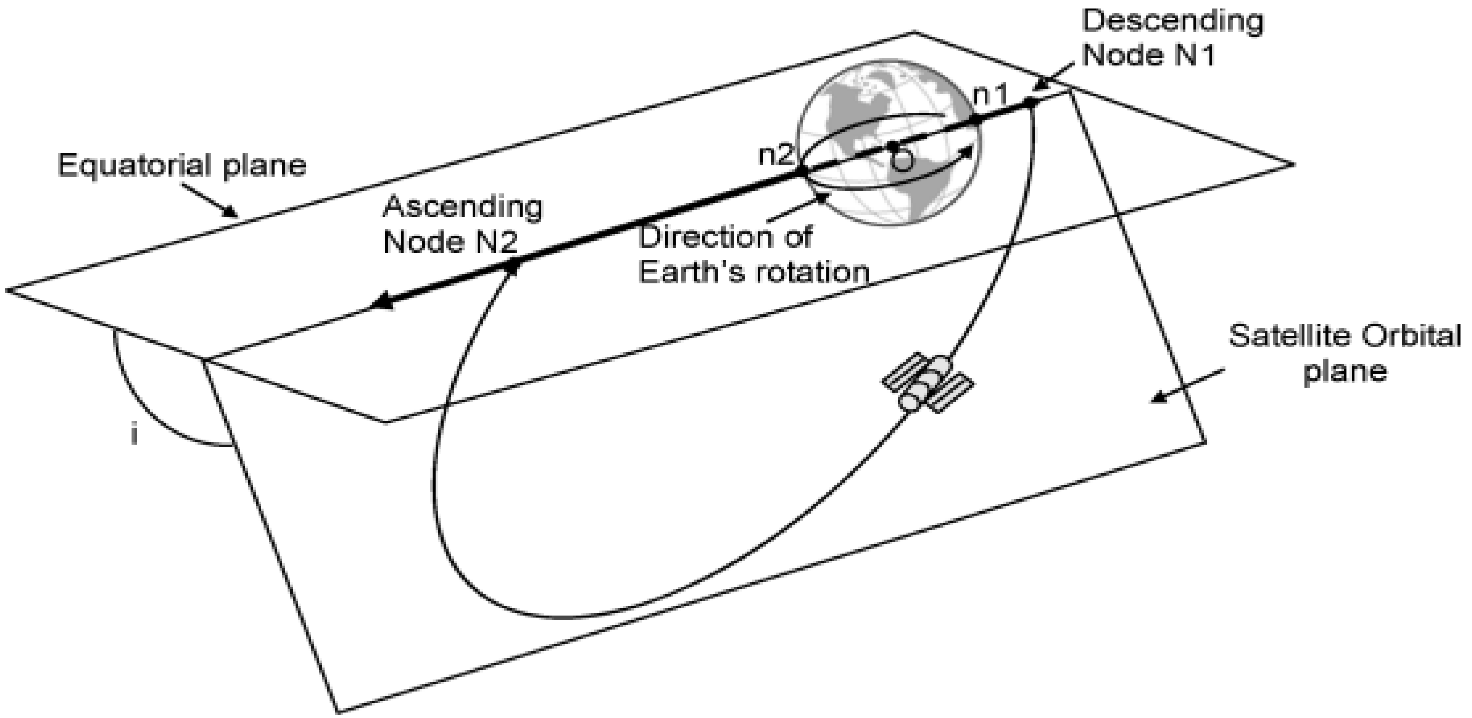
8. Right ascension of the ascending node

It describes 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 Ω** measured from the vernal equinox towards the line of nodes in the direction of rotation of Earth.



9. Inclination

Inclination is the angle that the **orbital plane of the satellite** makes with the **Earths's equatorial plane.**



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

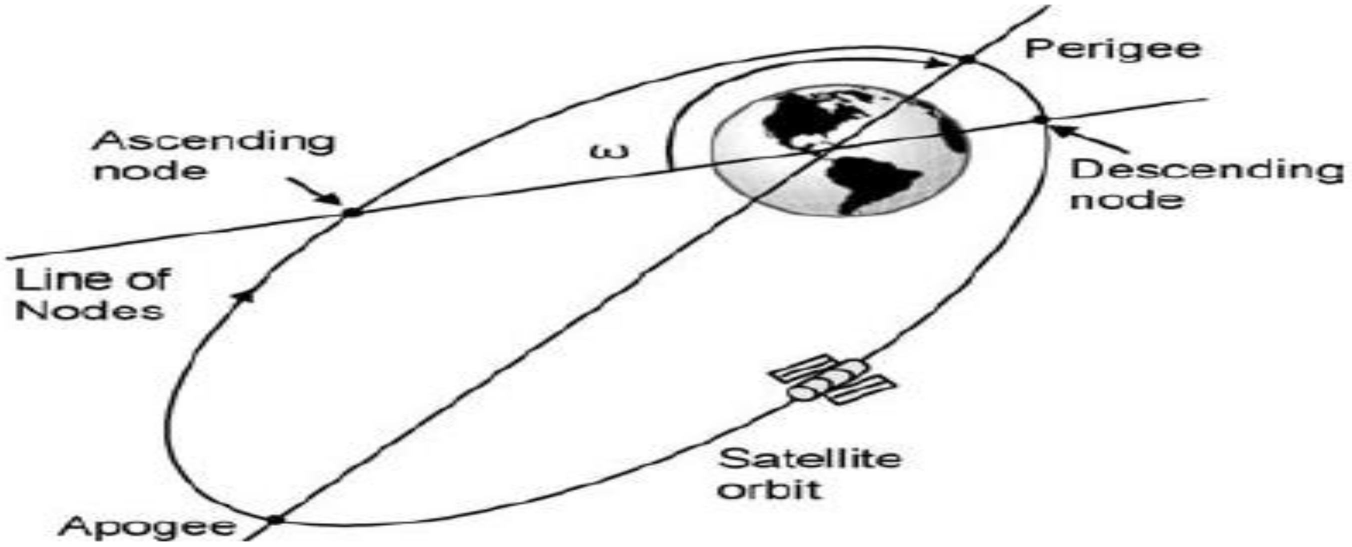
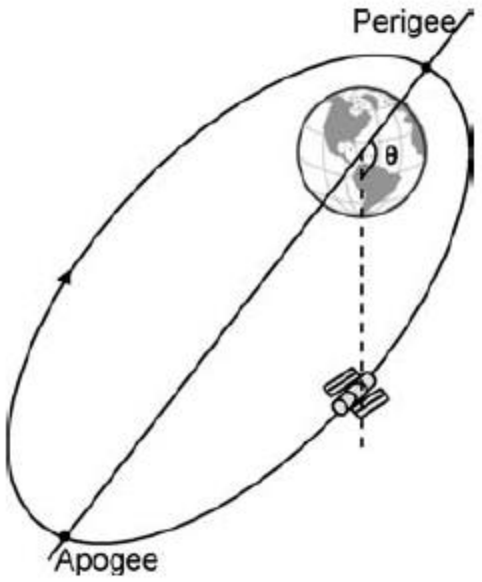


Figure 2.17 Argument of perigee

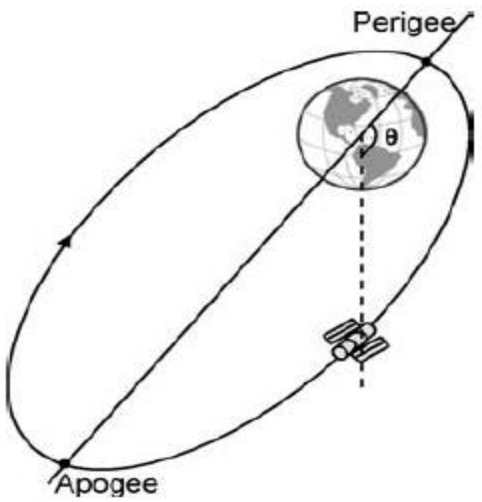
11. True anomaly of the satellite.

- 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 center of the Earth with the line joining the satellite and the center of the Earth



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.



