Satellite Communication

Code:17EC755

TEXT BOOK: SATELLITE COMMUNICATION

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Vision and Mission of the Department:

Vision:

Conquering technical frontiers in the field of electronics & communications .

Mission:

- To achieve and foster excellence in core Electronics and Communication engineering with focus on the hardware, simulation and design.
- To pursue Research, development and consultancy to achieve self sustenance.
- To create benchmark standards in Electronics and Communication by active involvement of all stakeholders.

Program Educational Objectives (PEOs) of ECE Program.

- Graduates from the **Electronics and Communication Engineering** are expected to attain or achieve the following Program Educational Objectives within a few years of graduation.

1) To demonstrate the ability to apply fundamental concepts obtained from mathematics, basic sciences and engineering subjects into Electronics and Communication domain.
2) To impart technical and analytical skills for the design and development of innovative electronic systems.
3) To develop design procedures for given specifications &operating conditions and test the same.
4) To provide opportunities for students to work in multidisciplinary fields encouraging team work to come up with new technologies for the benefit of the society.
5) To cultivate skills which helps in building leadership qualities, effective communication skills and ethics needed for a successful professional career and life-long learning.

SATELLITE COMMUNICATION
B.E., VII Semester, Electronics & Communication Engineering
[As per Choice Based Credit System (CBCS) Scheme]

Course Code	17EC755	**CIE Marks**	40
Number of Lecture Hours/Week	03	**SEE Marks**	60
Total Number of Lecture Hours	40 (8 Hours / Module)	**Exam Hours**	03

CREDITS – 03

Course Objectives: This course will enable students to

- Understand the basic principle of satellite orbits and trajectories.
- Study of electronic systems associated with a satellite and the earth station.
- Understand the various technologies associated with the satellite communication.
- Focus on a communication satellite and the national satellite system.
- Study of satellite applications focusing various domains services such as remote sensing, weather forecasting and navigation.

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. **L1, L2**

Module-2

Satellite subsystem: Power supply subsystem, Attitude and Orbit control, Tracking, Telemetry and command subsystem, Payload.

Earth Station: Types of earth station, Architecture, Design considerations, Testing, Earth station Hardware, Satellite tracking. **L1, L2**

Module-3

Multiple Access Techniques: Introduction, FDMA (No derivation), SCPC Systems, MCPC Systems, TDMA, CDMA, SDMA.

Satellite Link Design Fundamentals: Transmission Equation, Satellite Link Parameters, Propagation considerations. **L1, L2, L3**

Module-4
Communication Satellites: Introduction, Related Applications, Frequency Bands, Payloads, Satellite Vs. Terrestrial Networks, Satellite Telephony, Satellite Television, Satellite radio, Regional satellite Systems, National Satellite Systems. **L1, L2**
Module-5
Remote Sensing Satellites: Classification of remote sensing systems, orbits, Payloads, Types of images: Image Classification, Interpretation, Applications. **Weather Forecasting Satellites**: Fundamentals, Images, Orbits, Payloads, Applications. **Navigation Satellites**: Development of Satellite Navigation Systems, GPS system, Applications. **L1, L2, L3**

Text Book:

Anil K. Maini, Varsha Agrawal, Satellite Communications, Wiley India Pvt. Ltd., 2015, ISBN: 978-81-265-2071-8.

Reference Books :

1. Dennis Roddy, Satellite Communications, 4th Edition, McGraw- Hill International edition, 2006
2. Timothy Pratt, Charles Bostian, Jeremy Allnutt, Satellite Communications, 2nd Edition, Wiley India Pvt. Ltd , 2017, ISBN: 978-81-265-0833-4

Course Outcomes: At the end of the course, the students will be able to:

- Describe the satellite orbits and its trajectories with the definitions of parameters associated with it.
- Describe the electronic hardware systems associated with the satellite subsystem and earth station.
- Describe the various applications of satellite with the focus on national satellite system.
- Compute the satellite link parameters under various propagation conditions with the illustration of multiple access techniques.

History

- It began with an article by Arthur C. Clarke published in October 1945 issue of Wireless World, which theoretically proposed the feasibility of establishing a communication satellite in a geostationary orbit.
- In his article , he discussed how a geostationary orbit satellite would look static to an observer on Earth within the satellite's coverage, thus providing an uninterrupted communication service across the globe.

Satellite – definitions

- An artificial body placed in orbit round the earth or another planet in order to collect information or for communication.
- Satellites can be natural also: Eg.: Moon

Note :

For an Electronics engineer 'Satellite is a repeater in Space'

Introduction to Satellite Communication

- Applications of satellites are in the fields of

- ❑ **Remote sensing and Earth observation.**
- ❑ **Atmospheric monitoring and space exploration**
- ❑ **Defense related applications, which include secure communications, navigation, spying and so on.**
- ❑ **Data communication, mobile communication, etc.**
- ❑ **Internet.**

Introduction to Satellite Communication - 2

- Satellite is often referred to as an ‘orbiting radio star’. Orbiting radio stars assist ships and aircraft to navigate safely in all weather conditions.
- Even some categories of medium to long range ballistic and cruise missiles need the assistance of a satellite to hit their intended targets precisely.
- The satellite-based global positioning system (GPS) is used as an aid to navigate safely and securely in unknown territories.
- Earth observation and remote sensing satellites give information about the weather, ocean conditions, volcanic eruptions, earthquakes, pollution and health of agricultural crops and forests.

Introduction to Satellite Communication - 3

- Another type of Satellites keeps watch on <u>military activity around the world</u> and helps to some extent in enforcing or policing arms control agreements.
- Satellites are <u>also used to investigate all planets</u>. These satellites for <u>astrophysical applications have giant telescopes</u> on board and have sent data that has led to many new discoveries, throwing new light on the universe.
- United States, the United Kingdom, France, Japan, Germany, Russia and major developing countries like India have a full fledged and heavily funded space programme, managed by organizations with massive scientific and technical manpower and infrastructure.

Satellite ?

- A satellite in general is any natural or artificial body moving around a celestial body such as planets and stars.
- In this topic artificial satellites are discussed, and these are orbiting the planet earth. These satellites are put into the desired orbit and have payloads depending upon the intended application.
- Geostationary satellite

A communication satellite (Figure 1.1) is a kind of repeater station that receives signals from ground, processes them and then retransmits them back to Earth.

