Optical & Satellite Communications

Orbital Mechanics, Launchers & Satellites

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SATELLITE

A satellite in general is any natural or artificial body moving around a celestial body such as a planet or a star. In the present context, 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.

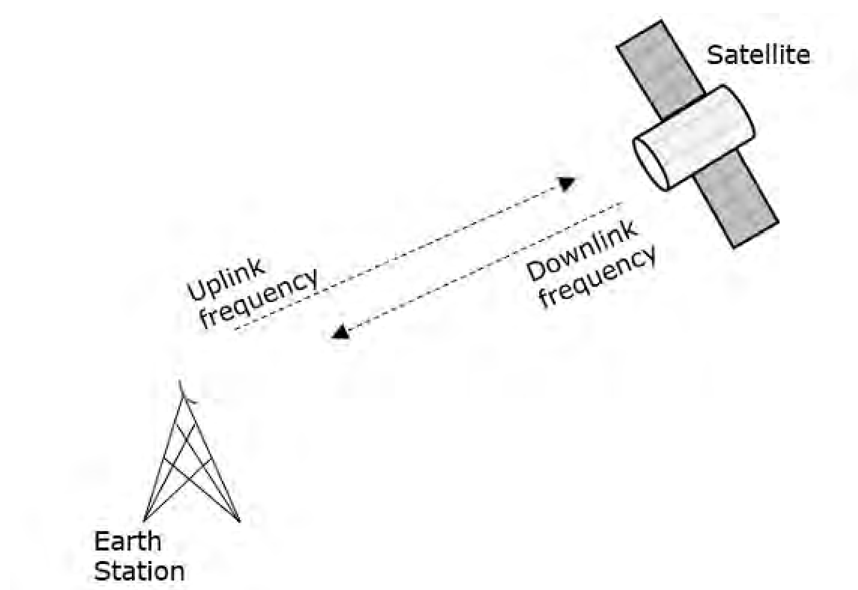
The fundamental principle to be understood concerning satellites is that a satellite is a projectile. That is to say, a satellite is an object upon which the only force is gravity. Once launched into orbit, the only force governing the motion of a satellite is the force of gravity.

A **communication satellite** is nothing but a **microwave repeater** station in space that is helpful in telecommunications, radio, and television along with internet applications.

A **repeater** is a circuit which increases the strength of the signal it receives and retransmits it. But here this repeater works as a **transponder**, which changes the frequency band of the transmitted signal, from the received one.

The frequency with which the signal is sent into the space is called **Uplink frequency**, while the frequency with which it is sent by the transponder is **Downlink frequency**.

The following figure illustrates this concept clearly.



Satellite Communication − Advantages

- Flexibility
- Ease in installing new circuits

- Distances are easily covered and cost doesn't matter
- Broadcasting possibilities
- Each and every corner of earth is covered
- User can control the network

Satellite Communication − Disadvantages

The initial costs such as segment and launch costs are too high.

- Congestion of frequencies
- Interference and propagation

Satellite Communication − Applications

- In Radio broadcasting.
- In TV broadcasting such as DTH.
- In Internet applications such as providing Internet connection for data transfer, GPS applications, Internet surfing, etc.
- For voice communications.
- For research and development sector, in many areas.
- In military applications and navigations.

ENVIRONMENTAL EFFECTS ON SATELLITES

Vacuum

The hard vacuum of space will cause out gassing which is the release of volatiles from materials. The outgassed molecules then deposit on line of sight surfaces and are more likely to deposit on cold surfaces. This molecular contamination can affect optical properties of vehicle and payload surfaces and space craft performance particularly for sensitive optics.

Atomic oxygen

It is produced when short wavelength UV radiation react with molecular oxygen in the upper atmosphere. Atomic oxygen oxidises many metals especially silver, copper and osmium. Atomic oxygen react strongly with any material containing carbon, nitrogen, sulphur and hydrogen bonds. Polymers react and erode.

Ultraviolet radiation

UV radiation damages polymers by cross linking (hardening) or chain scission (weakening) them. UV under high vacuum also create oxygen vacancies in oxides leading to significant color changes.

Ionising Radiations

It is due to charge particle radiation (from galactic cosmic rays, solar proton events and the trapped radiation belts). Particulate radiation or ionising radiation cause cross linking or chain scission and result in damage to materials. It can also cause electronic circuits to malfunction.

Plasma

Space plasma is composed of positively charged oxygen ions and free electrons and it varies with solar activity and altitude. Plasma can lead to negative charge build up leading to ion sputtering, arcing and parasitic currents in solar arrays as well as re-attraction of contamination.

Micrometeoroids

Micrometeoroids moving at very high velocities (greater than 50 km/s) are there in space. They can hit space craft and damage them unless satellites perform detection and avoidance manoeuvres. Small debris cannot be completely avoided.

Thermal extremes and thermal cycling

Thermal extremes and thermal cycling is experienced by satellites (-120^0C to 120^0 C). This can degrade the materials/equipments with time. Protective coating degrade with time and lead to cracking and peeling. Pinholes are formed in the coating and atomic oxygen will attack the underlying material.

ORBIT

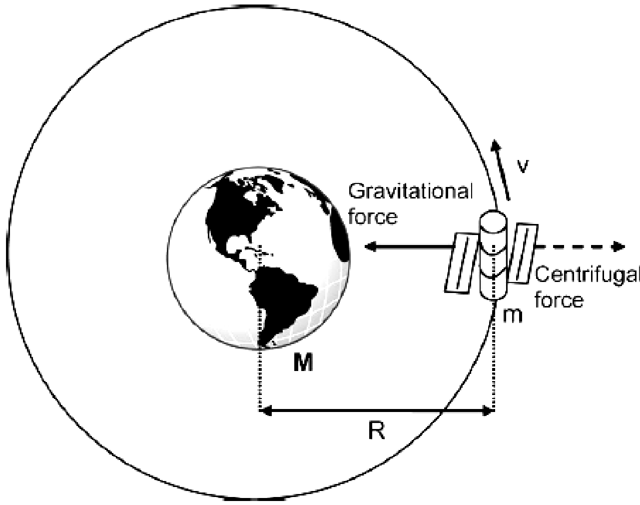
While a trajectory is a path traced by a moving body, an orbit is a trajectory that is periodically repeated. While the path followed by the motion of an artificial satellite around Earth is an orbit, the path followed by a launch vehicle is called the launch trajectory. The motion of different planets of the solar system around the sun and the motion of artificial satellites around Earth are examples of orbital motion.

TRAJECTORY

The term ‘trajectory’ 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.

BASIC PRINCIPLES OF SATELLITES (FORCES ON ARTIFICIAL SATELLITES)

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.



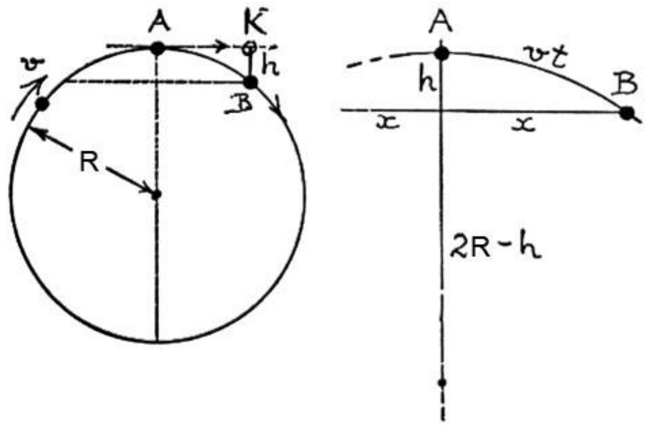
The **centrifugal force** is the force exerted during circular motion, by the moving object upon the other object around which it is moving.

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. Centripetal force further leads to a corresponding acceleration called **centripetal acceleration** as it causes a change in the direction of the satellite's velocity vector.

The **centrifugal force** is simply the reaction force exerted by the satellite in a direction opposite to that of the centripetal force. This is in accordance with **Newton's third law** of motion, which states that for every action there is an equal and opposite reaction. This implies that there is a centrifugal acceleration acting outwards from the centre of the Earth due to the centripetal acceleration acting towards the centre of the Earth.

Proof of $F = mv^2/R$

Consider following figure



The circle represents the orbit of a satellite of radius R, moving with speed v . The satellite moves from A to B in a time t. Without a force the satellite would have moved to K at constant speed.

Now 'switch-on' gravity and the satellite will fall a distance h in the same time from the tangent from A to the point B.