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 the satellite 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

$$V_P = \sqrt{\left[\left(\frac{2\mu}{r}\right) - \left(\frac{2\mu}{R+r}\right)\right]}$$

where ,

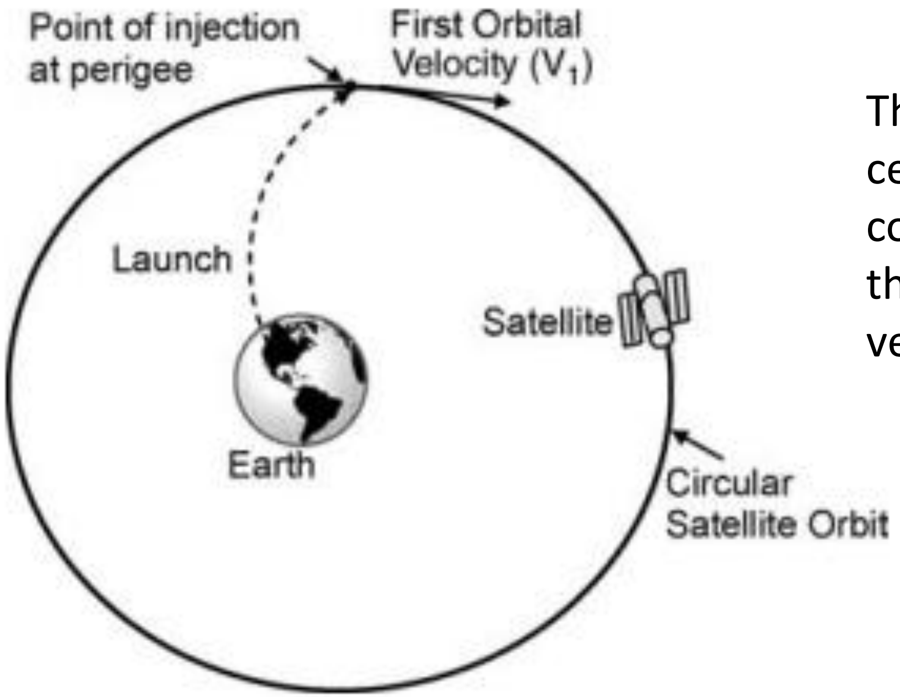
R = apogee distance

r = perigee distance

μ = GM = constant

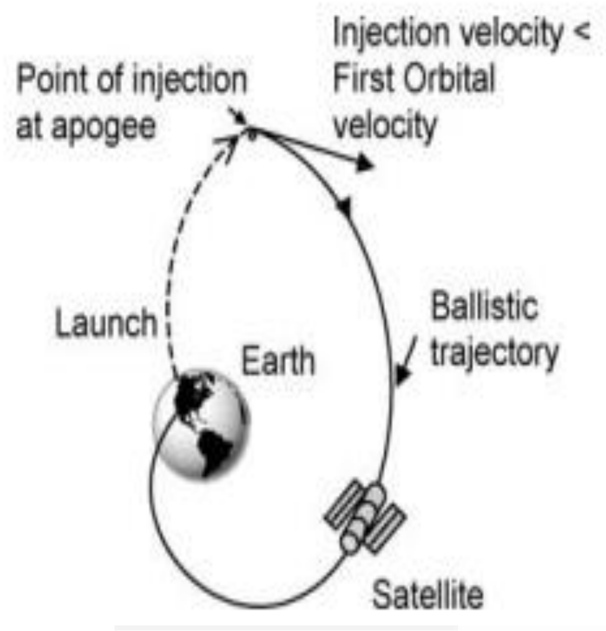
The **first cosmic velocity V1** is the one at which **apogee and perigee distances are equal**, i.e. R = r, and the orbit is circular. The above expression then reduces to

$$V_1 = \sqrt{\left(\frac{\mu}{r}\right)}$$



Thus, irrespective of the distance r of the satellite from the centre of the Earth, if the injection velocity is equal to the first cosmic velocity, also sometimes called the **first orbital velocity**, the satellite **follows a circular orbit** and moves with a uniform velocity equal to √(μ/r).

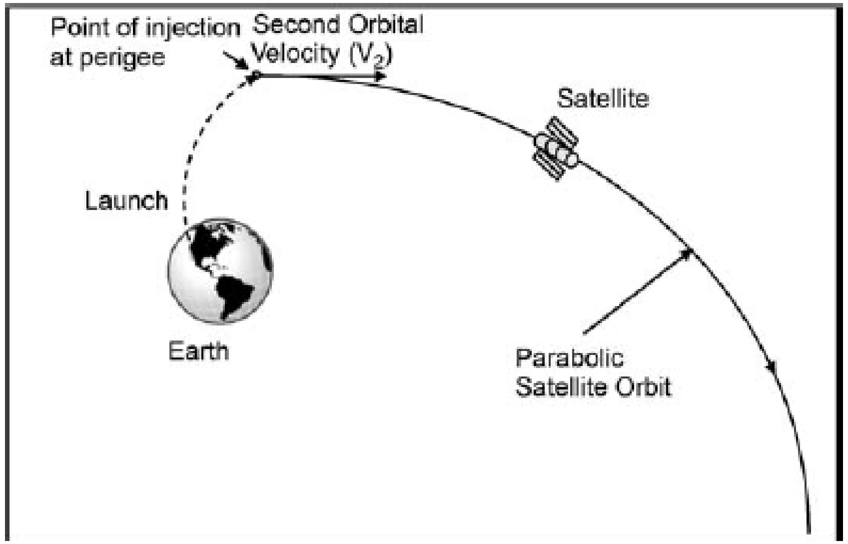
Satellite's path where the injection velocity is equal to the first orbital velocity



Satellite's path where the injection velocity is less than the first orbital velocity

- If the **injection velocity** happens to **be less than the first cosmic velocity,** the satellite follows a ballistic trajectory and falls back to Earth.
- In fact, in this case, the orbit is **elliptical** and the **injection point is at the apogee** and not the perigee.
- If the perigee lies in the atmosphere or exists only virtually below the surface of the Earth, the satellite accomplishes a ballistic flight and falls back to Earth

- For **injection velocity greater** than the **first cosmic velocity** and **less than the second cosmic velocity**, i.e. $V > \sqrt{(\mu/r)}$ and $V < \sqrt{(2\mu/r)}$, the orbit is **elliptical and eccentric**.
- The orbit **eccentricity is** between 0 and 1.
- The injection point in this case is the perigee and the apogee distance attained in the resultant elliptical orbit depends upon the injection velocity.
- The higher the injection velocity, the greater is the apogee distance.



Satellite's path where the injection velocity is equal to the second orbital velocity

- When the **injection velocity equals √(2*μ*/*r*)**, the apogee distance *R* becomes infinite and the orbit takes the shape of a parabola (Figure 2.29) and the orbit eccentricity is 1.
- This is the **second cosmic velocity *v*2.** At this velocity, the satellite escapes Earth's gravitational pull.
- For an injection velocity greater than the second cosmic velocity, the trajectory is hyperbolic within the solar system and the orbit eccentricity is greater than 1.

- If the injection velocity is increased further, a stage is reached where the **satellite succeeds** in **escaping from the solar system.**
- This is known as the **third cosmic velocity** and is related to the motion of planet Earth around the sun.
- The third cosmic velocity ($V3$) is mathematically expressed as

$$V_3 = \sqrt{\left[\frac{2\mu}{r} - V_t^2(3 - 2\sqrt{2})\right]}$$

*V*t is the speed of Earth's revolution around the sun.

- Generalized expression for the velocity of the satellite in elliptical orbits according to which *V*p = velocity at the perigee point

$$Vp = \sqrt{\left[(2\mu)\left(\frac{1}{r} - \frac{1}{R+r}\right)\right]}$$

For a given **perigee distance *r*,** it can be proved that the injection velocities and corresponding apogee distances are related by

$$\left(\frac{v_2}{v_1}\right)^2 = \frac{1 + r/R_1}{1 + r/R_2}$$

Types of Satellite Orbits

The satellite orbits can be classified on the basis of:

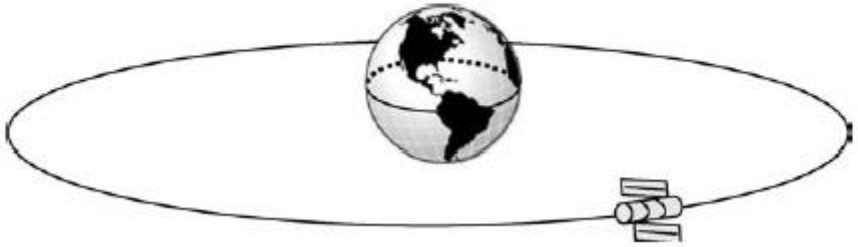
1. Orientation of the orbital plane
2. Eccentricity
3. Distance from Earth

Orientation of the Orbital Plane

- The **orbital plane** of the satellite can have **various orientations** with respect to the **equatorial plane of Earth.**
- The **angle** between the **two planes** is called the **angle of inclination of the satellite.**
- On this basis, the orbits can be classified as **equatorial orbits, polar orbits and inclined orbits.**

Equatorial 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 (Figure 2.32).
- A satellite in the equatorial orbit has a latitude of 0°.