Figure 1.1 Communication satellite

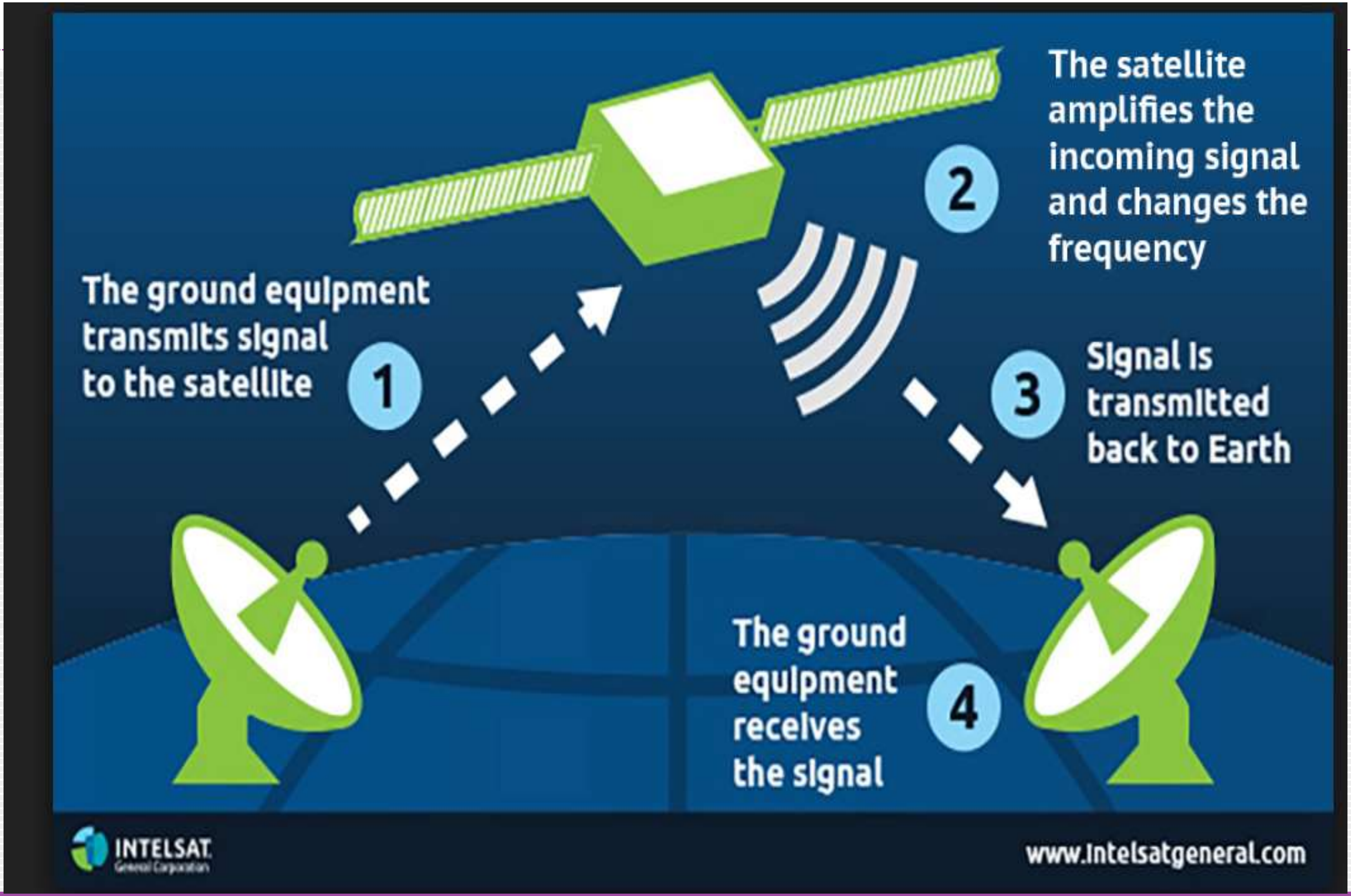
a photographer that takes pictures of regions of inAn Earth observation satellite (Figure 1.2) is terest during its periodic motion.

Figure 1.2 Earth observation satellite

Why Satellite Communication ?

- No geographical boundaries
- Important areas of application such as Remote Sensing , Earth observation , GPS , weather forecast, Internet
- Long distance communication
- Distance insensitive system

Satellite communication System



Satellite Orbits and Trajectories

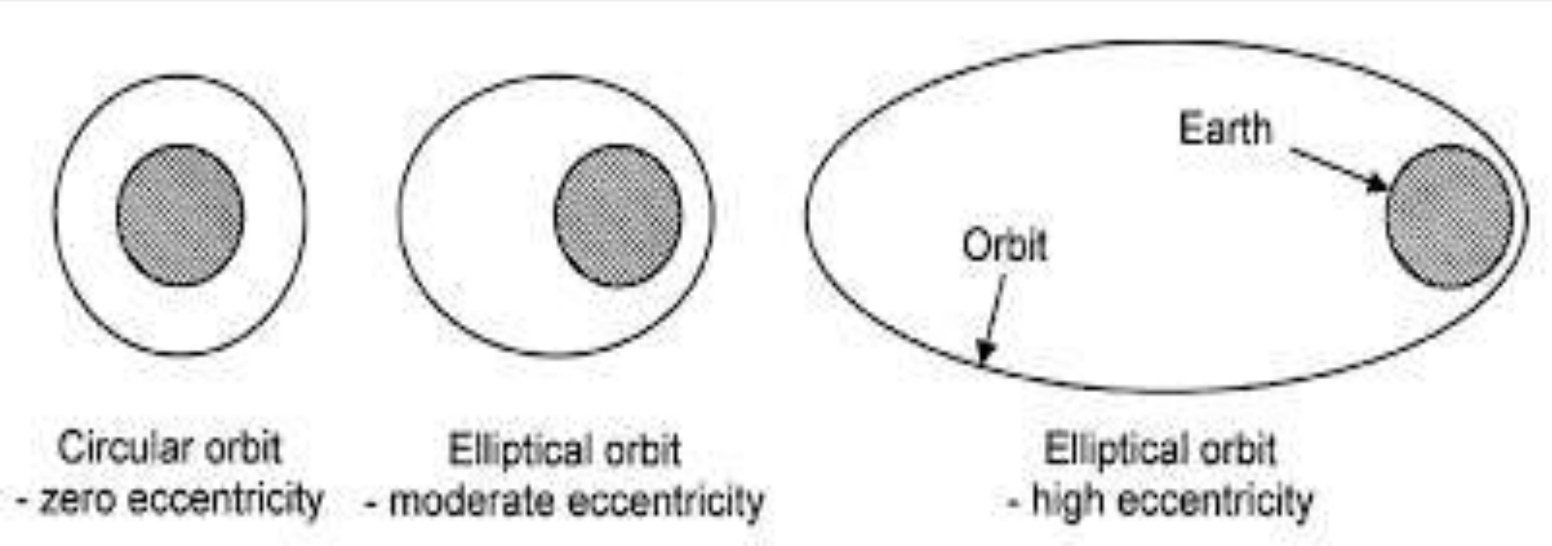
- An understanding of the orbital dynamics is important to address the following issues.

1. types of orbits and their suitability for a given application,
2. Orbit stabilization,
3. Orbit correction and station keeping,
4. launch requirements and typical launch trajectories for various orbits,
5. Earth coverage and so on.

Definition of an Orbit and a Trajectory

- Trajectory: This is a path traced by a moving body.
- Orbit : This is a trajectory that is periodically repeated.
- 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.

General types of orbit:



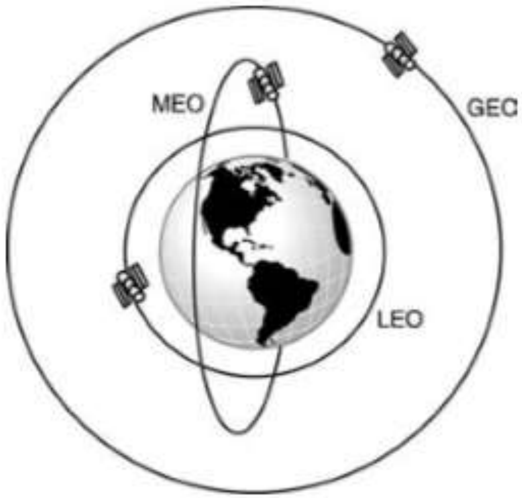
Basic principles of orbiting Satellites

The motion of natural and Artificial Satellites are governed by two forces:

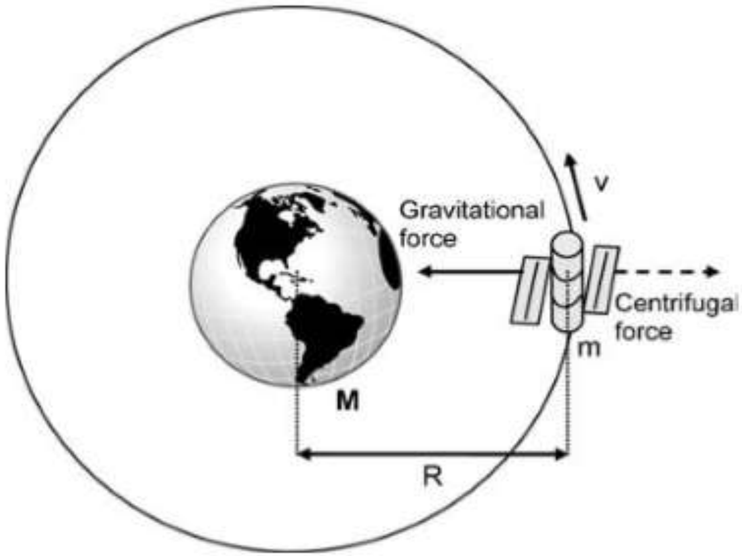
- Centripetal Force- Satellite is directed towards the centre of the earth because of gravitational force of attraction of earth.
- Centrifugal force – this is the force that acts out wards from the centre of the earth

Hence an orbiting Satellite , exerts a centrifugal force, but maintains circular motion due to centripetal force.