From crossed chords property

$$h(2R - h) = x^2$$

But 2R >> h

∴ $$2v = x^2$$

So $$h = \frac{x^2}{2R} \quad (1)$$

Now, x = AK which is almost the

$$\text{arc AB} = vt \quad (2)$$

From (1) and (2)

$$h = \frac{(vt)^2}{2R} \quad (3)$$

h is vertical fall and so using

$$s = \frac{1}{2}at^2 \quad (4)$$

From (3) and (4)

$$\frac{1}{2}at^2 = \frac{(vt)^2}{2R}$$

leading to $$a = \frac{v^2}{R}$$

Using F = ma

$$F = \frac{mv^2}{R}$$

NEWTON'S LAW OF GRAVITATION

According to 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 and written as

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

where m_1, m_2 = masses of the two particles

r = distance between the two particles

G =gravitational constant= 6.67 × 10−11 m3⁄kg s2

NEWTON'S SECOND LAW OF MOTION

According to Newton's second law of motion, the force equals the product of mass and acceleration. 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. This is the **centrifugal force** directed outwards from the centre of the Earth and for a satellite is equal in magnitude to the gravitational force.

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

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

where m_1 = mass of Earth

m_2 = mass of the satellite

$\mu = Gm_1 = 3.986013 \text{ X } 10^5 \text{ km}^3/\text{s}^2 = 3.986013 \text{ x } 10^{14} \text{ Nm}^2/\text{kg}$

The orbital period in such a case can be computed from

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

The velocity at any point on an elliptical orbit at a distance d from the centre of the Earth is given by

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

where a = semi-major axis of the elliptical orbit

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

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

Example

A satellite is orbiting Earth in a uniform circular orbit at a height of 630 km from the surface of Earth. Assuming the radius of Earth and its mass to be 6370 km and 5.98 x 10^{24} kg respectively, determine the velocity of the satellite (Take the gravitational constant G = 6.67 x 10^{-11} Nm^2/kg^2).

Solution

Orbit radius R = 6370 + 630 = 7000 km = 7000000 m

Also, constant μ = GM = 6.67 x 10^{-11} x 5.98 x 10^{24}

= 39.8 x 10^{13} N m^2/kg

= 39.8 x 10^{13} m^3/s^2

The velocity of the satellite can be computed from

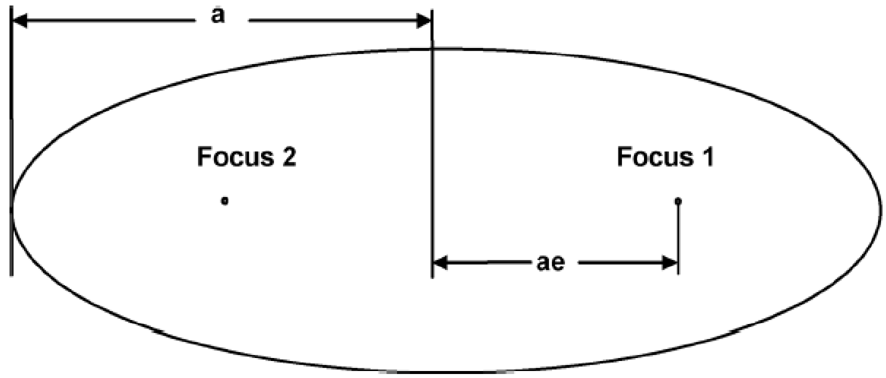
$$V = \sqrt{\left(\frac{\mu}{R}\right)} = \sqrt{\left(\frac{39.8x10^{13}}{7000,000}\right)} = 7.54\,km/s$$

KEPLER'S THREE LAWS OF PLANETARY MOTION

- The orbit of any smaller body about a larger body is always an ellipse, with the center of mass of the larger body as one of the two foci.
- The orbit of the smaller body sweeps out equal areas in equal time.
- The square of the period of revolution of the smaller body about the larger body equals a constant multiplied by the third power of the semi major axis of the orbital ellipse. That is $T^2 = 4\pi r^2/\mu$ where T is the orbital period, a is the semi major axis of the orbital ellipse, and μ is **Kepler's constant**.

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 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.

Other important parameters of an elliptical satellite orbit include its **apogee** (farthest point of the orbit from the Earth's centre) and **perigee** (nearest point of the orbit from the Earth's centre) distances.

The sum of the kinetic energy (K.E.) and the potential energy (P.E.) of a satellite always remain constant. The value of this constant is equal to **$-Gm_1m_2/(2a)$,** where

The kinetic and potential energies of a satellite at any point at a distance r from the centre of the Earth are given by

$$K.E. = \frac{1}{2}(m_2 v^{2)}$$

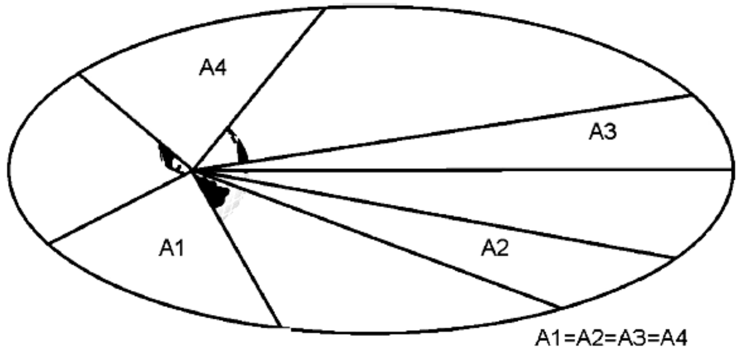
$$P.E. = -\frac{Gm_1 m_2}{r}$$

$$\therefore \quad \frac{1}{2}(m_2 v^2) - \frac{Gm_1 m_2}{r} = -\frac{Gm_1 m_2}{2a}$$

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

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; i.e. the rate (dA/dt) at which it sweeps area A is constant.

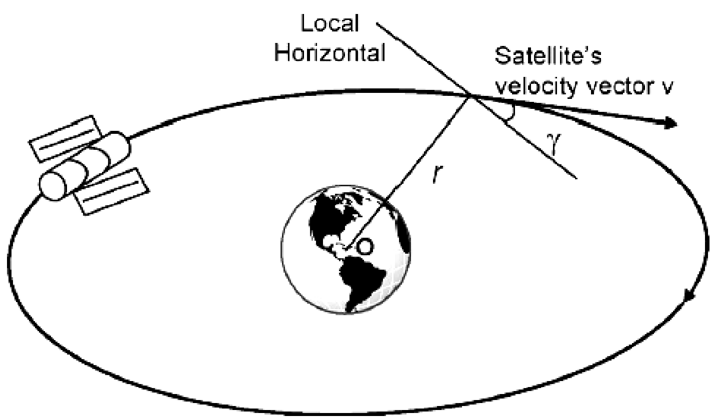


The rate of change of the swept-out area is given by

dA/dt = angular momentum of the satellite/2m

where m is mass of satellite.

The angular momentum of the satellite of mass m is given by $mr^2\omega$, where ω is the angular velocity of the satellite. This further implies that the product $mr^2\omega = (m\omega r)(r) = mv'r$ remains constant. Here r′ is the component of the satellite's velocity v in the direction perpendicular to the radius vector and is expressed as vcosγ, where γ is the angle between the direction of motion of the satellite and the local horizontal, which is in the plane perpendicular to the radius vector r.



This leads to the conclusion that the product rvcosγ is constant.

The product reduces to rv in the case of circular orbits and also at **apogee** and **perigee** points in the case of elliptical orbits due to angle γ becoming zero.

This implies that the satellite is at its lowest speed at the apogee point and the highest speed at the perigee point. In other words, for any satellite in an elliptical orbit, the dot product of its velocity vector and the radius vector at all points is constant. Hence,

$$v_p r_p = v_a r_a = vr\cos\gamma$$

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

According to the Kepler's third law, also known as the law of periods, 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.

Equating the gravitational force with the centrifugal force gives

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

Put $v = \omega r$

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

which gives $\omega^2 = \frac{Gm_1}{r^3}$

Put $\omega = 2\pi / T$

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

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

Example (AMIE Winter 2018, 8 marks)

A satellite is in a 322 km high circular orbit. Determine (i) the orbital angular velocity in radians per second (ii) the orbital period in minutes (iii) the orbital velocity in metres per second.

Assume the average radius of the earth is 6378.137 km and Kepler's constant has the value $3.986\ 004\ 418 \times 10^5\ km^3/s^2$.