For an angle of inclination equal to 90◦, the satellite is said to be in the **polar orbit** (Figure 2.33).

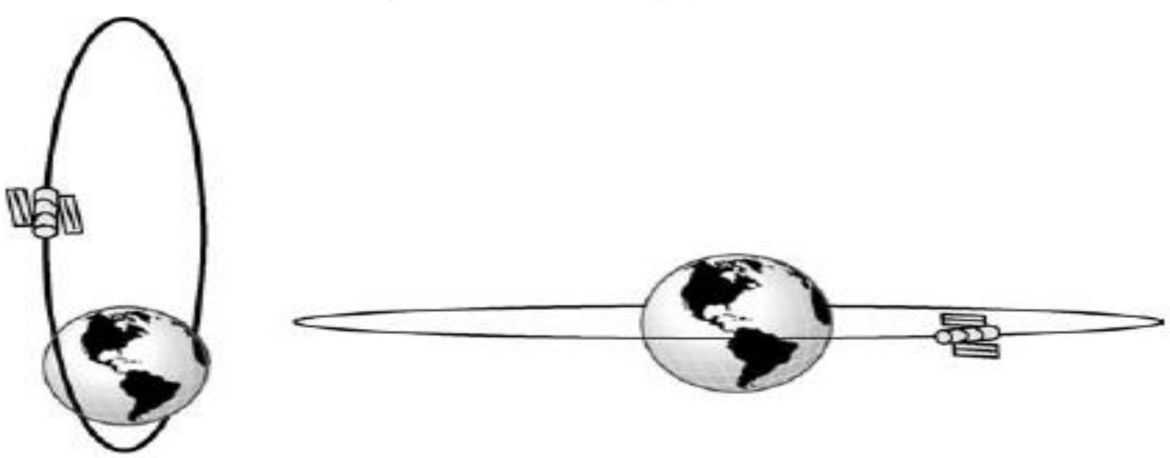
For an angle of inclination between 0◦ and 180◦, the orbit is said to be an **inclined orbit.**

Polar orbit

Prograde orbit

Eccentricity of the Orbit

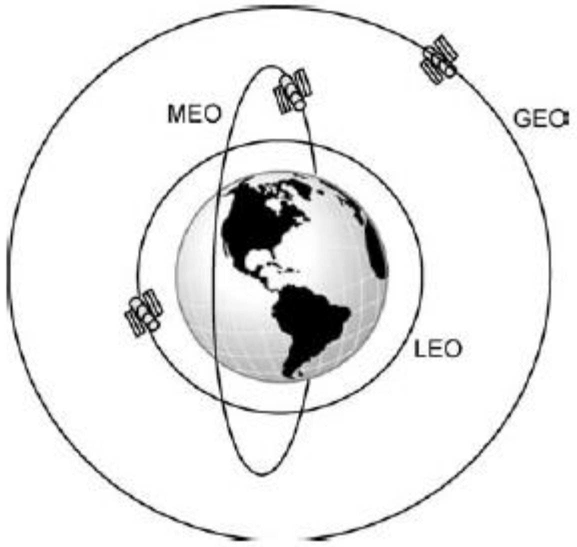
- On the basis of eccentricity, the orbits are classified as elliptical (Figure (a)) and circular (Figure (b)) orbits.
- **when the orbit eccentricity lies between 0 and 1,** the orbit is elliptical with the centre of the Earth lying at one of the foci of the ellipse.
- **When the eccentricity is zero**,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



(a) (b)

Distance from Earth

Depending upon the distance, these are classified as low Earth orbits (LEOs), medium Earth orbits (MEOs) and geostationary Earth orbits (GEOs), as shown in Figure



LEO

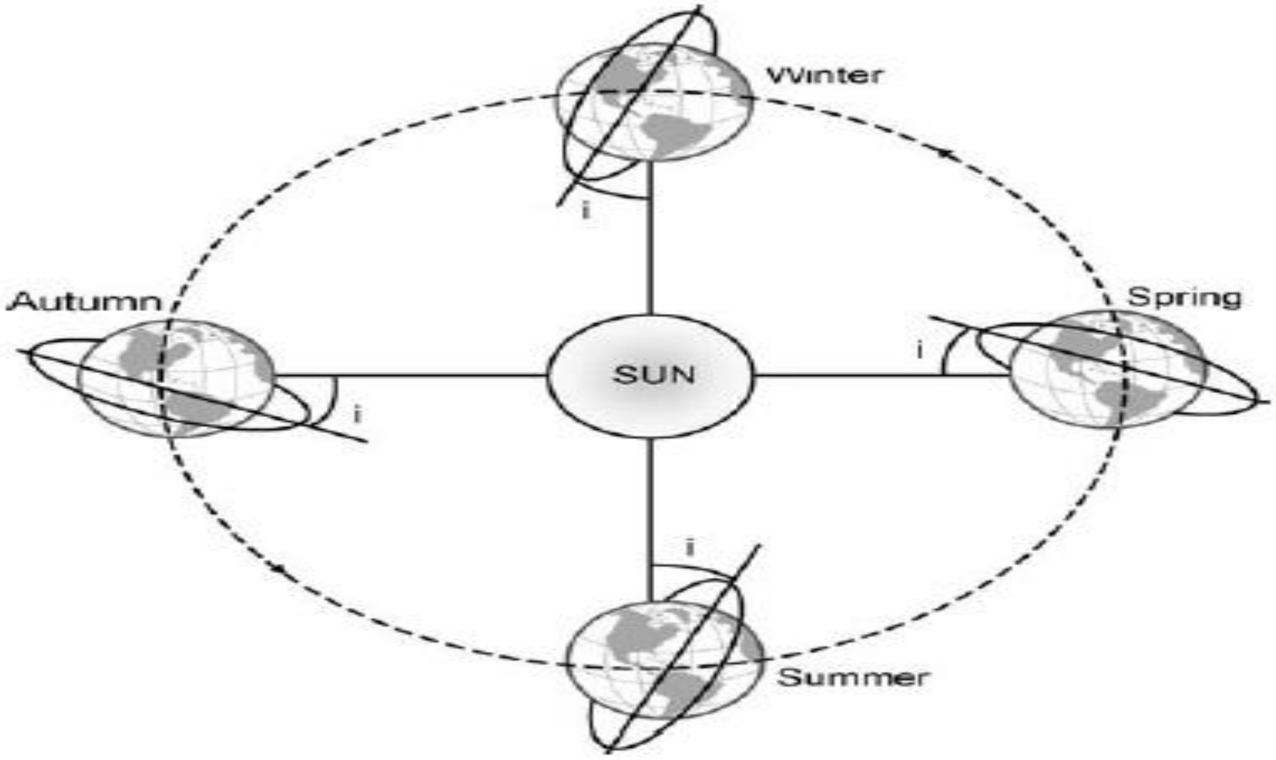
- Satellites in the low Earth orbit (LEO) circle Earth at a height of around **160 to 500 km above the surface of the Earth.**
- These satellites, being closer to the surface of the Earth, have much shorter orbital periods and smaller signal propagation delays.
- A lower propagation delay makes them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

MEO

- Medium Earth orbit (MEO) satellites orbit at a distance of approximately **10 000 to 20 000** km 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.

Sun-synchronous Orbit

- A **sun-synchronous orbit**, also known as a **helio-synchronous orbit**, is one that lies in a plane that **maintains a fixed angle** with respect to the **Earth–sun direction.**
- In other words, the orbital plane has a **fixed orientation** with respect to the **Earth–sun direction** and the angle between the orbital plane and the Earth–sun line remains constant throughout the year.
- Satellites in sun-synchronous orbits are particularly suited to applications like **passive remote sensing, meteorological, military reconnaissance and atmospheric studies.**


Winter
Autumn
Spring
SUN
Summer
i
i
i
i

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 bodies like **solar and lunar attraction, magnetic field interaction, solar radiation pressure, asymmetry of Earth's gravitational field etc.**

- Due to these factors, the satellite **orbit tends to drift** and its orientation also changes and hence the true orbit of the satellite is different from that defined using Kepler's laws.