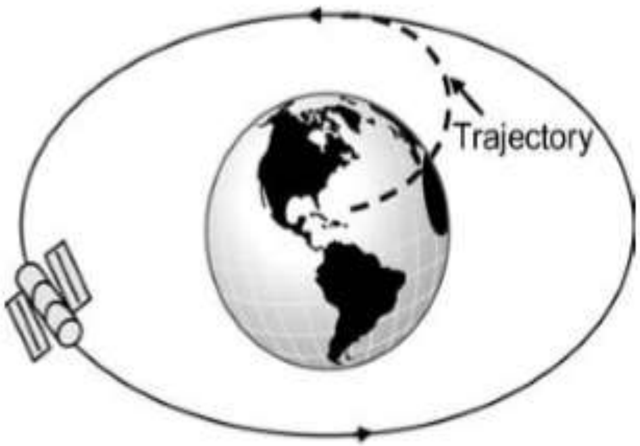
Centripetal force hence transforms straight line motion of Satellite to circular motion



Orbital motion of Satellite:
Satellites revolving around Earth.



Gravitational force and the centrifugal force acting on bodies orbiting Earth.



Orbit and Trajectory: Path followed by a rocket on its way during satellite launch

Low Earth Orbit (LEO)
LEOs will orbit at a distance of **500 to 1000 miles** above the earth's surface. (satellite phones and GPS)

Medium Earth Orbit
MEOs will orbit at distances of about **8000 miles** from earth's surface.

Geostationary Orbit (GEO)
These Satellites are placed at 35,786kms above the Earth's surface of the earth.

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

$$F = \frac{G m_1 m_2}{r^2} \tag{2.1}$$

where

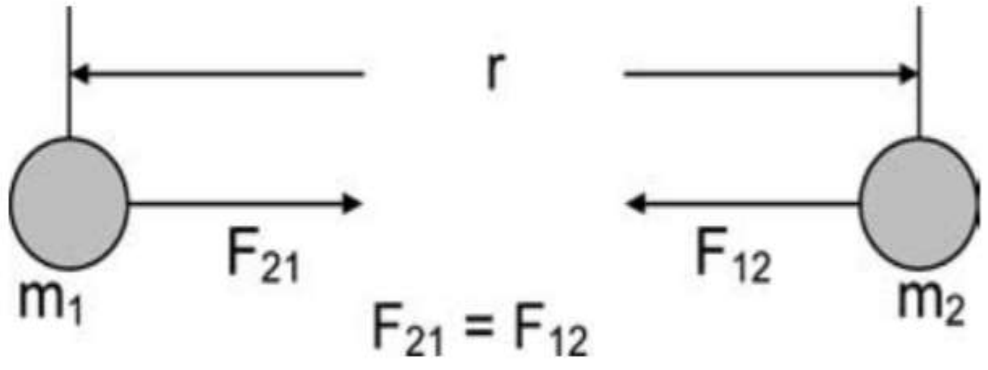
m_1, m_2 = masses of the two particles

r = distance between the two particles

G = gravitational constant = 6.67×10^{-11} $m^3/kg\,s^2$

Newton's Law of Gravitation contd..

- The force with which the particle with mass **m_1** attracts the particle with mass **m_2** **equals** the force with which particle with mass **m_2** attracts the particle with mass **m_1**. The forces are equal in magnitude but opposite in direction.
- The acceleration, which is force per unit mass, experienced by the two particles, would depend upon their masses. A larger mass experiences lesser acceleration.



Newton's law of gravitation

Newton's Second Law of motion

- As per 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 $\mathbf{v^2/r}$.
- If the mass of satellite is m, it would experience a reaction force of $\mathbf{mv^2/r}$. 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.

- If the satellite orbited Earth with a uniform velocity v, which would be the case when the satellite orbit is a **circular one**, then equating the two forces mentioned above would lead to an expression for the orbital velocity v as follows.

Newton's Second Law of motion contd..

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

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

where

$m_1 =$ mass of Earth
$m_2 =$ mass of the satellite
$\mu = Gm_1 = 3.986\,013 \times 10^5\ \mathrm{km^3/s^2} = 3.986\,013 \times 10^{14}\ \mathrm{N\,m^2/kg}$

The orbital period in such a case can be computed from

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

Newton's Second Law of motion contd..

- For an elliptical orbit:

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

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

Kepler's Laws

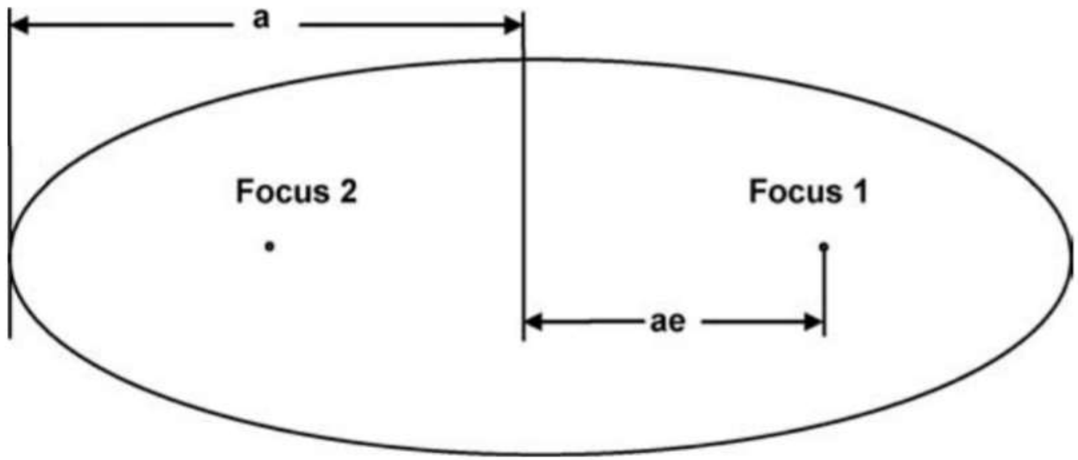
- Three empirical expressions of **Kepler's Laws** explained planetary motion.
- These laws are valid for the motion of natural and artificial satellites around Earth or for any body revolving around another body.

Kepler's First Law

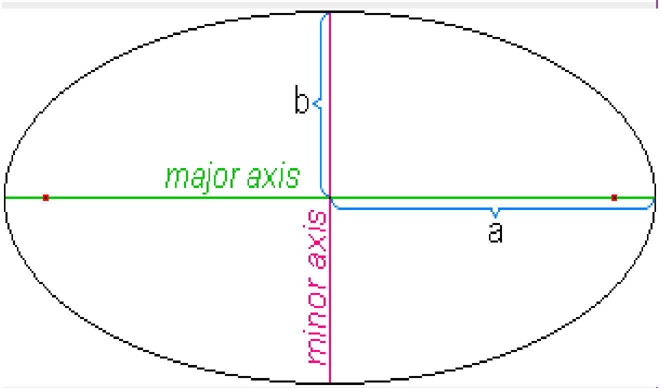
- The orbit of a satellite around Earth is elliptical with the centre of the Earth lying at one of the foci of the ellipse. (shown in the figure in the next slide).
- The elliptical orbit is characterized by its semi-major axis a and eccentricity e.
- Eccentricity is the ratio of the distance between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

Kepler's Laws

- **Kepler's First Law:**



Kepler's first law



For any elliptical motion, the law of conservation of energy is valid at all points on the orbit. The law of conservation of energy states that energy can neither be created nor destroyed; it can only be transformed from one form to another. 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 $-Gm_1m_2/(2a)$, where

m_1 = mass of Earth
m_2 = mass of the satellite
a = semi-major axis of the orbit

Kepler's First Law Contd....