Solution

Orbital velocity

$$v = \left(\frac{\mu}{r}\right)^{1/2} = \left(\frac{3.986004418x10^5}{6700.137}\right) = 7.713066\,km/s = 7713.066\,m/s$$

Orbital period

$$T = \frac{(2\pi r^{3/2})}{\sqrt{\mu}} = \frac{(2\pi x 6700.137^{3/2})}{(3.986004418x10^5)^{1/2}} = 5458.037372\sec = 90.97\min utes$$

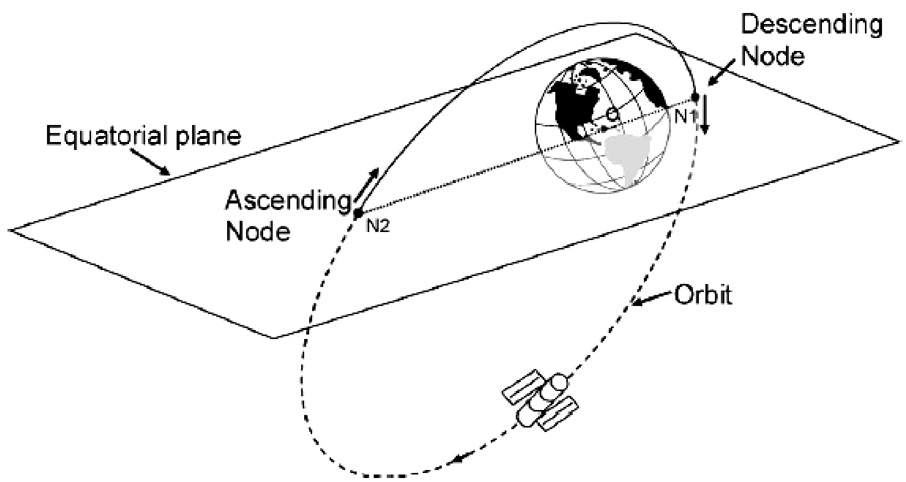
Orbital angular velocity

The orbital period from above is 5458.037372 seconds. One revolution of the earth covers 360° or 2π radians. Hence 2π radians are covered in 5458.037 372 seconds, giving the orbital angular velocity as 2π/5458.037 372 rad/s = 0.001 1512 rad/s.

ORBITAL PARAMETERS

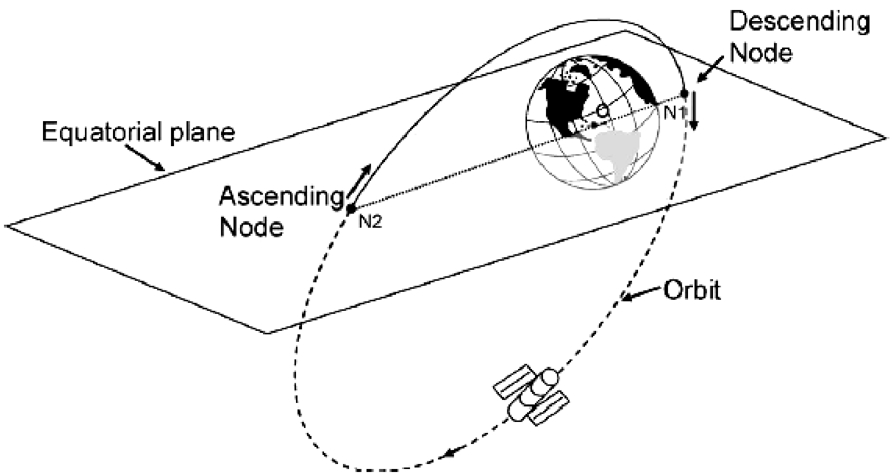
Ascending and descending nodes

The satellite orbit cuts the equatorial plane at two points: the first, called the descending node (N_1), where the satellite passes from the northern hemisphere to the southern hemisphere, and the second, called the ascending node (N_2), where the satellite passes from the southern hemisphere to the northern hemisphere.



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 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**.

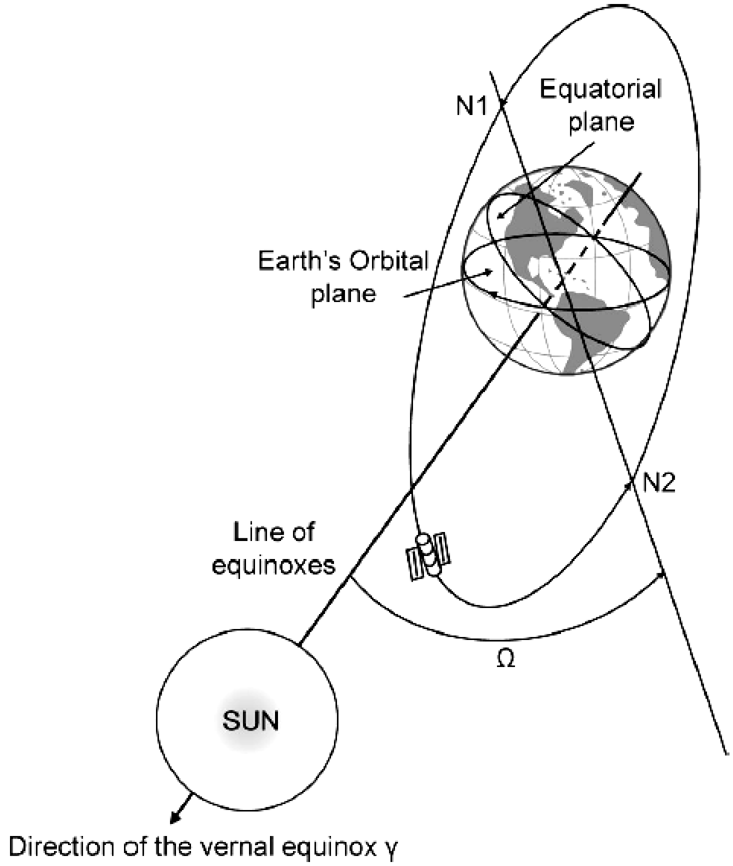


The sinusoidal variation of the angle of inclination is defined by

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

where T is 365 clays. 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 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**. 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).



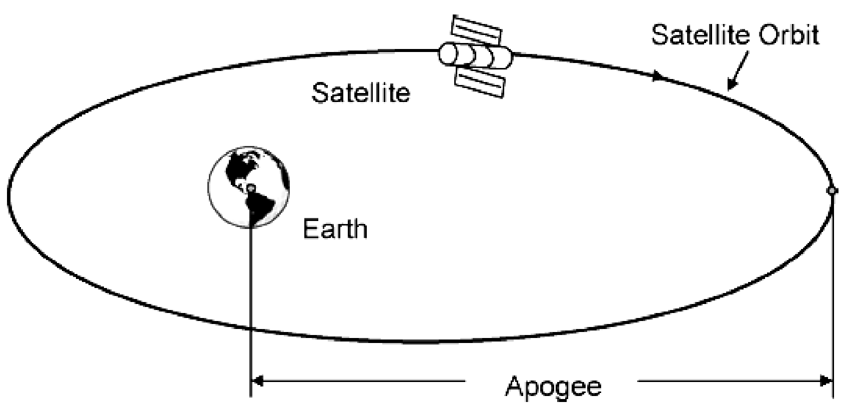
vernal equinox

Solstices

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**.

Apogee

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



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)

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

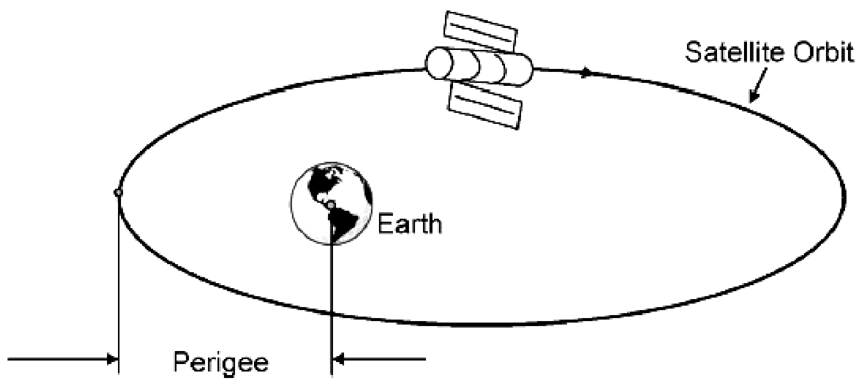
$$V_p = \sqrt{\left(\frac{2\mu}{Perigee\,distance} - \frac{2\mu}{Perigee\,distance + Apogee\,distance}\right)}$$

where $$V_p = V\left(\frac{d\cos\gamma}{Perigee\,distance}\right)$$

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

Perigee

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



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)

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:

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

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

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

where a and b are semi-major and semi-minor axes respectively.

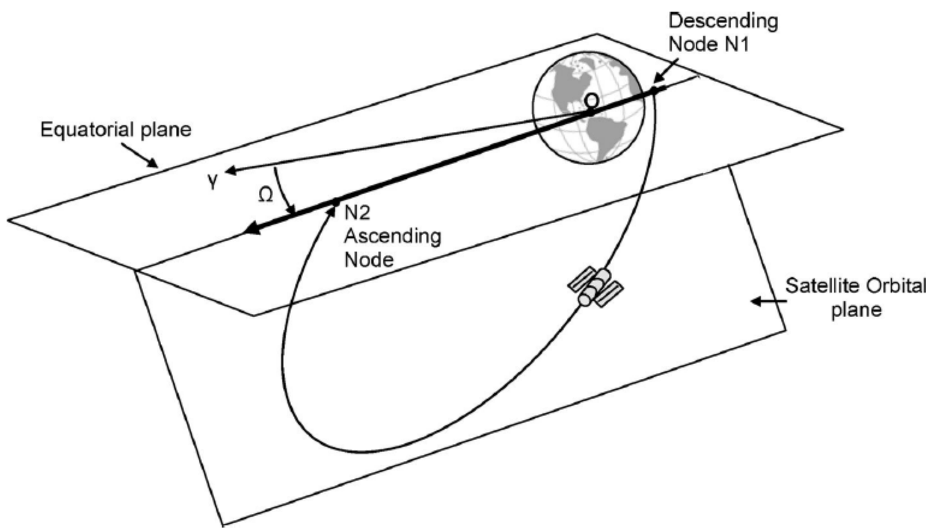
Semi-major axis

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

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

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 Ω measured from the vernal equinox towards the line of nodes in the direction of rotation of Earth. The angle could be anywhere from 0° to 360°.