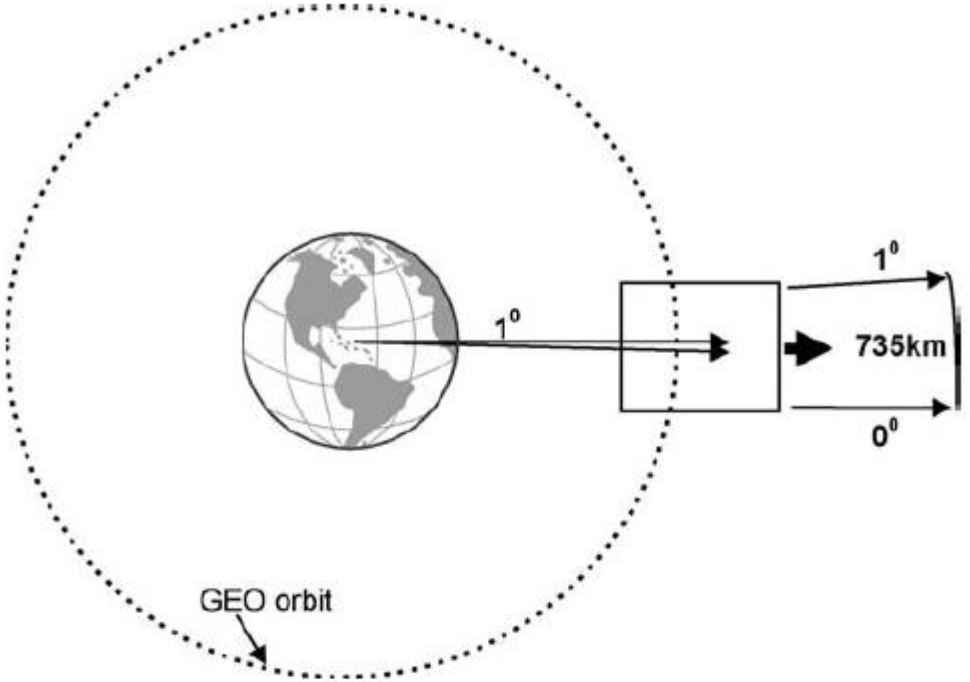
- The satellite's position thus needs to be **controlled** both in the east–west as well as the north–south directions.

- The **east–west location** needs to be maintained to prevent **radio frequency (RF) interference** from neighboring 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.**

- The **north–south orientation** has to be maintained to have proper satellite inclination.

Orbital Perturbations



Drift of a geostationary satellite

Orbital Perturbations

- In addition to the variation in the gravitational field of the Earth, the **satellite is also subjected to the gravitational pulls of the sun and the moon.**

- The Earth's orbit around the sun is an ellipse whose plane is inclined at an **angle of 7 degree** with respect to the equatorial plane of the sun.

- The **Earth is tilted around 23degree away** from the normal to the ecliptic. The moon revolves around the Earth with an inclination of around **5** ∘ to the equatorial plane of the Earth.

- Hence, the satellite in orbit is subjected to a variety of out-of-plane forces which change the inclination on the satellite's orbit.
- 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.

Orbital Perturbations

$$t_A = \frac{2\pi}{\omega_{\text{mod}}}$$

$$\omega_{\text{mod}} = \omega_0 \left[1 + \frac{K(1 - 1.5\sin^2 i)}{a^2(1 - e^2)^{3/2}} \right]$$

$\omega 0$ is the angular velocity for spherical Earth,
K = 66 063.1704km2,
a is the semi-major axis,
e is the eccentricity
and i = cos−1 WZ,
WZ is the Z axis component of the orbit normal.

Orbital Perturbations

- The **attitude and orbit control system** maintains the satellite's position and its orientation and keeps the antenna pointed correctly in the desired direction.
- The orbit control is performed by firing thrusters in the desired direction or by releasing jets of gas. It is also referred to as station keeping.
- **Thrusters** and gas jets are used to correct the longitudinal drifts (in-plane changes) and the inclination changes (out-ofplane changes).
- It may be mentioned that the **manoeuvres** required for correcting longitudinal drifts (referred to as the north-south manoeuvre) require a much larger velocity increment as compared to the manoeuvres required for correcting the inclination changes (referred to as the east-west manoeuvre).
- Hence, generally a different set of thrusters or gas jets is used for north-south and east-west manoeuvres.

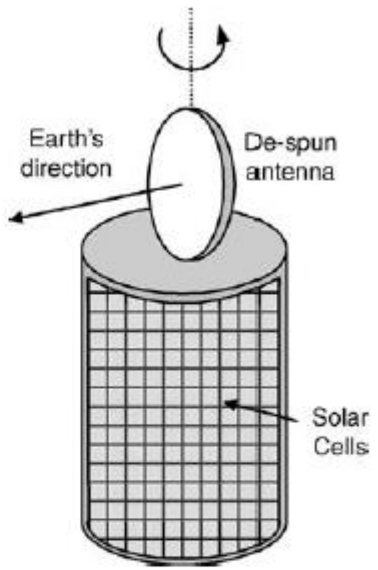
Satellite Stabilization

Commonly employed techniques for satellite attitude control include:

1. Spin stabilization
2. Three-axis or body stabilization

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 (Figure 3.28)



Spin stabilization

- 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(Resist Angular Acceleration).

There are **two types of spinning configurations employed in spin-stabilized satellites**. These

- Simple spinner configuration
- Dual spinner configuration.

Spin stabilization

Simple spinner configuration:

- The **satellite payload** and **other subsystems** are placed in the **spinning section**, while the **antenna** and **the feed** are placed in the de-spun platform.
- The **de-spun** platform is spun in a direction opposite to that of the spinning satellite body.

Dual spinner configuration:

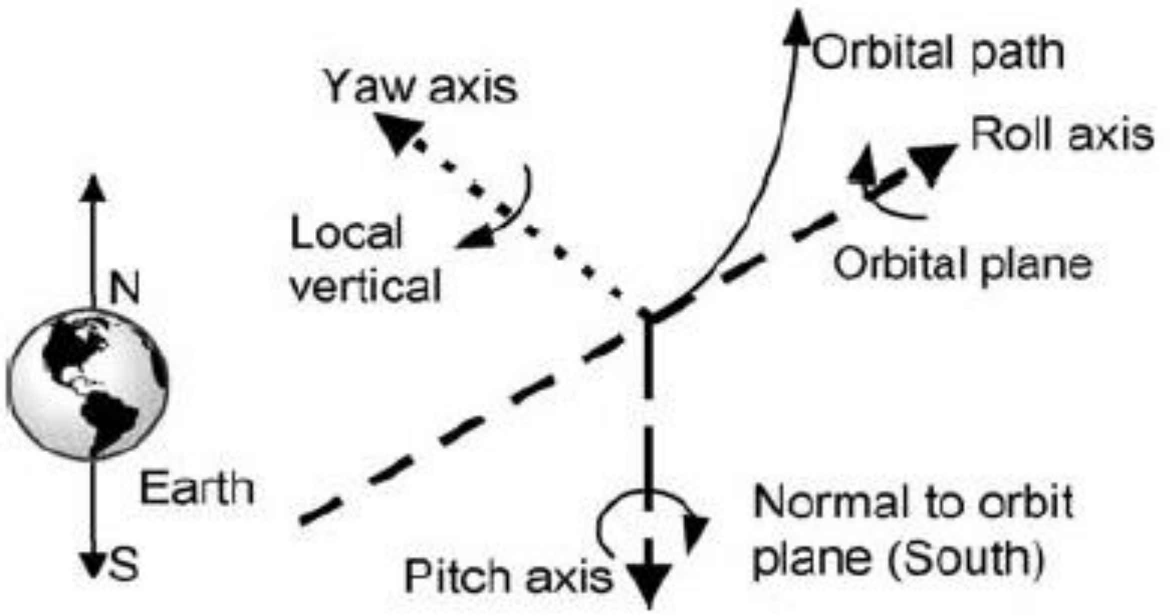
- The **entire payload along with the antenna and the feed** is placed on the **de-spun platform** and the other subsystems are located on the spinning body

Spin stabilization

- Modern spin-stabilized satellites almost invariably employ the **dual spinner configuration**.
- In both configurations, **solar cells are mounted on the cylindrical body** of the satellite.
- Intelsat-1 to Intelsat-4, Intelsat-6 and TIROS-1 are some of the popular spin-stabilized satellites.

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.**



Three-axis or Body Stabilization

- 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 change of the wheel.
- An alternative approach is to **use reaction wheels.**
- **Three reaction wheels are used,** one for each axis. They can be rotated in either direction depending upon the active correction force.
- The satellite body is generally box shaped for three-axis stabilized satellites.
- Antennae are mounted on the **Earth-facing side** and on the lateral sides adjacent to it.
- Some popular satellites belonging to the category of three-axis stabilized satellites include Intelsat-5, Intelsat-7, Intelsat-8, GOES-8, GOES-9, TIROS-N and the INSAT series of satellites.

Comparison between Spin-stabilized and Three-axis stabilized Satellites

1. In comparison to spin-stabilized satellites, **three-axis stabilized satellites have more power generation capability** and more additional mounting area available for complex antennae structures.
2. **Spin-stabilized satellites are simpler** in design and less expensive than three-axis stabilized satellites.