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

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

$$\text{Potential energy} = -\frac{Gm_1 m_2}{r}$$

Therefore,

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

$$v^2 = Gm_1\left(\frac{2}{r} - \frac{1}{a}\right)$$

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

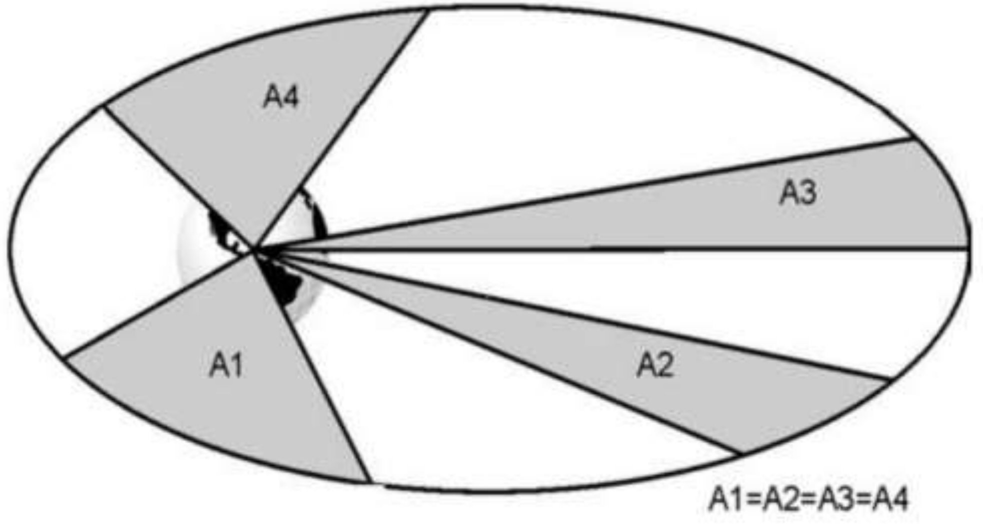
Kepler's second Law

- 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.(shown in the figure); 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

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

- where m is the mass of the satellite. Hence, 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.

Kepler’s second Law



Kepler’s second law

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 $mr^2\omega = (m\omega r)\ (r) = mvr$ remains constant. Here v is the component of the satellite’s velocity v *in the direction perpendicular* to the radius vector and is expressed as $v \cos \gamma$, *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 *(Shown in the Figure).*

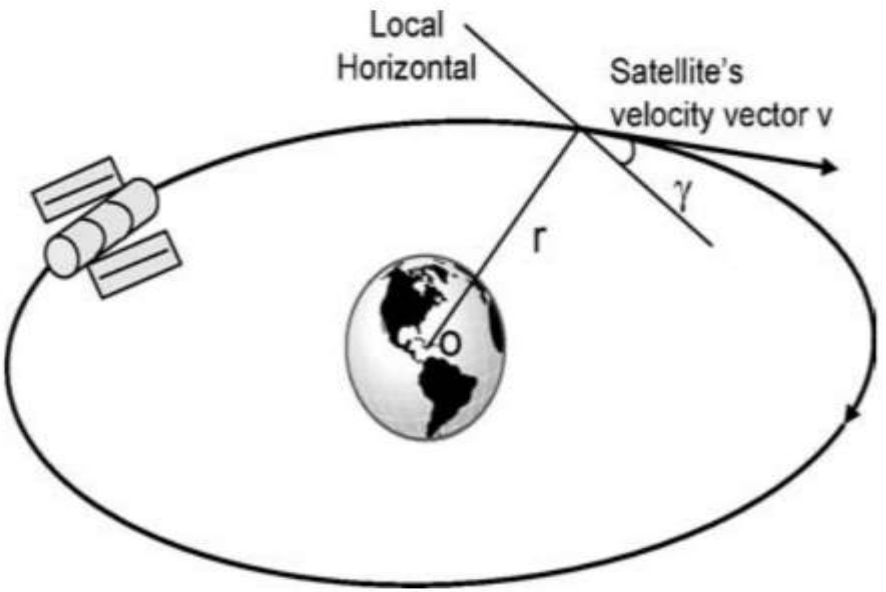
Satellite's position at any given time

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

- The law states that " The square of the time period is proportional to the cube of the semi major axis of the elliptical orbit".
- This Law is also called as "Law of Periods".

- The expression for the time period can be derived as follows. A circular orbit with radius *r is* assumed. Here 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.

Kepler's Third Law(Law of periods)

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

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

Satellite Communication frequency bands

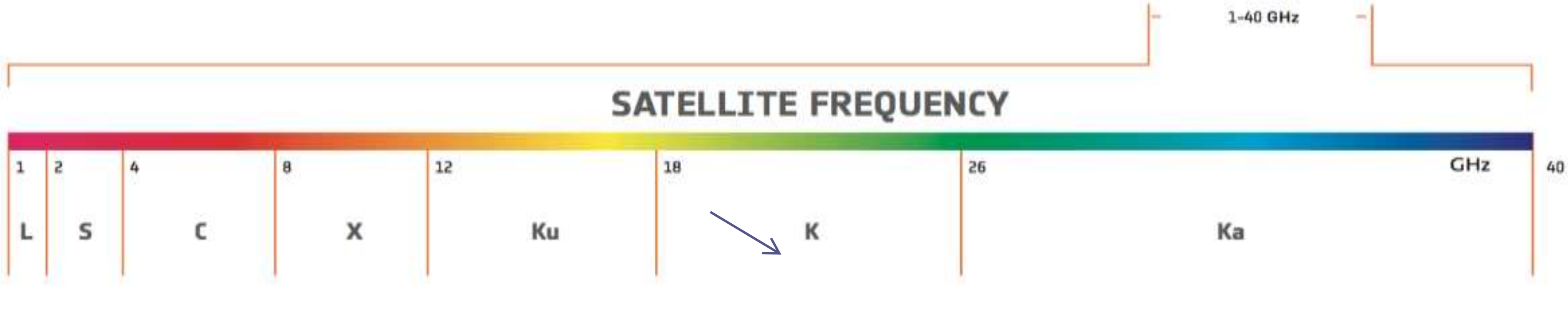
- Frequency Bands

ITU(UN)
for ICT

Frequency Band Designations

Frequency range, (GHz)	Band designation
0.1–0.3	VHF
0.3–1.0	UHF
1.0–2.0	L
2.0–4.0	S
4.0–8.0	C
8.0–12.0	X
12.0–18.0	Ku
18.0–27.0	K
27.0–40.0	Ka
40.0–75	V
75–110	W
110–300	mm
300–3000	µm

Satellite Communication frequency bands Cont….



Water absorption at 22GHz

L-band (1-2GHz) -Global Positioning System (GPS) carriers and also satellite mobile phones, such as Iridium (USA); Inmarsat (UK) providing communications at sea, land and air; WorldSpace satellite radio.

S-band (2–4 GHz):

Weather radar, surface ship radar, and some communications satellites, especially those of NASA for communication with International space station (ISS) and Space Shuttle.

- **C-band (4–8 GHz)**

  Primarily used for satellite communications, for full-time satellite TV networks or raw satellite feeds.

- **X-band (8–12 GHz)**

  Primarily used by the military. Used in radar applications. X-band radar frequency sub-bands are used in civil, military and government institutions for weather monitoring, air traffic control, maritime vessel traffic control, defence tracking and vehicle speed detection for law enforcement.