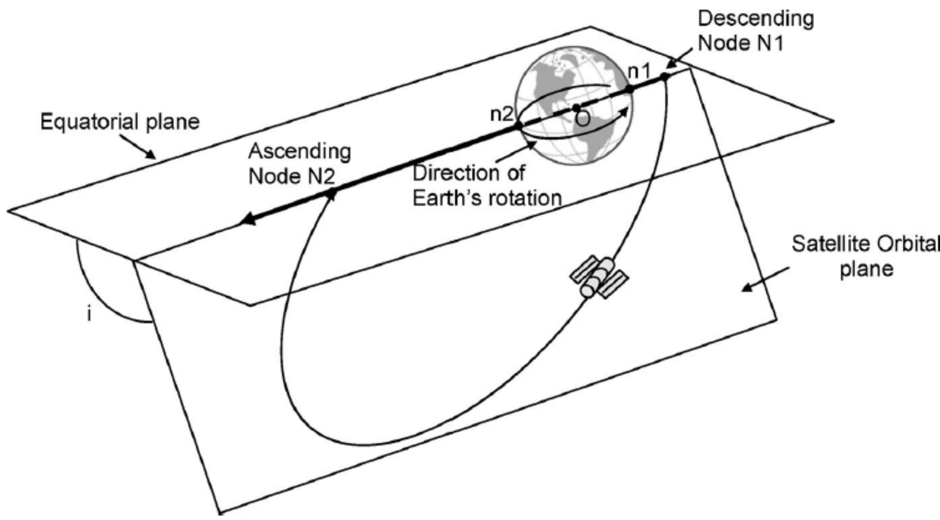
Angle Ω can be computed as the difference between two angles. One is the angle α between the direction of the vernal equinox and the longitude of the injection point and the other is the angle β between the line of nodes and the longitude of the injection point.

$$\sin\beta = \frac{\cos i \sin l}{\cos l \sin i}$$

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

Inclination

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

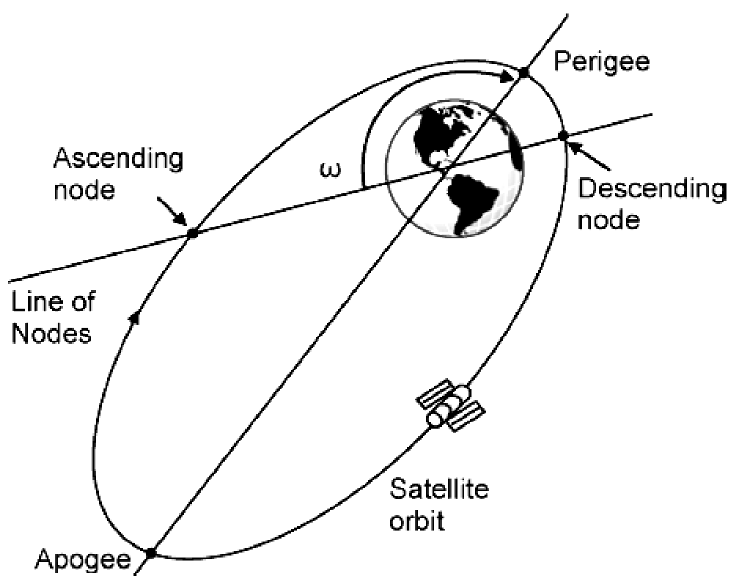


The inclination angle can be determined from the latitude l at the injection point and the angle A_z between the projection of the satellite's velocity vector on the local horizontal and north. It is given by

$$\cos i = \sin A_z \cos l$$

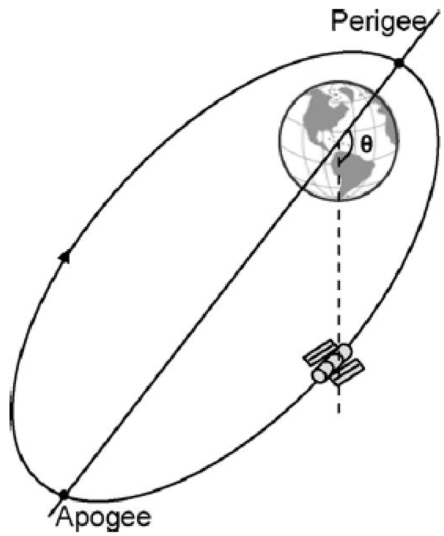
Argument of the perigee

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.



True anomaly of the satellite

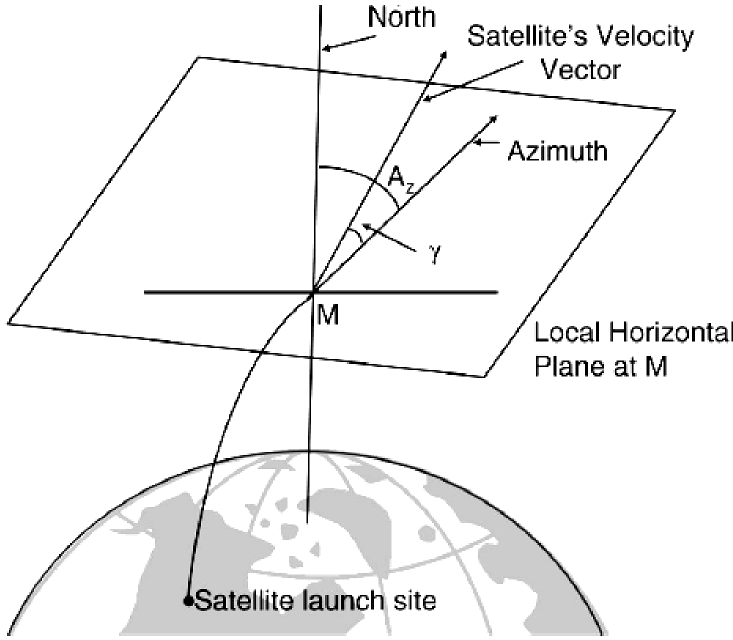
This parameter is used to indicate the position of the satellite in its orbit. This is done by defining an angle θ. called the true **anomaly** of the satellite, formed by the line joining the perigee and the centre of the Earth with the line joining the satellite and the centre of the Earth.



True anomaly of a satellite

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 A_z between the north and the projection of the satellite's velocity vector on the local horizontal.



Angles defining the direction of the satellite

Example

The apogee and perigee distances of a satellite orbiting in an elliptical orbit are respectively 45000 km and 7000 km. Determine the following:

1. *Semi-major axis of the elliptical orbit*
2. *Orbit eccentricity*
3. *Distance between the centre of the Earth and the centre of the elliptical orbit*

Solution

Semi major axis, a

$$a = \frac{apogee + perigee}{2} = \frac{45000 + 7000}{2} = 26000\,km$$

Eccentricity

$$e = \frac{apogee - perigee}{2a} = \frac{45000 - 7000}{2x26000} = \frac{38000}{52000} = 0.73$$

Distance between centre of earth and centre of ellipse

This distance is given by a x e = 26000 x 0.73 = 18980 km

Example

A satellite is moving in an elliptical orbit with the major axis equal to 42000 km. If the perigee distance is 8000 km, find the apogee and the orbit eccentricity.

Solution

Apogee

Major axis = 42000 km

Also, major axis = apogee + perigee = 42000 km

Therefore apogee = 42000 - 8000 = 34000 km

Eccentricity

$$e = \frac{apogee - perigee}{major\, axis} = \frac{34000 - 8000}{42000} = 0.62$$

Example

The elliptical eccentric orbit of a satellite has its semi-major and semi-minor axes as 25000 km and 18330 km respectively. Determine the apogee and perigee distances.

Solution

Eccentricity

$$e = \frac{\sqrt{(a^2 - b^2)}}{a} = \frac{\sqrt{(25000^2 - 18330^2)}}{25000} 00.68$$

Apogee distance

It will be $a(1+e) = 25000 x 1.68 = 42000\, km$

Perigee distance

It will be $a(1-e) = 25000 x 0.32 = 8000\, km$

Example

Calculate the orbital period of a satellite moving in an elliptical orbit having a major axis of 50000 km. (Take μ = 39.8 x 10^{13} Nm^2/kg).