Comparison between Spin-stabilized and Three-axis stabilized Satellites

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2. **Spin-stabilized satellites are simpler** in design and less expensive than three-axis stabilized satellites.

Orbital Effects on Satellite's Performance

- As we know the satellite is revolving constantly around the Earth.
- The motion of the satellite has significant effects on its performance.
- These include the,
  - Doppler shift
  - Effect due to variation in the orbital distance
  - Effect of solar eclipse
  - Sun's transit outrage.

Doppler shift:

- The geostationary satellites appear stationary with respect to an Earth station terminal whereas in the case of satellites orbiting in low Earth orbits(LEO), the satellite is in relative motion with respect to the terminal.
- As the satellite is moving with respect to the Earth station terminal, the frequency of the satellite transmitter also varies with respect to the receiver on the Earth station terminal**. If the frequency transmitted by the satellite is f_T , then the received frequency f_R is given by equation**.

$$\left(\frac{f_R - f_T}{f_T}\right) = \left(\frac{\Delta f}{f_T}\right) = \left(\frac{v_T}{v_P}\right)$$

Where,

- *vT* 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)

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.

Solar Eclipse

- There are times when the satellites do not receive solar radiation due to obstruction from a celestial body. **During these periods the satellites operate using onboard batteries.**
- The **design of 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.
- The satellite is designed in such a manner so as to cope with these **thermal stresses.**

Sun Transit Outrage

- There are times when the satellite passes **directly between the sun and the Earth as shown in Figure 3.32.**
- 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).
- 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.

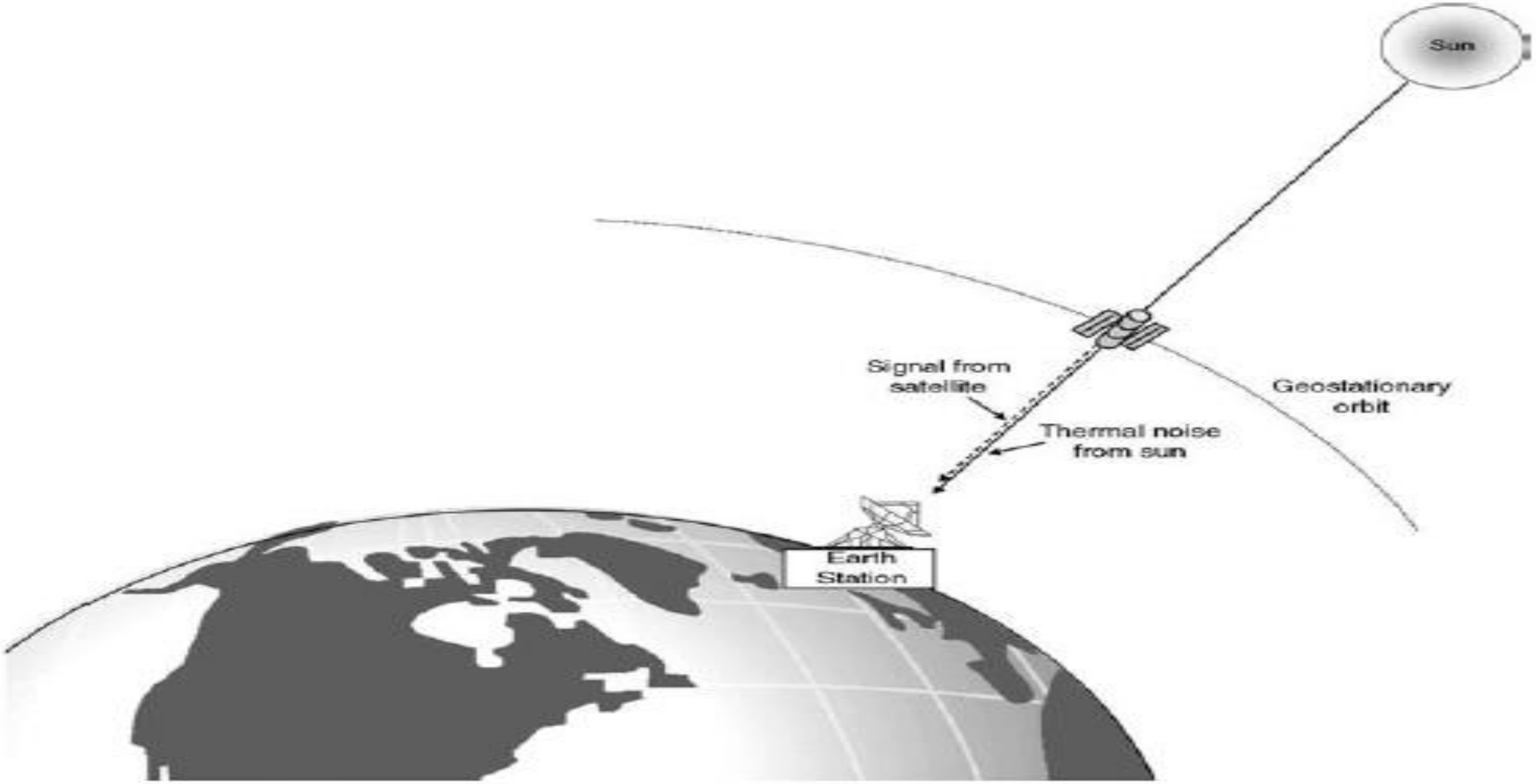


Figure 3.32 Sun outrage conditions

Eclipses

- With **reference to satellites**, an eclipse is said to occur when the **sunlight fails to reach the satellite's solar pane**l due to an **obstruction from a celestial body.**
- The major and most frequent source of an eclipse is due to the **satellite coming in the shadow of the Earth.**
- **This is known as a solar eclipse.**
- The eclipse is total; i.e. the satellite fails to receive any light whatsoever if it passes through the **umbra, which is the dark central region** of the shadow, and receives **very little light if it passes through the penumbra,** which is the less dark region surrounding the umbra