- **Ku-band (12–18 GHz)**

  Used for satellite communications. In Europe, Ku-band downlink is used from 10.7 GHz to 12.75 GHz for direct broadcast satellite services, such as Astra.

- **Ka-band (26–40 GHz)**

  Communications satellites, uplink in either the 27.5 GHz and 31 GHz bands, and high-resolution, close-range targeting radars on military aircraft.

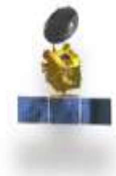
INDIAN SPACECRAFT

Communication Satellites

Supports telecommunication, television broadcasting, satellite news gathering, societal applications, weather forecasting, disaster warning and Search and Rescue operation services.

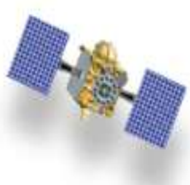
Earth Observation Satellites

The largest civilian remote sensing satellite constellation in the world - thematic series of satellites supporting multitude of applications in the areas of land and water resources; cartography; and ocean & atmosphere.



Scientific Spacecraft

Spacecraft for research in areas like astronomy, astrophysics, planetary and earth sciences, atmospheric sciences and theoretical physics.

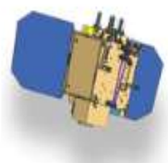


Navigation Satellites

Satellites for navigation services to meet the emerging demands of the Civil Aviation requirements and to meet the user requirements of the positioning, navigation and timing based on the independent satellite navigation system.

Experimental Satellites

A host of small satellites mainly for the experimental purposes. These experiments include Remote Sensing, Atmospheric Studies, Payload Development, Orbit Controls, recovery technology etc..



Small Satellites

Sub 500 kg class satellites - a platform for stand-alone payloads for earth imaging and science missions within a quick turn around time.

Student Satellites

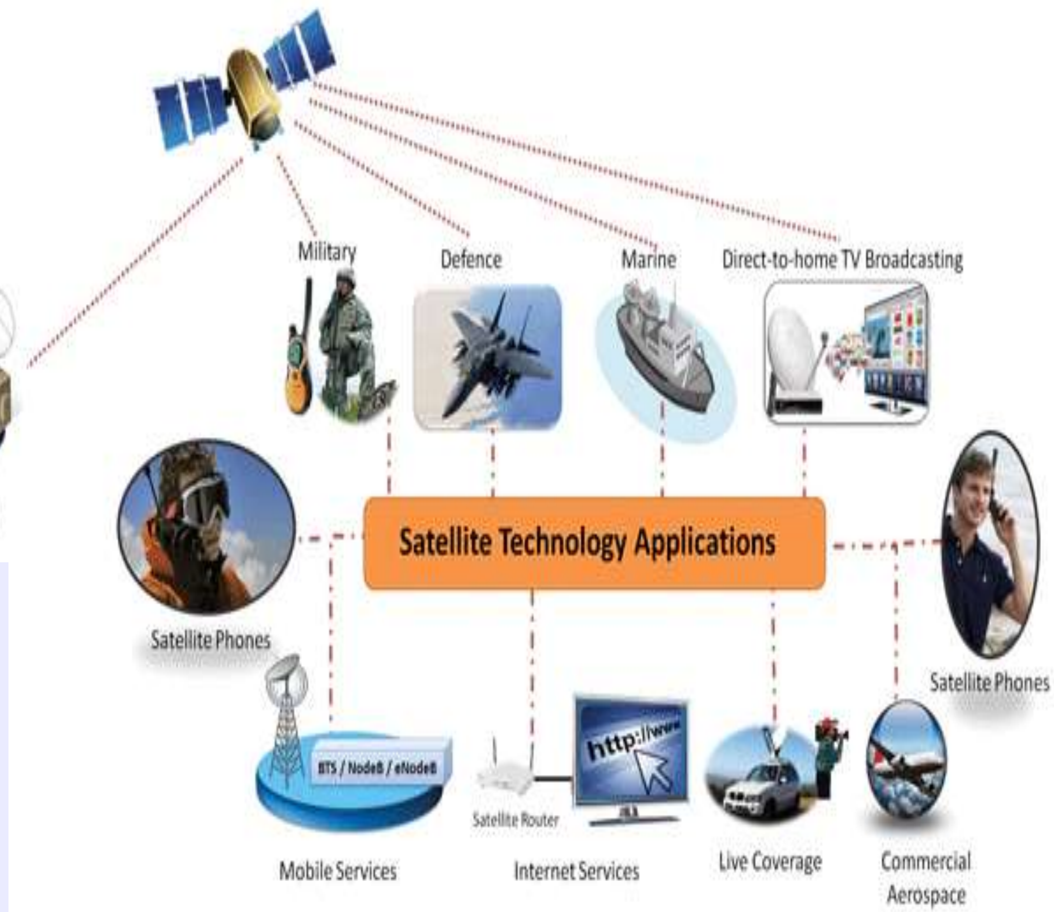
ISRO's Student Satellite programme is envisaged to encourage various Universities and Institutions for the development of Nano/Pico Satellites.

Satellites by India

List of Communication Satellites

	Launch Date	Launch Mass	Launch Vehicle	Application
GSAT-6A	Mar 29, 2018		GSLV-F08/GSAT-6A Mission	Communication
GSAT-17	Jun 29, 2017	3477 kg	Ariane-5 VA-238	Communication
GSAT-19	Jun 05, 2017	3136 Kg	GSLV Mk III-D1/GSAT-19 Mission	Communication
GSAT-9	May 05, 2017	2230 kg	GSLV-F09 / GSAT-9	Communication
GSAT-18	Oct 06, 2016	3404 kg	Ariane-5 VA-231	Communication
GSAT-15	Nov 11, 2015	3164 kg	Ariane-5 VA-227	Communication, Navigation
GSAT-6	Aug 27, 2015	2117 kg	GSLV-D6	Communication
GSAT-16	Dec 07, 2014	3181.6 kg	Ariane-5 VA-221	Communication
GSAT-14	Jan 05, 2014	1982 kg	GSLV-D5/GSAT-14	Communication
GSAT-7	Aug 30, 2013	2650 kg	Ariane-5 VA-215	Communication
GSAT-10	Sep 29, 2012	3400 kg	Ariane-5 VA-209	Communication, Navigation