Solution

Semi-major axis, a

a = 50 000/2 = 25000 km

Orbital time period, T

$$T = 2\pi\sqrt{\left(\frac{a^3}{\mu}\right)} = 2x3.14x\sqrt{\frac{25000000^3}{39.8x10^{13}}}$$

= 39 344 seconds = 10 hours 55 minutes 44 seconds

Problem

A satellite is in an elliptical orbit with a perigee of 1000 km and an apogee of 4000 km. Using a mean earth radius of 6378.14 km, find the period of the orbit in hours, minutes, and seconds, and the eccentricity of the orbit.

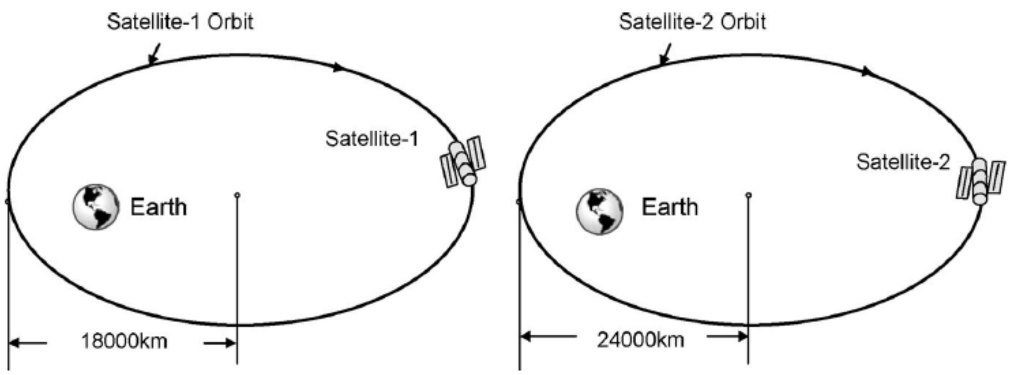
Answer: 2 hours 18 minutes 45.19 seconds

Hint: The major axis of the elliptical orbit is a straight line between the apogee and perigee. Hence, for a semi major axis length a, earth radius re, perigee height h_p, and apogee height h_a,

$$2a = 2r_e + h_p + h_a = 2 \times 6378.14 + 1000.0 + 4000.0 = 17756.28 \text{ km}$$

Example

The semi-major axes of the two satellites shown in the figure are 18000 km (satellite 1) and 24000 km (satellite 2). Determine the relationship between their orbital periods.



Solution

We know

$$T_1 = 2\pi\sqrt{\left(\frac{a_1^3}{\mu}\right)}$$

and $$T_1 = 2\pi\sqrt{\left(\frac{a_2^3}{\mu}\right)}$$

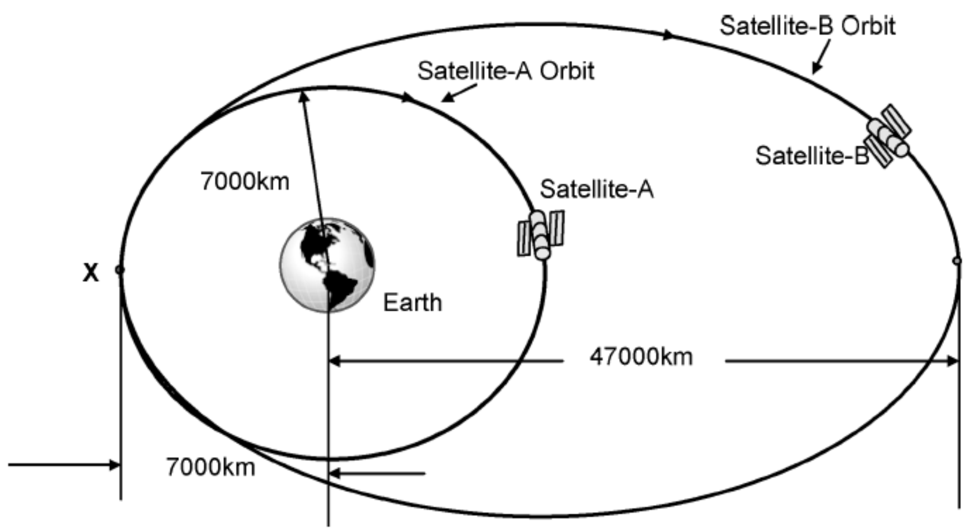
This gives

$$\frac{T_2}{T_1} = \left(\frac{a_2}{a_1}\right)^{3/2} = \left(\frac{24000}{18000}\right)^{3/2} = 1.54$$

The orbital period of satellite 2 is 1.54 times that of satellite 1.

Example

Satellite A is orbiting Earth in a near-Earth circular orbit of radius 7000 km. Satellite B is orbiting Earth in an elliptical orbit with apogee and perigee distances of 47000 km and 7000 km respectively. Determine the velocities of the two satellites at point X. (Take μ = 39.8 x 10^{13} m^3/s^2.)



Solution

Velocity of satellite A at point X

The velocity of a satellite moving in a circular orbit is constant throughout the orbit and is given by

$$V = \sqrt{\frac{\mu}{R}}$$

Velocity of satellite A at point X = $\sqrt{\frac{39.8x10^{13}}{700,000}} = 7.54\, km/s$

Velocity of satellite B at point X

Velocity of the satellite at any point in an elliptical orbit is given by

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

where a is the semi-major axis and R is the distance of the point in question from the centre of the Earth.

Here R = 7000 km and a = (47 000 + 7000)/2 = 27000 km.

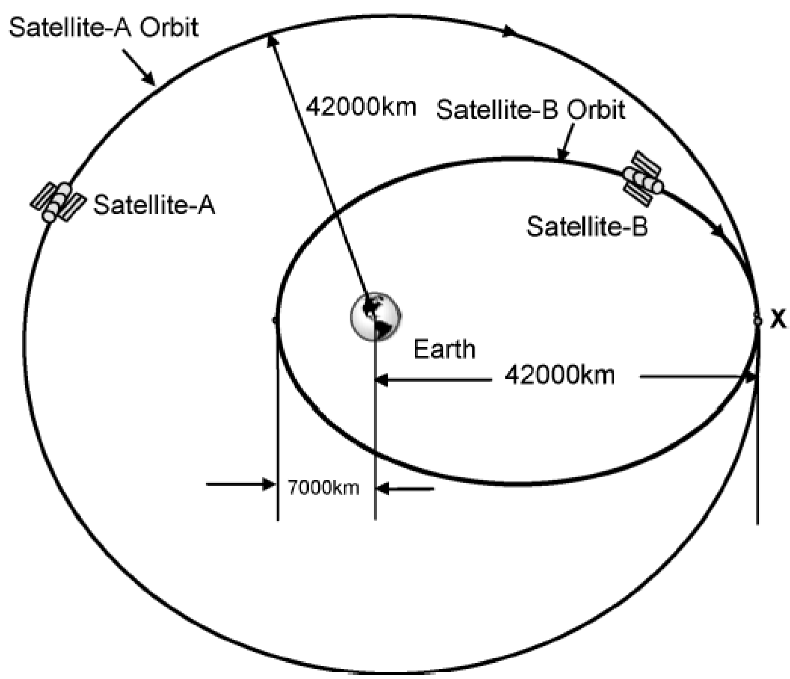
Therefore, velocity of satellite B at point X is given by

$$V = \sqrt{\left[\left(39.8x10^{13}\right)x\left(\frac{2}{7000,000} - \frac{1}{27000,000}\right)\right]}$$

= 9.946 km/s

Problem

Satellite A is orbiting Earth in an equatorial circular orbit of radius 42000 km. Satellite B is orbiting Earth in an elliptical orbit with apogee and perigee distances of 42000 km and 7000 km respectively. Determine the velocities of the two satellites at point X. (Take = 39.8 x 10^{13} m^3/s^2.)



Answer: 3.078 km/s, 1.645 km/s

LEO, MEO AND GEO SATELLITES

Low Earth orbits (LEOs)

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 low Earth orbits (LEOs), medium Earth orbits (MEOs) and geostationary Earth orbits (GEOs).

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. The system is intended to provide a variety of telecommunication services at the global level. Other applications where LEO satellites can be put to use are surveillance, weather forecasting, remote sensing and scientific studies.

Medium Earth orbit (MEO) satellites

Medium Earth orbit (MEO) satellites orbit at a distance of approximately 10000 to 20000 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.

Geostationary Earth orbits (GEOs).

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 36 000 km, 35 786 km to be precise, above the surface of the Earth.

Satellites in geostationary orbits play an essential role in relaying communication and TV broadcast signals around the globe. They also perform meteorological and military surveillance functions very effectively.