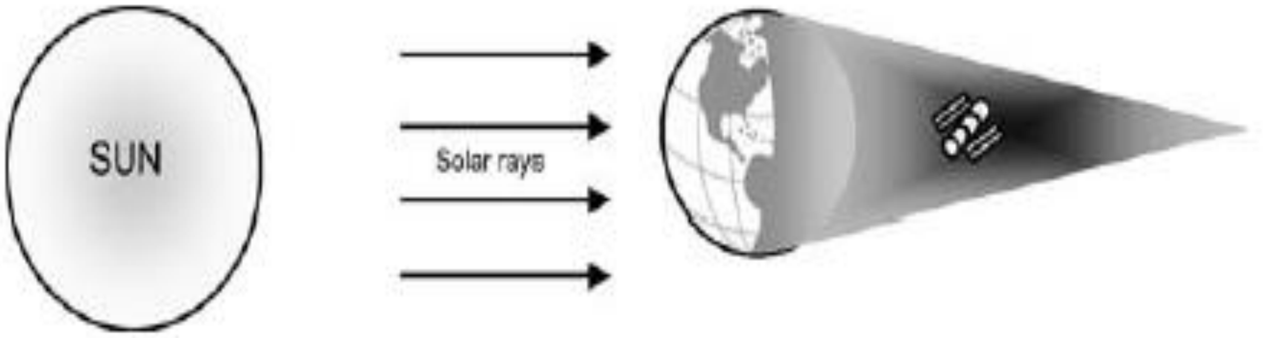
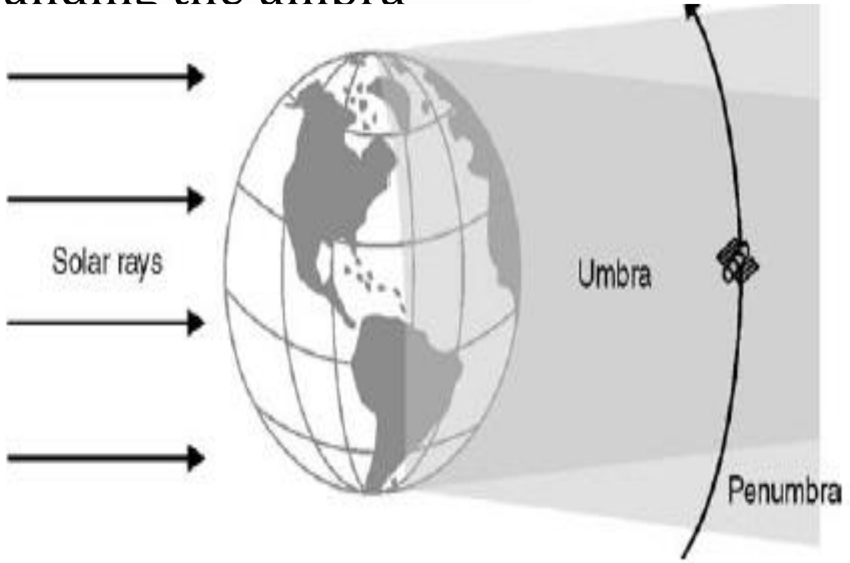
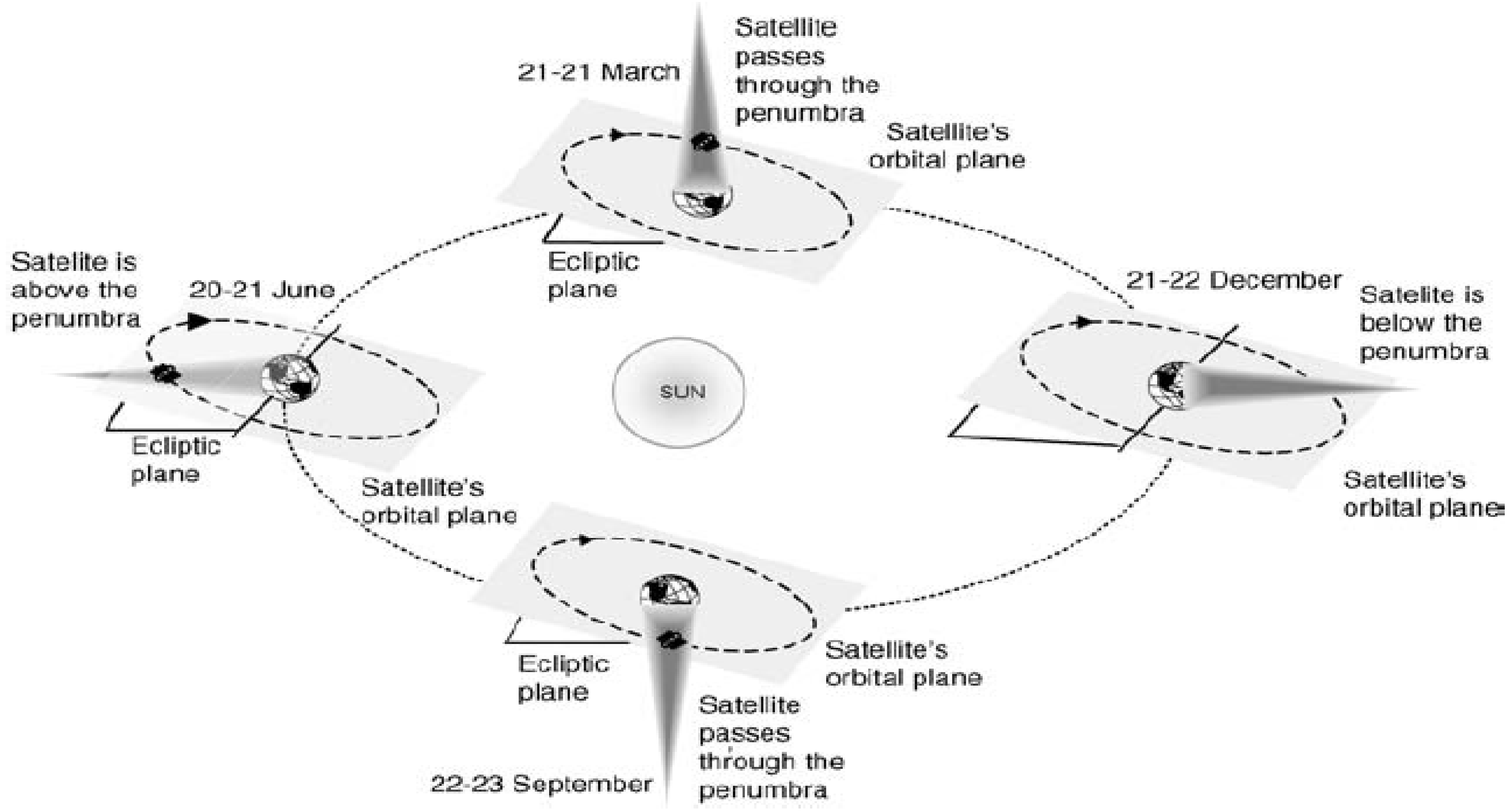


Figure 3.33 Solar eclipse



- The **eclipse occurs** as the **Earth's equatorial plane** is inclined at a constant angle of about **23.5°** to its **ecliptic plane**.
- The eclipse is seen on **42 nights** during the spring and an equal number of nights during the autumn by the geostationary satellite.
- The effect is the worst during the **equinoxes** and **lasts for about 72 minutes.**
- **The equinox is the point in time when the sun crosses the equator, making the day and night equal in length**. The spring and autumn equinoxes respectively occur on 20–21 March and 22–23 September.
- During the **equinoxes** in March and September, the satellite, the Earth and the sun are aligned at midnight local time and the satellite spends **about 72 minutes in total darkness.**
- From **21 days before and 21 days after the equinoxes,** the satellite crosses the umbral cone each day for some time, there by 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

Eclipses



Positions of the geostationary satellite during the equinoxes and solstices

Eclipses

- 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.

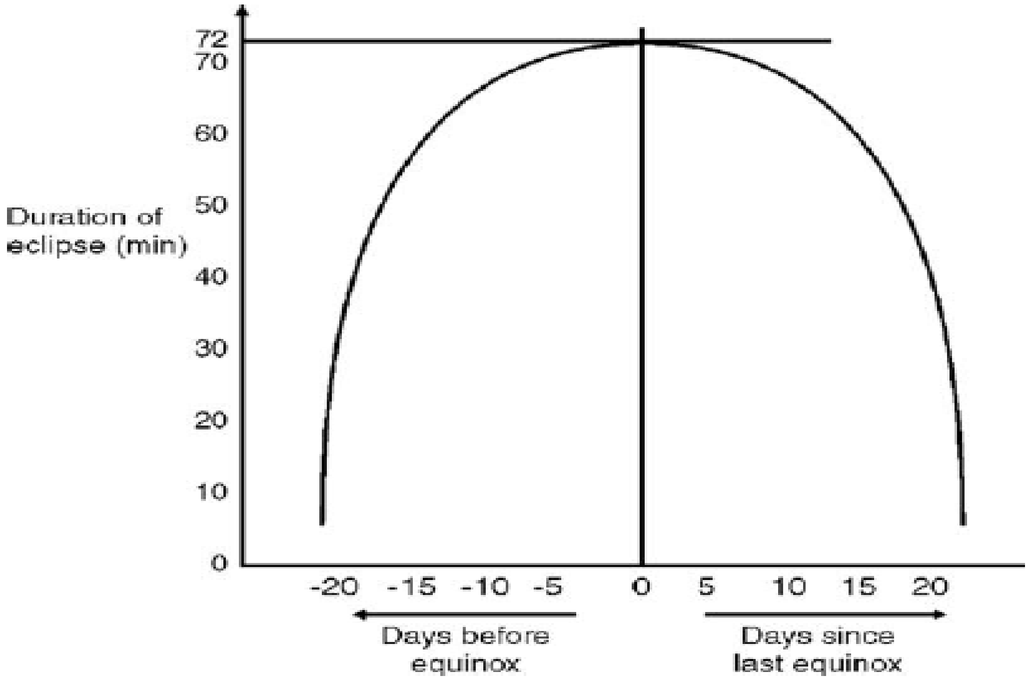


Figure 3.36 Duration of the eclipse before and after the equinox

Eclipses

Another type of eclipse known as **the lunar eclipse** occurs when the moon's shadow passes across the satellite 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.