Applications of Satellite Communication



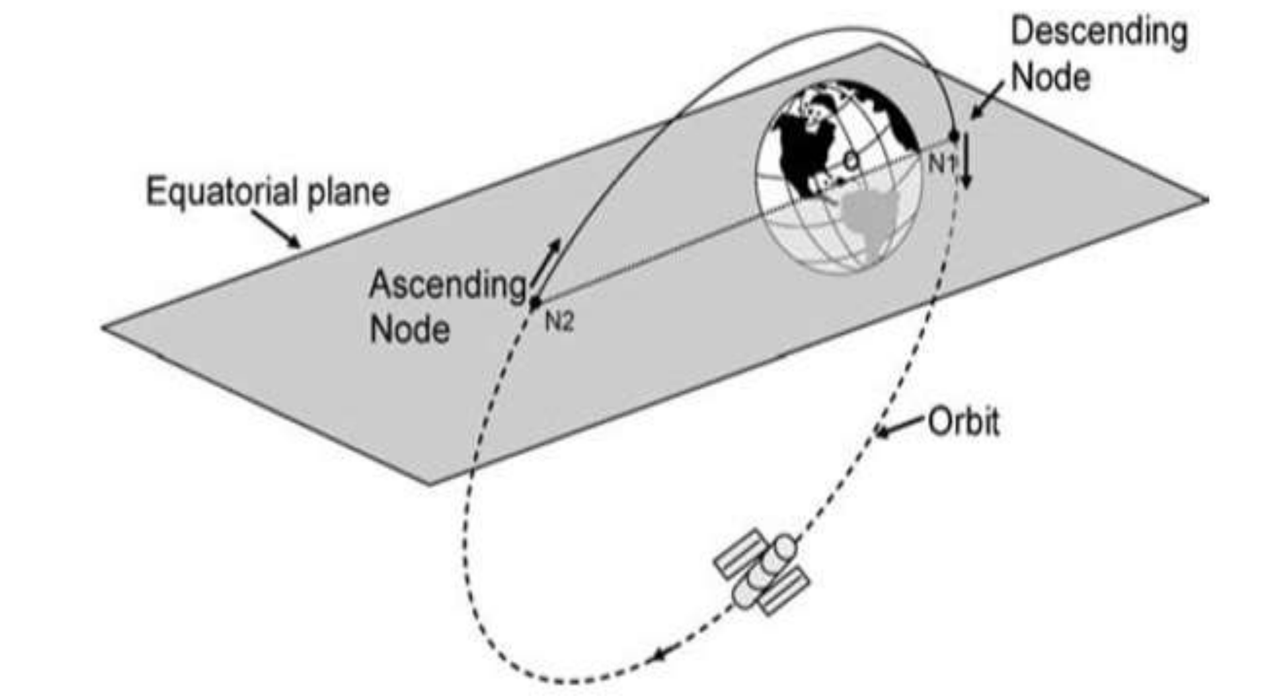
Orbital parameters

- The satellite orbit, which is elliptical, is characterized by a number of parameters.

- These not only include the geometrical parameters of the orbit but also parameters that define its orientation with respect to Earth.

1. Ascending and descending nodes
2. Equinoxes
3. Solstices
4. Apogee
5. Perigee
6. Eccentricity
7. Semi-major axis
8. Right ascension of the ascending node
9. Inclination
10. Argument of the perigee
11. True anomaly of the satellite
12. Angles defining the direction of the satellite

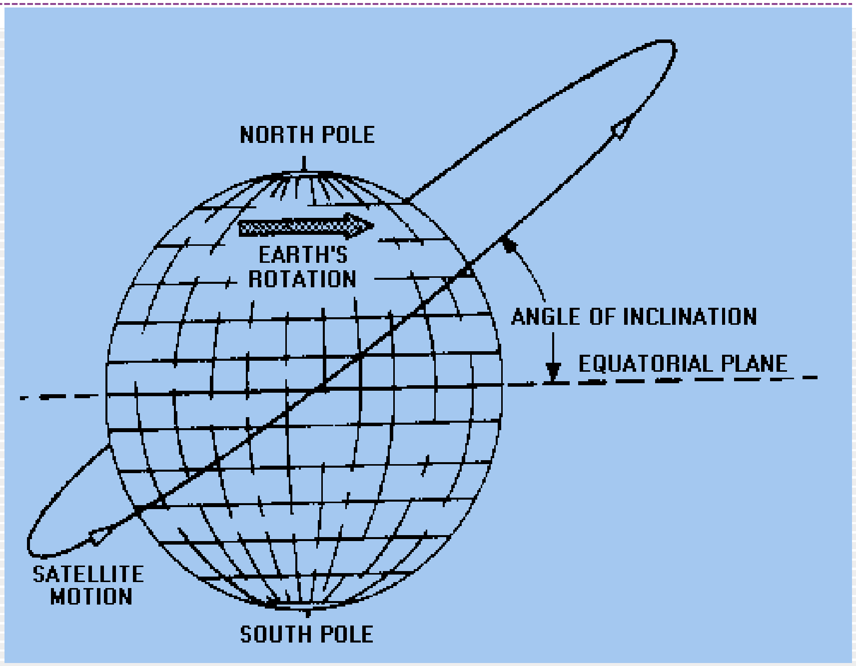
Ascending and Descending Nodes



The satellite orbit cuts the equatorial plane at two points:
N1 (descending node) and N2(ascending node)

Descending node (N1), where the satellite passes from the northern hemisphere to the southern hemisphere, and the second, called the Ascending node(N2), where the satellite passes from the southern hemisphere to the northern hemisphere.

Equinoxes



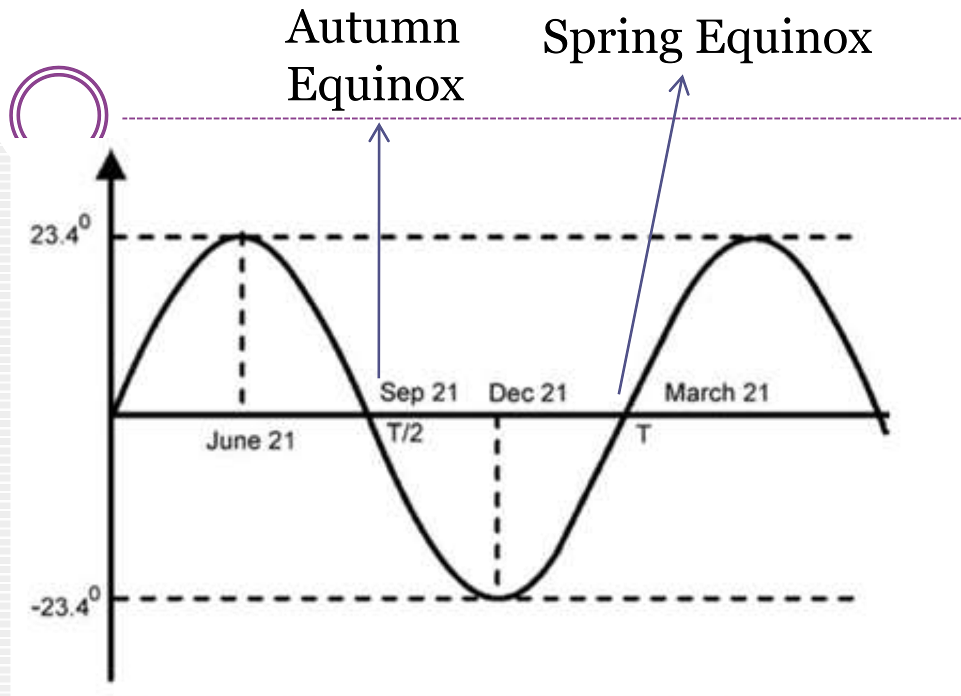


Fig.: Yearly variation of angular inclination of Earth with the sun.

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.

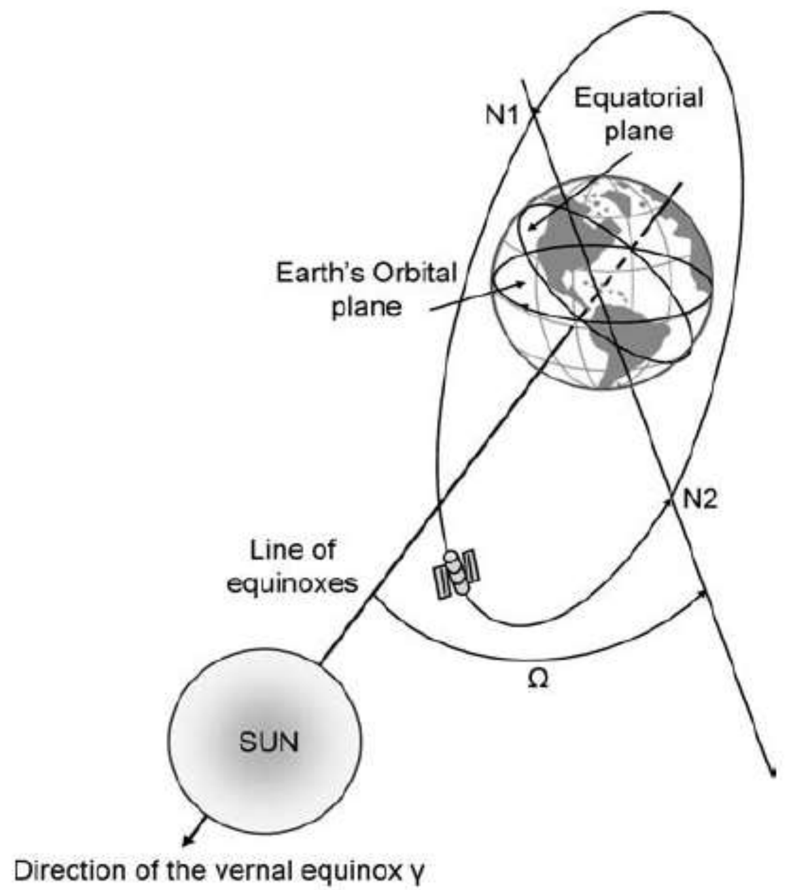


Figure 2.11 Vernal equinox

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 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.