Example

Verify that a geostationary satellite needs to be at a height of about 35780 km above the surface of the Earth. Assume the radius of the Earth to be 6380 km and μ = 39.8 x 10^{13} Nm^2/kg

Solution

Orbital period of a satellite in circular orbit

The orbital period of a satellite in circular orbit is given by

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

Orbital period of a geostationary satellite

Orbital period of a geostationary satellite is equal to 23 hours, 56 minutes, which is equal to 86160 seconds.

$$2x3.14x(r)^{3/2} / \sqrt{(39.8x10^{13})} = 86160$$

Solving r = 4215.5 x 10^4 m = 42155 km

Height of satellite orbit above the Earth

Height of satellite orbit above the surface of the Earth = 42155 - 6380 = 35775 km

PARAMETERS DEFINING THE SATELLITE ORBIT

The satellite orbit is completely defined or specified by the following parameters:

- Right ascension of the ascending node
- Inclination angle
- Position of the major axis of the orbit
- Shape of the elliptical orbit
- Position of the satellite in its orbit

Right ascension of the ascending node

The angle Ω defining the right ascension of the ascending node is basically the difference between two angles, θ_1 and θ_2 where θ_1 is the angle made by the longitude of the injection point at the time of launch with the direction of vernal equinox and θ_2 is the angle made by the longitude of the injection point at the time of launch with the line of nodes, as shown in following figure.

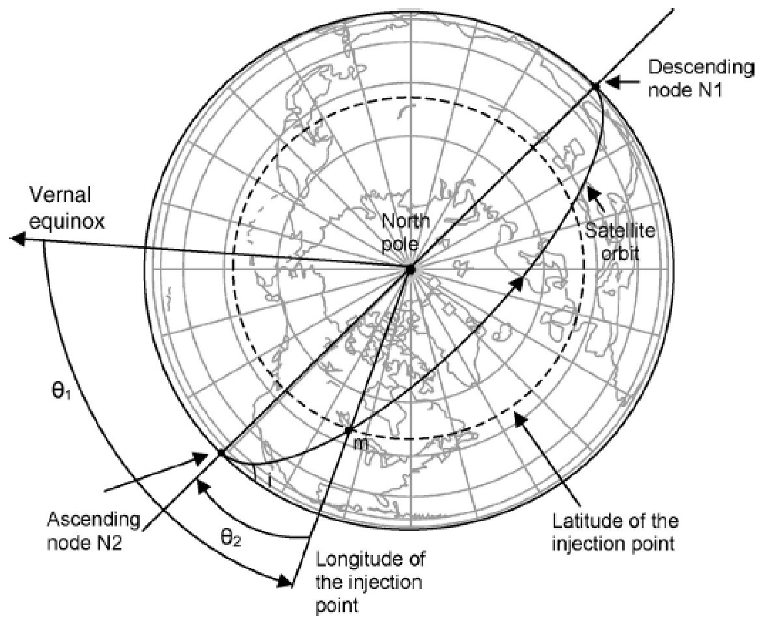
$$\sin\theta_2 = \frac{\cos i \sin L}{\cos L \sin i}$$

where

i = angle of inclination and

L = longitude of the injection point.

Thus, for a known angle of inclination, the time of launch and hence the longitude of the injection point can be so chosen as to get the desired angle (Ω).



Inclination angle

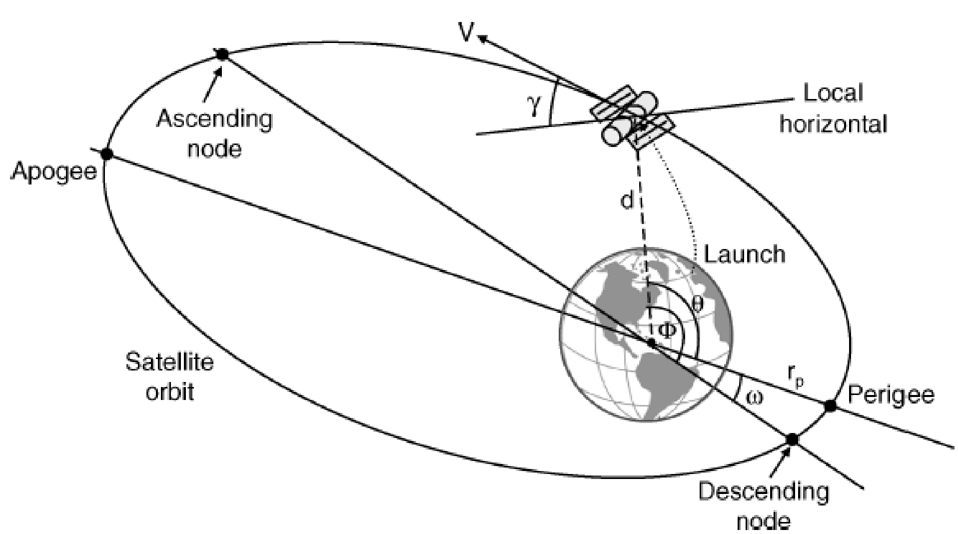
The angle of inclination (i) of the orbital plane can be determined from the known values of the angle of azimuth A_z and the latitude of the injection point l using the expression

$$\cos i = \sin A_z \cos l$$

With the help of this equation, it can be concluded that the satellite will tend to orbit in a plane which will be inclined to the equatorial plane at an angle equal to or greater than the latitude of the injection point.

Position of the major axis of the orbit

The position of the major axis of the satellite orbit is defined by argument of the perigee ω. Refer to following figure.



If the injection point happens to be different from the perigee point, then ω is the difference of two angles, ϕ and θ as shown in figure.

Angle ϕ can be computed from

$$\sin\phi = \frac{\sin l}{\sin i}$$

and θ can be found as

$$\cos\theta = \frac{dV^2\cos^2\gamma - \mu}{e\mu}$$

Shape of the elliptical orbit

The shape of the orbit is defined by the orbit eccentricity "e", the semi-major axis "a", the **apogee distance** "r_a" and the **perigee distance** "r_p".

The elliptical orbit can be completely defined by either "a" and "e" or by r_a and r_p.

$$r_p = \frac{V^2 d^2 \cos^2\gamma}{\mu(1+e)}$$

and

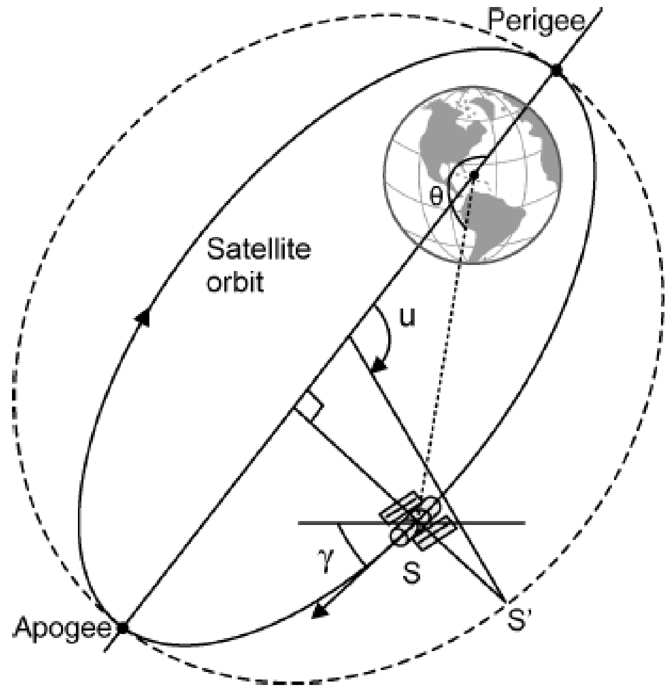
$$r_a = \frac{V^2 d^2 \cos^2\gamma}{\mu(1-e)}$$

Position of the satellite in its orbit

The position of the satellite in its orbit can be defined by a time parameter t, which is the time that elapsed after a time instant t_0 when the satellite last passed through a reference point. The reference point is usually the perigee. The time t that has elapsed after the satellite last passed through the perigee point can be computed from

$$t = \left(\frac{T}{2\pi}\right)(u - e\sin u)$$

where T is the orbital period and the angle "u" is the eccentric anomaly of the current location of the satellite as shown in following figure.



Example

A geostationary satellite was injected into a highly eccentric elliptical transfer orbit from a height of 200 km. The projection of the injection velocity vector in the local horizontal plane made an angle of 85° with the north. Determine the inclination angle attained by this transfer orbit given that latitude and longitude are 5.2°N and 52.7°W respectively.

Solution

The inclination angle can be computed from

$$\cos i = \sin A_z \cos l$$

where

A_z = azimuth angle of the injection point = 85^0

l = latitude of the launch site = 5.2^0

$\therefore \quad \cos i = \sin 85^0 \cos 5.2^0 = 0.9921$

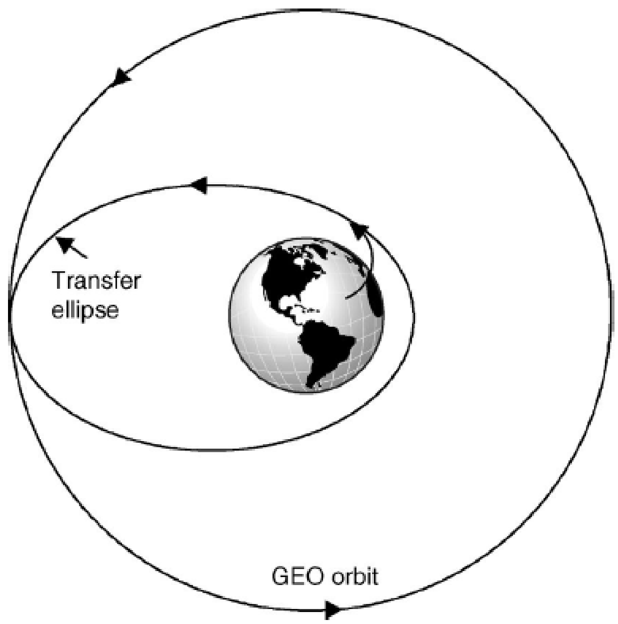
This gives $i = 7.2^0$

LAUNCH SEQUENCE

There are two broad categories of launch sequence, one that is employed by **expendable launch vehicles** and the other that is employed by a **reusable launch vehicle**.