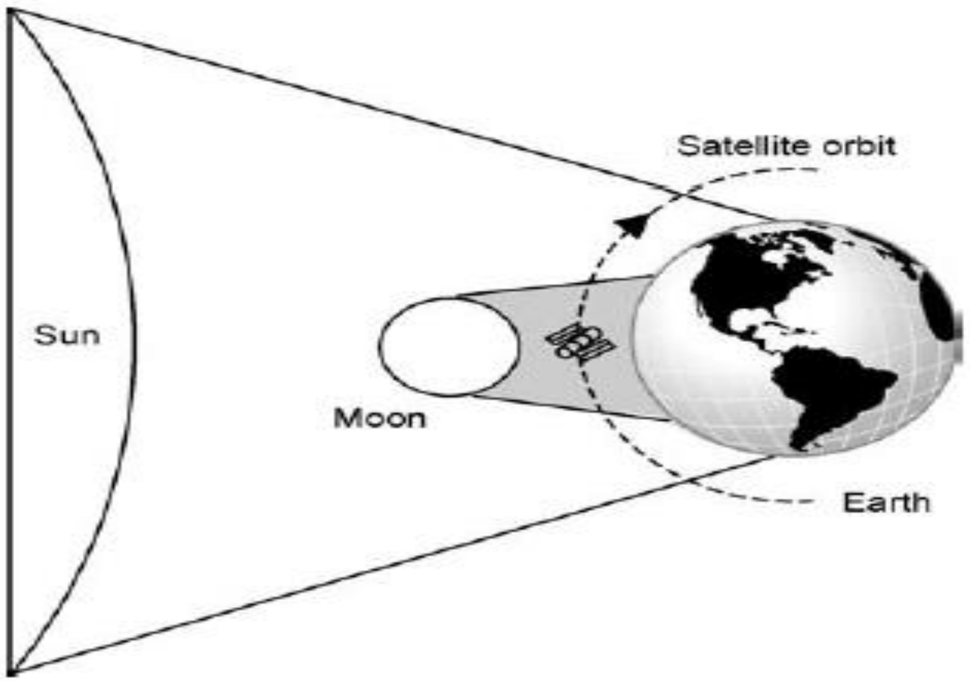


Figure 3.37 Lunar eclipse

Look Angles of a Satellite

- 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.

- 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.**

- 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

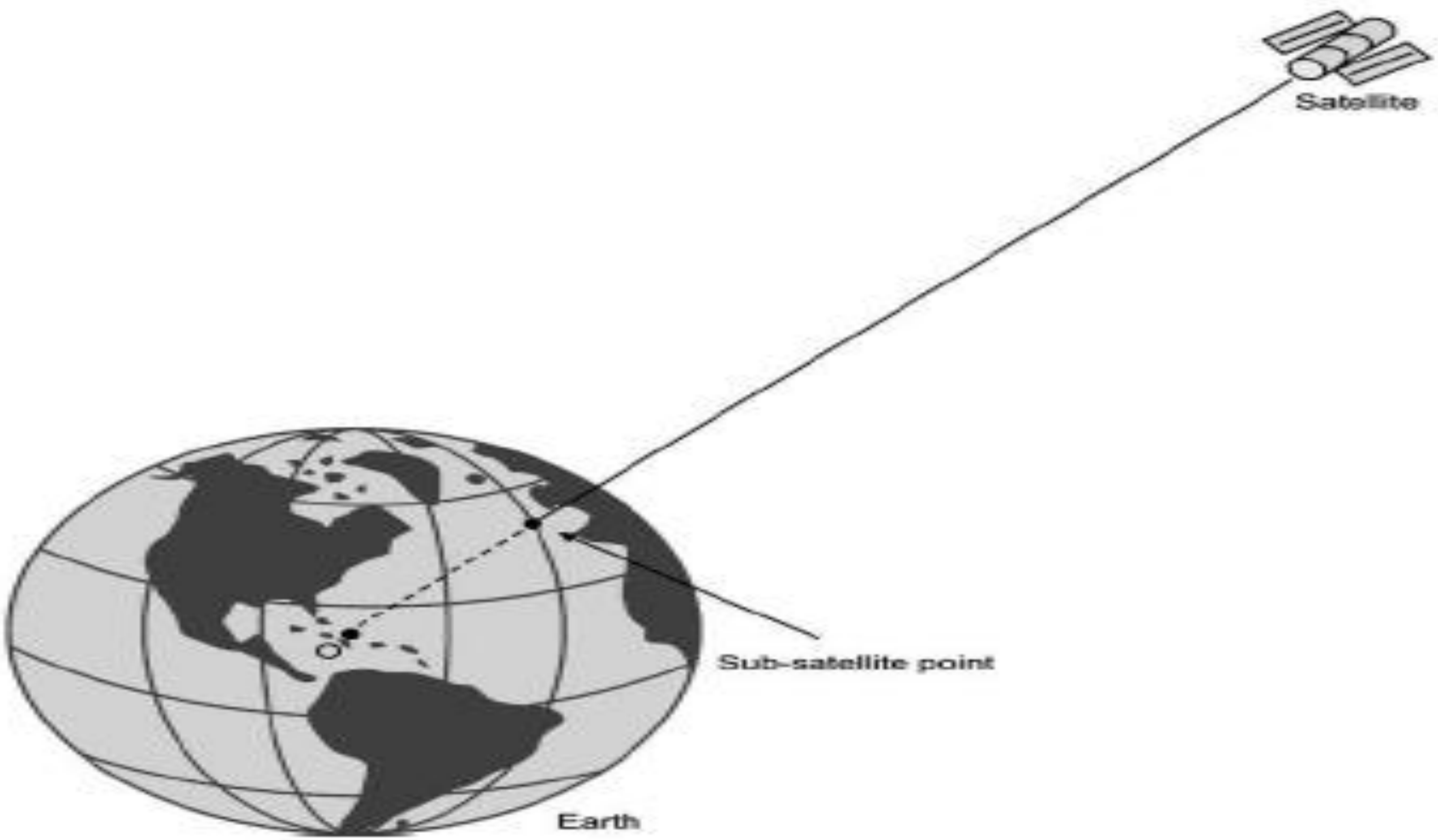


Figure 3.38 Sub-satellite point

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** (Figure 3.39). 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, the azimuth angle can be computed as follows:

Earth station in the northern hemisphere:

$A = 180^\circ - A$, when the Earth station is to the west of the satellite

$A = 180^\circ + A$. when the Earth station is to the east of the satellite

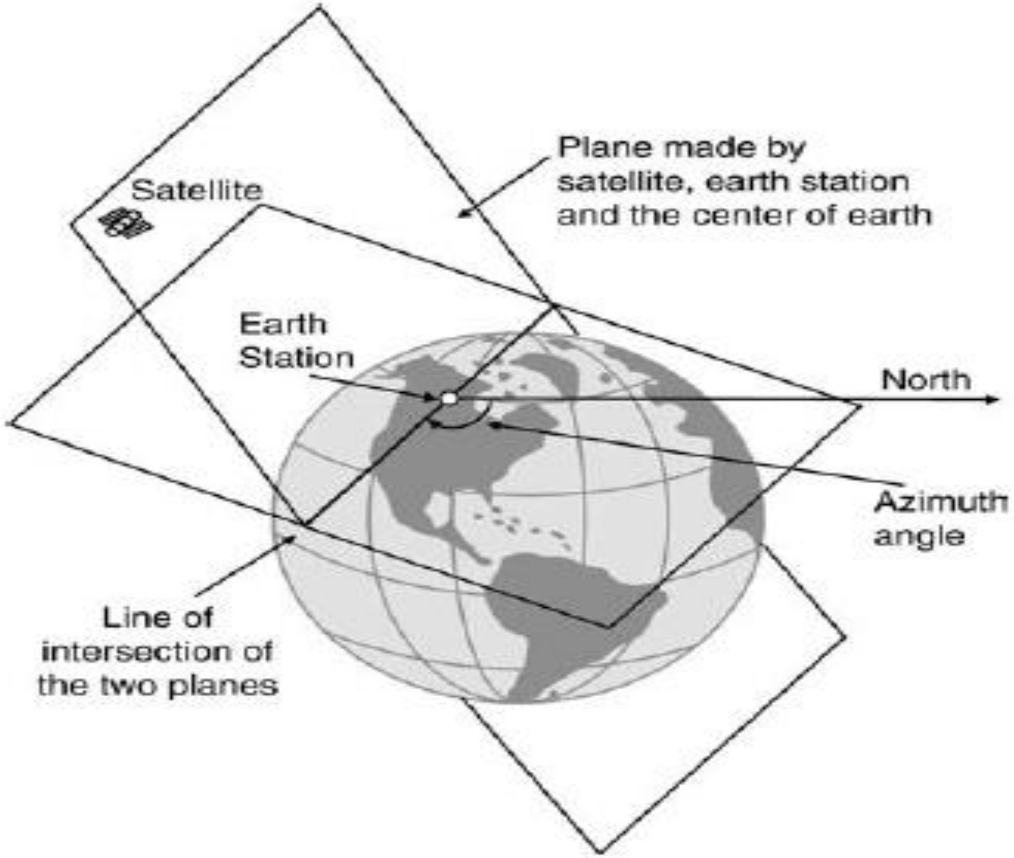


Figure 3.39 Azimuth angle

Earth station in the southern hemisphere:

$$A = A' \ldots \quad \text{when the Earth station is to the west of the satellite} \tag{3.21}$$

$$A = 360^\circ - A' \ldots \quad \text{when the Earth station is to the east of the satellite} \tag{3.22}$$

where A' can be computed from

$$A' = \tan^{-1}\left(\frac{\tan|\theta_S - \theta_L|}{\sin\theta_l}\right) \tag{3.23}$$