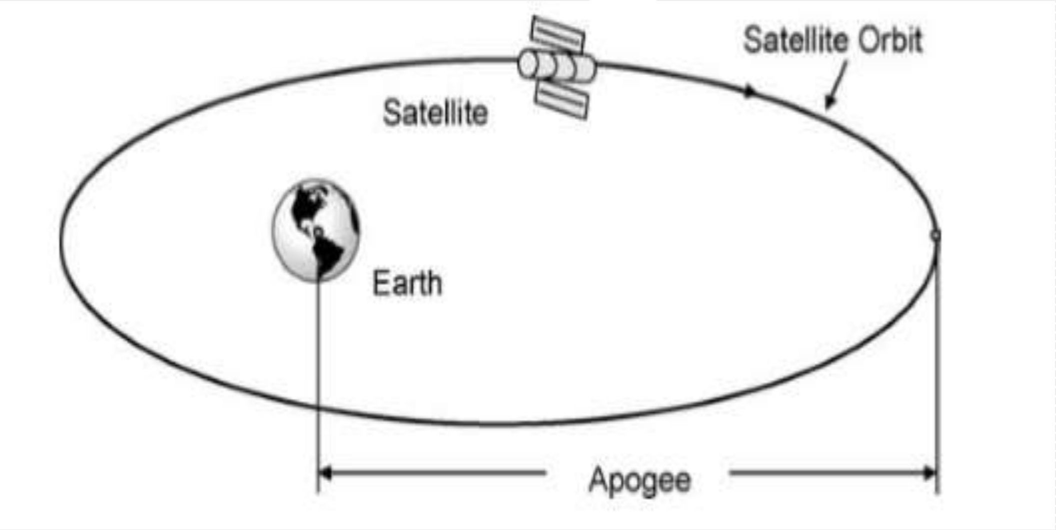
Equinoxes contd..

- 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)

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.
- Also indicates longest day and shortest day of the year.
- Note: This phenomenon creates our changing seasons, because the hemisphere facing the sun receives longer and more powerful exposure to sunlight.

Apogee



Apogee: Apogee is the point on the satellite orbit that is at the farthest distance from the centre of the Earth (Figure). The apogee distance can be computed from the known values of the orbit eccentricity *e and the semi-major axis a from*

Apogee distance = $a\,(1 + e)$

Apogee contd…

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

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

where

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

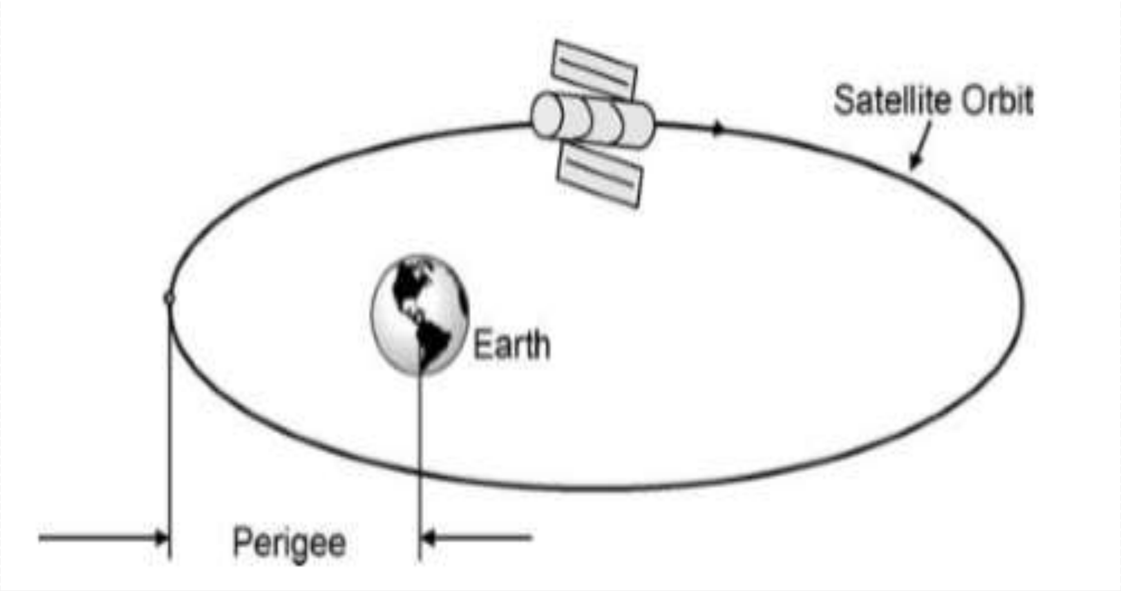
with V being the velocity of the satellite at a distance d from the centre of the Earth.

Perigee distance: P, Apogee distance: A

Perigee

Perigee is the point on the orbit that is nearest to the centre of the Earth (Figure). The perigee distance can be computed from the known values of orbit eccentricity *e and the semi-major axis a as*

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



Eccentricity

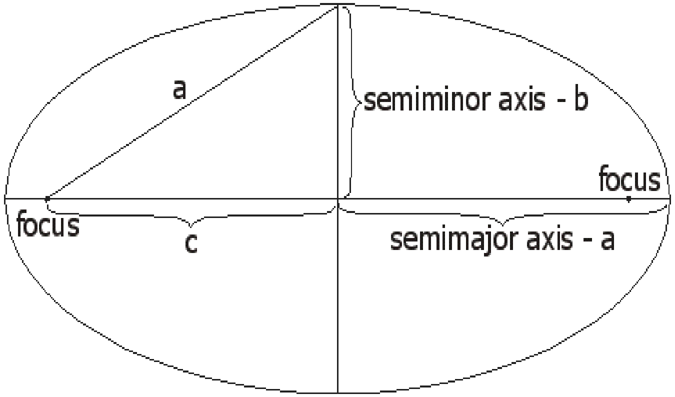
- The orbit eccentricity *e is the ratio of the distance between the centre of the* ellipse and the centre of the Earth to the semi-major axis of the ellipse. It can be computed from any of the following expressions:

$$e = \frac{A - P}{A + P}$$

$$e = \frac{A - P}{2a}$$

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

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



e = c/a ; c = ae; A+P = 2a
a = (A+P)/2 ; c = a - P

- The eccentricity of an ellipse is a measure of how nearly circular the ellipse is.