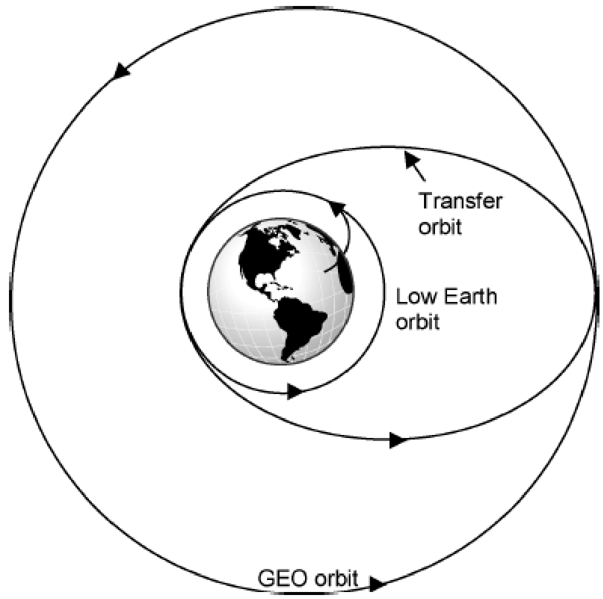
Irrespective of whether a satellite is launched by a reusable launch vehicle or an expendable vehicle, the satellite heading for a geostationary orbit is *first* placed in a **transfer orbit**. The transfer orbit is elliptical in shape with its perigee at an altitude between 200 km and 300 km and its apogee at the geostationary altitude.

In some cases, the launch vehicle injects the satellite directly into a transfer orbit of this type. Following this, an apogee manoeuvre circularizes the orbit at the geostationary altitude. The last step is then to correct the orbit for its inclination. This type of launch sequence is illustrated in following figure.



Possible geostationary satellite launch sequence

In the second case, the satellite is first injected into a low Earth circular orbit. In the second step, the low Earth circular orbit is transformed into an elliptical transfer orbit with a perigee manoeuvre. Circularization of the transfer orbit and then correction of the orbit inclination follow this. This type of sequence is illustrated in following figure.



Another possible geostationary satellite launch sequence

LAUNCH VEHICLES

Launch vehicles are rocket powered vehicles that are used to carry payloads from the Earth's surface into outer space either into an orbit around Earth or to some other destination in outer space. The payload can be an unmanned spacecraft or a space probe or an artificial satellite. The overall launch system includes the launch vehicle, the launch pad and other infrastructure. Launch vehicles are broadly classified as expendable and reusable launch vehicles.

Expendable launch vehicles (ELV)

Expendable launch vehicles (ELV) are designed to be used *only once* and their components are not recovered after launch. They mostly comprise multi-stage rockets and the job of each stage is to provide the desired orbital manoeuvre. As the job of the stage is completed, the different stages of the rocket are expended. The process goes on until the satellite is placed into the desired trajectory. The number of rocket stages can be as many as five.

It may be mentioned here that some launch vehicles used to put the satellite into small low Earth orbits can comprise a single rocket stage. In addition to the rocket stages, launch vehicles also comprise boosters that are used to aid the rockets during main orbital manoeuvres or to provide small orbital corrections.

Reusable launch vehicles (RLV)

Reusable launch vehicles (RLV) are designed to be recovered intact and used again for subsequent launches. It is generally used for human spaceflight missions.

On the basis of launch capacity, which is the most common method of classification, launch vehicles are classified as heavy lift launch vehicles (HLLVs), large launch vehicles (LLVs), medium launch vehicles (MLVs) and small launch vehicles (SLVs).

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 and so on. 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. 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 east--west location needs to be maintained to prevent radio frequency (RF) interference from neighbouring satellites.
- The north-south orientation has to be maintained to have proper satellite inclination.

Following factors result 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 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.
- The equatorial radius of the Earth is not constant.
- The average density of Earth is not uniform.

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.

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 gravitational pulls of Earth, sun and moon have negligible effect of the satellites orbiting in LEO orbits, where the effect of atmospheric drag is more predominant.

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 (T_A) is given by following equation:

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

The attitude and orbit control system maintains the satellite's position and its orientation and keeps the antenna pointed correctly in the desired direction (bore-sighted to the centre of the coverage area of the satellite). 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-of-plane changes).

ORBITAL EFFECTS ON SATELLITE'S PERFORMANCE

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, 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 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 following 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. v_T is the component of the satellite transmitter velocity vector directed towards the Earth station receiver v_P is the phase velocity of light in free space (3 x 10^8 m/s).

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. The Earth station antenna will receive signals from the satellite as well as the microwave radiation emitted by the sun. 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.

GEOSYNCHRONOUS VS. GEOSTATIONARY ORBITS

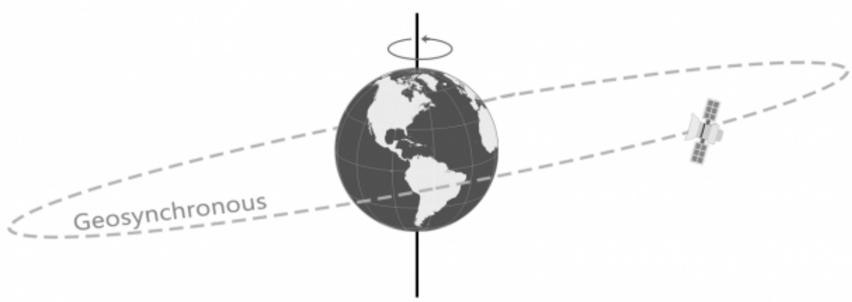
Geosynchronous Orbit

About 35,786 kilometres above the Earth's surface, satellites are in geostationary orbit. From the center of the Earth, this is approximately 42,164 kilometres. This distance puts it in the high Earth orbit category.

At any inclination, a geosynchronous orbit synchronizes with the rotation of the Earth. More specifically, the time it takes for the Earth to rotate on its axis is 23 hours, 56 minutes, and 4.09 seconds, which is the same as a satellite in a geosynchronous orbit.

If you are an observer on the ground, you would see the satellite as if it's in a fixed position without movement.

This makes geosynchronous satellites particularly useful for telecommunications and other remote sensing applications.

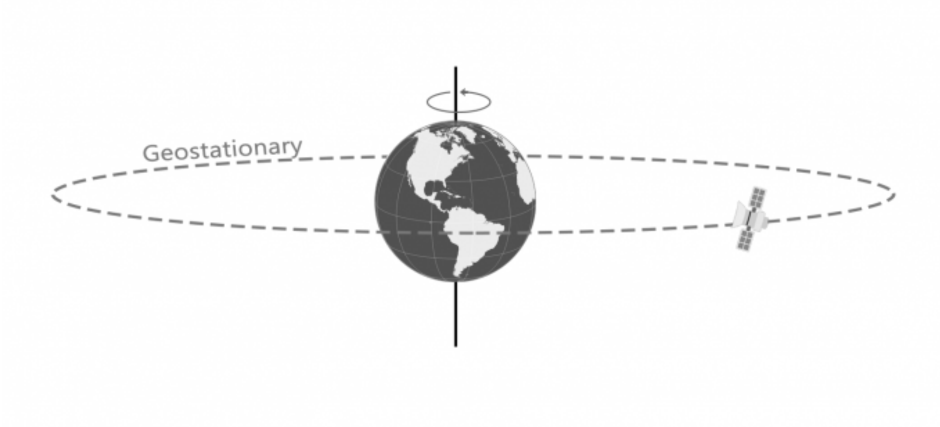


Geostationary Orbit

While geosynchronous satellites can have any inclination, the key difference to geostationary orbit is the fact that they lie on the same plane as the equator.

Geostationary orbits fall in the same category as geosynchronous orbits, but it's parked over the equator. This one special quality makes it unique from geosynchronous orbits.

Weather monitoring satellites are in geostationary orbits because they have a constant view of the same area.



While the geostationary orbit lies on the same plane as the equator, the geosynchronous satellites have a different inclination.

This is the key difference between the two types of orbits.