where

θ_S = satellite longitude

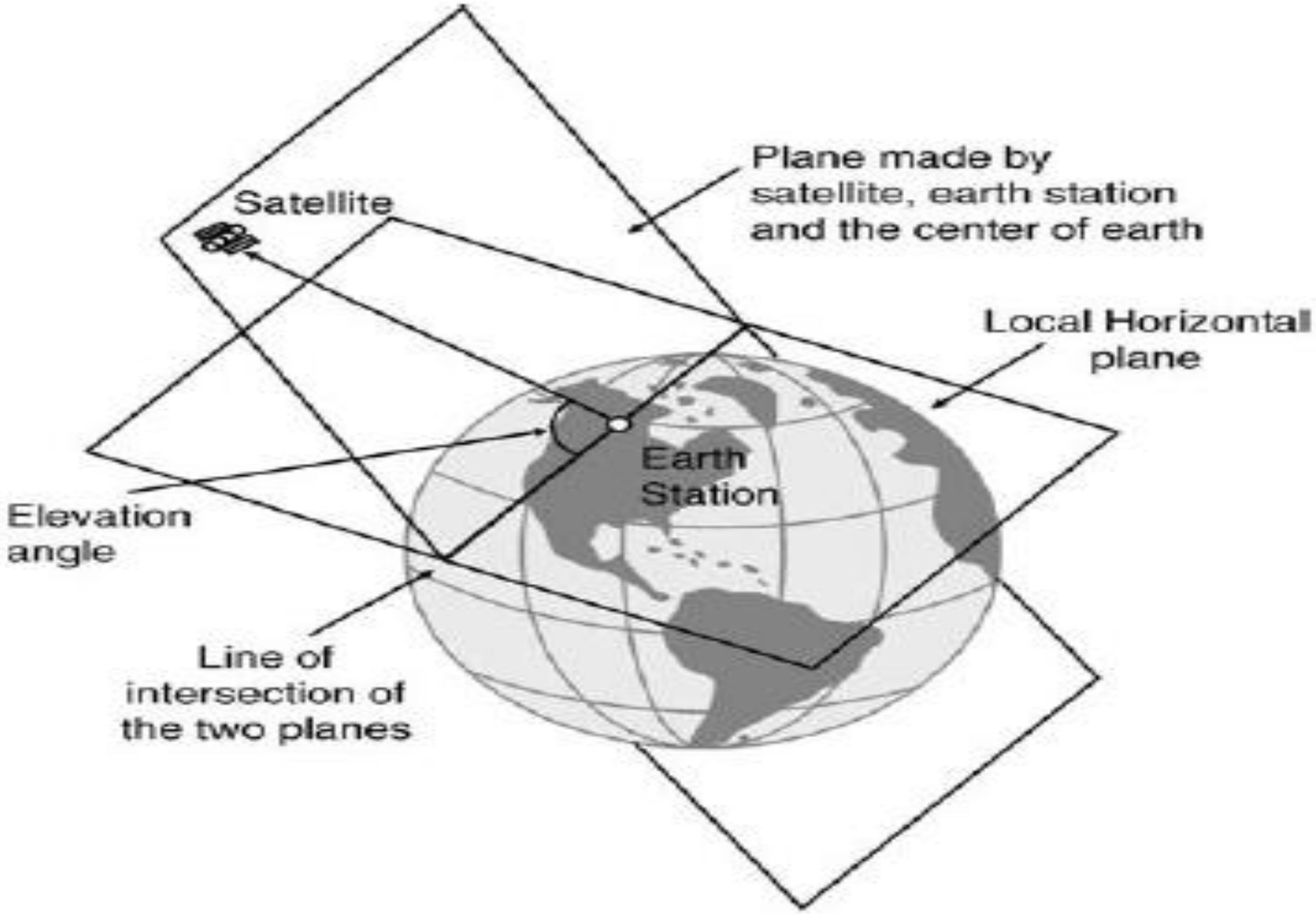
θ_L = Earth station longitude

θ_l = Earth station latitude

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. Figures 3.40 (a) and (b) show the elevation angles for two different satellite and Earth station positions. It can be computed from

$$E = \tan^{-1}\left[\frac{r - R\ \cos\theta_l\ \cos|\theta_s - \theta_L|}{R\sin\{\cos^{-1}(\cos\theta_l\cos|\theta_s - \theta_L|)\}}\right] - \cos^{-1}(\cos\theta_l\cos|\theta_s - \theta_L|) \qquad (3.24)$$


Plane made by
satellite, earth station
and the center of earth
Satellite
Local Horizontal
plane
Earth
Station
Elevation
angle
Line of
intersection of
the two planes

(a)

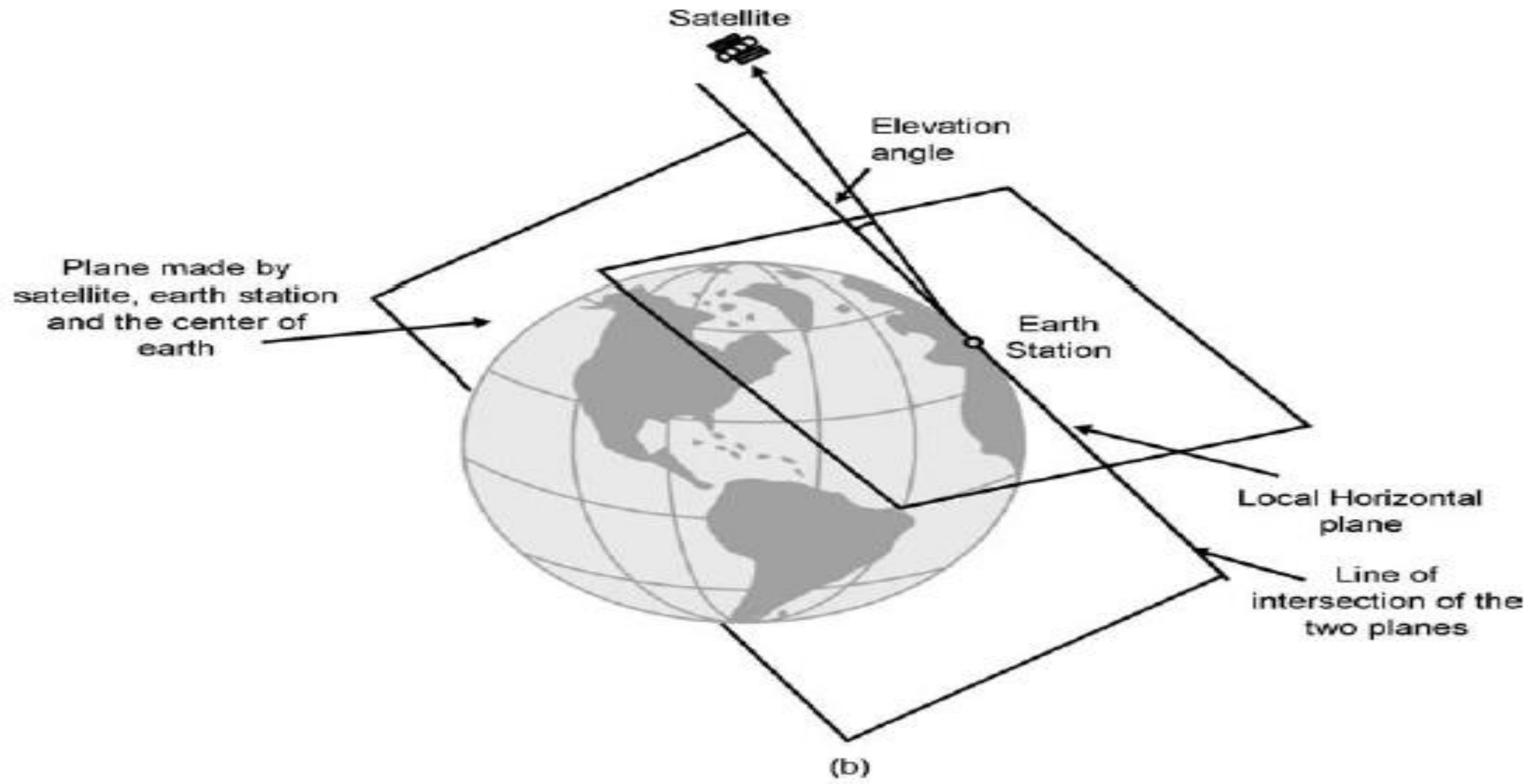


Figure 3.40 Earth station elevation angle

where

r = orbital radius, R = Earth's radius
θ_s = Satellite longitude, θ_L = Earth station longitude, θ_l = Earth station latitude