OPTICAL FIBRE VS. SATELLITE COMMUNICATION

Optic Fiber Communication

Optic Fiber communication transmits information by sending pulses of light (using laser) through an optic fiber. The low signal loss in optic fibers and high data rate of transmission systems, allow signals with high data rates (exceeding several Gbps) to travel over long distances (more than 100 km) without a need of repeater or amplifier. Moreover, using wavelength division multiplexing (WDM) allows a single fiber to carry multiple signals (upto 10 different signals) of multi-Gbps transmissions. Optic Fiber communication offers extremely high bandwidth, immunity to electromagnetic interference, non-existent delays and immunity from interception by external means.

These advancements in optic fiber communication has resulted in *decrease* of satellite communications for several types of communications. For instance, transmission between fixed locations or point-to-point communications, where large bandwidths are required (such as transoceanic telephone systems) are made through optic fiber instead of using satellite communication. Optic Fiber communication is also used to transmit telephone signals, Internet communication, LAN (Gigabit LAN) and cable television signals.

Satellite Communication

Satellite communications use artificial satellites as relays between a transmitter and a receiver at different locations on Earth. Satellite systems allow users to bypass typical carrier offices and to broadcast information to multiple locations. Communications satellites are used for radio, TV, telephone, Internet, military and other applications. There are more than 2,000 satellites around Earth’s orbit, being used for communication by both government and private organizations.

Communication Satellites are LOS (line-of-sight) microwave systems with a repeater. These satellites rotate around the earth with the speed of earth and are known as geostationary satellites. The limitations of antenna size also limits focusing capability making the coverage for a single satellite transmitter very large. This makes satellite communication ideal for TV and radio services as the signal has to flow from a single point to many points in a single direction. The large distance of satellites from the earth (about 22,300 miles) results in delays which adversely effects two-way communication like mobile conversations. Low earth orbit satellites can be used for two-way mobile communication because less power is required to reach those satellites.

Comparison of Optic Fiber and Satellite Communication

- **Bandwidth and data rates:** Optic Fiber supports higher bandwidth and data rates as compared to satellite.
- **Mobility:** Optic Fiber cannot be used in mobile applications and is suitable for fixed locations. Satellite communication is suitable for mobile applications.
- **Reliability:** Fiber Optic communication is more reliable than satellite.
- **Terrain:** Fiber optic is more suitable for urban areas and plains where digging / laying is easier. Satellite communication is suitable for remote areas and rough terrains like mountainous areas.
- **Delay**: Optic fiber has minimum or no delays making is suitable for real time applications. Satellite communication has an inherent propagation delay.
- **Interference**: Optic fiber has less or no Electromagnetic Interference EMI whereas Satellite communication has high EMI.
- **Coverage**: Satellites are suitable for providing point to multi-point services with large coverage like TV and radio.
- **Cost**: (i) Initial Cost: Depends on the size of network and whether the user wants to deploy complete network or part of it and lease the rest (ii) Recurring Cost: Satellite has higher recurring cost than optic fiber communication.

Active and Passive Satellite Communication

The requirement for satellite communication is obvious because we need to transmit the signal too far and wide places, where the curvature of the Earth interferes.

This obstacle is hurdled by setting communication satellites to dispatch the signals beyond the curvature in space. Satellite communication employs two types of artificial satellites to transmit the signals: active satellites and passive satellites.

ACTIVE AND PASSIVE SATELLITE

Active Satellite

In the active satellites, which amplify and retransmit the signal from the earth have several advantages over the passive satellites.

The advantages of active satellites are:

- Require lower power earth station
- Less costly
- Not open to random use
- Directly controlled by operators from the ground.

Disadvantages of active satellites are:

- Disruption of service due to a failure of electronics components onboard the satellites.
- A requirement of onboard power supply.
- A requirement of larger and powerful rockets to launch heavier satellites in orbit.

Passive Satellite

The principle of communication by passive satellite is based on the properties of scattering of electromagnetic waves from different surface areas. Thus an electromagnetic wave incident on a passive satellite is scattered back towards the earth and a receiving station can receive the scattered wave. The passive satellites used in the *early years* of satellite communications were both artificial as well as natural.

Although passive satellites were simple, the communications between two distant places were successfully demonstrated only after overcoming many technical problems. The large attenuation of the signal while travelling the large distance between the transmitter and the receiver via the satellite was one of the most serious problems.

The disadvantages of passive satellites for communications are:

- Earth Stations required high power (10 kW) to transmit signals strong enough to produce an adequate return echo.
- Large Earth Stations with tracking facilities were expensive.
- Communications via the Moon is limited by simultaneous visibility of the Moon by both the transmit and the receive stations along with the larger distance required to be covered compared to that of closer to earth satellite.
- A global system would have required a large number of passive satellites accessed randomly by different users.
- Control of satellites not possible from the ground.

Difference between Active and Passive Satellite

- Active satellites are a complicated structure having processing equipment called Transponder. A passive satellite only reflects received signals back to earth.
- Examples: An active satellite act as a repeater. A passive satellite can be a natural satellite or an artificial satellite for example moon is the natural satellite of earth.

ASSIGNMENT

Q.1. (AMIE W18, 15 marks): State and explain all the three Kepler's laws of planetary motion.

Q.2. (AMIE S19, 12 marks): Explain the Kepler's first laws for satellite orbits and launching methods.

Q.3. (AMIE W19, S20, 5 marks): Explain the Kepler's law with neat sketch and essential equations.

Q.4. (AMIE S17, 10 marks): Explain the interpretation of Kepler's laws. Draw a neat diagram of the path of satellite in the orbital plane and derive an expression for the orbital period T as where "a" and "b" are the semi-major and semi minor axes of the orbital ellipse?

Q.5. (AMIE W16, 17, 19, S18, 20, 5 marks): What are the orbital parameters required to determine a satellite orbit? Name and explain them.

Q.6. (AMIE S19, 4 marks): What do you mean by inclined orbits.

Q.7. (AMIE S15, 6 marks): Differentiate geostationary orbits from geosynchronous orbits. Explain the difference between them.

Q.8. (AMIE W17, 6 marks): Prove that the altitude of a geostationary satellite is nearly 36,000 km.

Q.9. (AMIE S20, 5 marks): Discuss the basic principles of a satellite.

Q.10. (AMIE W18, 6 marks): Derive an expression for centrifugal force on satellite.

Q.11. (AMIE W18, S19, 5 marks): Derive expression for orbital velocity of satellite.

Q.12. (AMIE W19, 5 marks): How does environment affect the working of a satellite?

Q.13. (AMIE S19, W19, 10 marks): Write a short note on different types of satellites based on orbits.

Q.14. (AMIE S17, 10 marks): With a neat sketch explain launching mechanism. Explain wideband receiver operation with neat diagram.

Q.15. (AMIE W15, 16, 9 marks): Compare the performance characteristics of LEO, MEO and GEO satellites. Show how communication satellite is affected by the orbital effects like Doppler shift and solar eclipse.

Q.16. (AMIE W16, 6 marks): Explain the basic difference between active and passive satellite. Whether a passive satellite can be used for communication?

Q.17. (AMIE W17, S18, 4 marks): Comparisons are sometimes made between satellite and optical fibre communication systems. State briefly the areas of applications. Where each system is best suited?

Q.18. (AMIE W19, 4 marks): What do you mean by Orbital Perturbations?

Q.19. (AMIE W15, 6 marks): A geostationary satellite is moving in an equatorial circular orbit at a height of 35,800 km from the earth's surface. Estimate the theoretical maximum coverage angle, if the radius of earth's is 6364 km.

Answer: 17.362^0

Q.20. (AMIE S16, 6 marks): A satellite is moving in a elliptical transfer orbit with apogee and perigee at a distance of 39,342 km and 615 km. If the radius of earth is 6371 km, determine velocity of the satellite at any point in its orbit.

Answer: 1520.91 m/s

Q.21. (AMIE W16, 8 marks): Two satellites are moving in different elliptical orbits with the same perigee but different apogee distances. The semi major axis of the two orbits are 16,000 km and 24,000 km. Determine the orbital period of satellite 2, it the orbital period of satellite 1 is 600 min.

Answer: 616.70 minutes

Q.22. (AMIE W17, 4 marks): Two satellites are moving in an elliptical orbit with same perigee but different apogee distances. Satellite 1 is having an orbital period of 5 hours and semi major axis, 20,000 km. The orbital period of satellite 2 is 2 hrs 50 minutes. Determine the semi major axis of satellite.

Answer: 10164 km

Q.23. (AMIE S18, 5 marks): A satellite is orbiting in a geostationary orbit of radius 41,500 km. Find the velocity and time of orbit. What is change in velocity if radius reduces to 36000 km.

Given g_0 = 398600 km^3/s^2 = gravitational coefficient

Answer: 0.229 km/s

Q.24. (AMIE W18, 6 marks): Calculate the radius of a circular orbit for which the period is 1 day.

Answer: 42241 km

Q.25. (AMIE W18, 6 marks): A satellite is orbiting in the equatorial plane with a period from perigee to perigee of 12 h. Given that the eccentricity is 0.002, calculate the semi major axis. The earth's equatorial radius is 6378.1414 km.

Answer: 26610 km

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