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

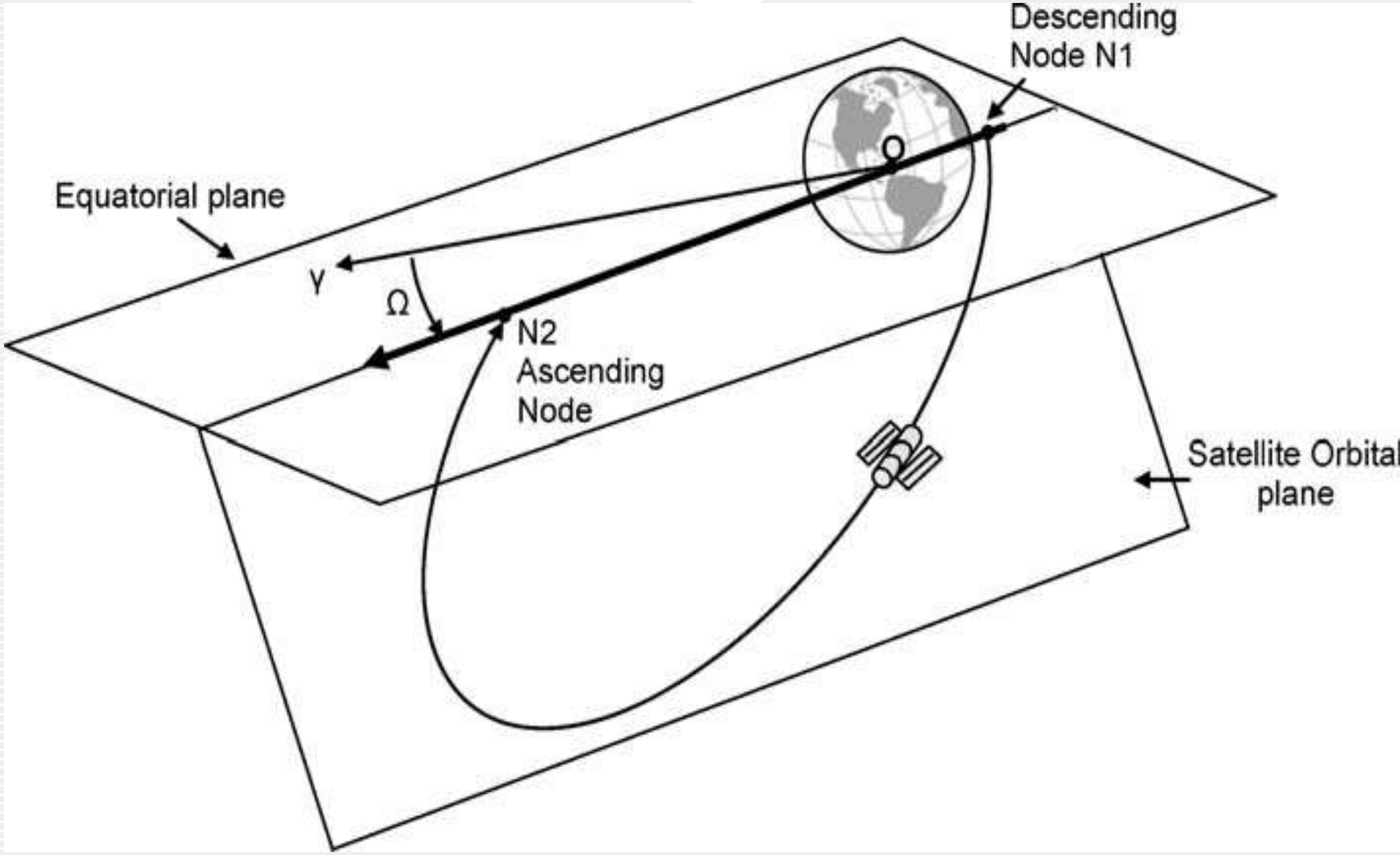
$$a = \frac{A + P}{2}$$

Because A + P = 2a

Right ascension of the ascending node

- The right ascension of the ascending node tells about the orientation of the line of nodes, which is the line joining the ascending and descending nodes, with respect to the direction of the vernal equinox.
- It is expressed as an angle *measured from the vernal equinox towards the line of nodes in the direction* of rotation of Earth (Figure 2.14).
- The angle could be anywhere from 0° to 360°.
- Ascending node (**Ω**) is important to ensure that the satellite orbits in the given plane. This can be achieved by choosing an appropriate injection time depending upon the longitude.

Figure :Right ascension of the ascending node



- **Figure 2.14: Right ascension of the ascending node**

Right ascension of the ascending node

- <u>Calculation of angle **Ω**</u>**:** This can be computed as the difference between two angles.
- One is the angle <u>*α between the direction of the vernal* equinox and the longitude of the injection point</u> and the other is the <u>angle *β between the* line of nodes and the longitude of the injection point</u>, as shown in Figure 2.15. Angle β can be computed from

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

where $\angle i$ is the orbit inclination and l is the latitude at the injection point.

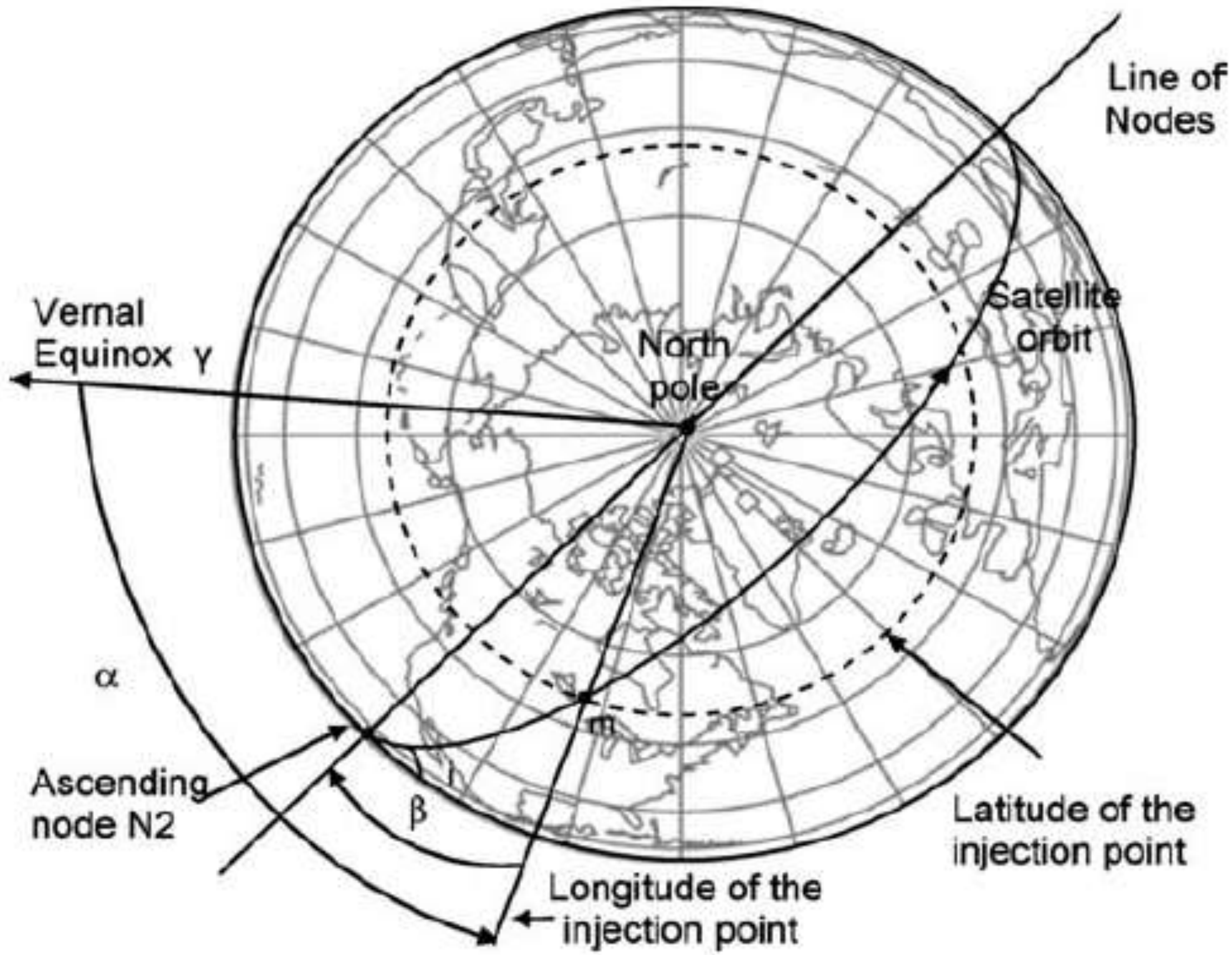


Figure 2.15 Computation of the right ascension of the ascending node

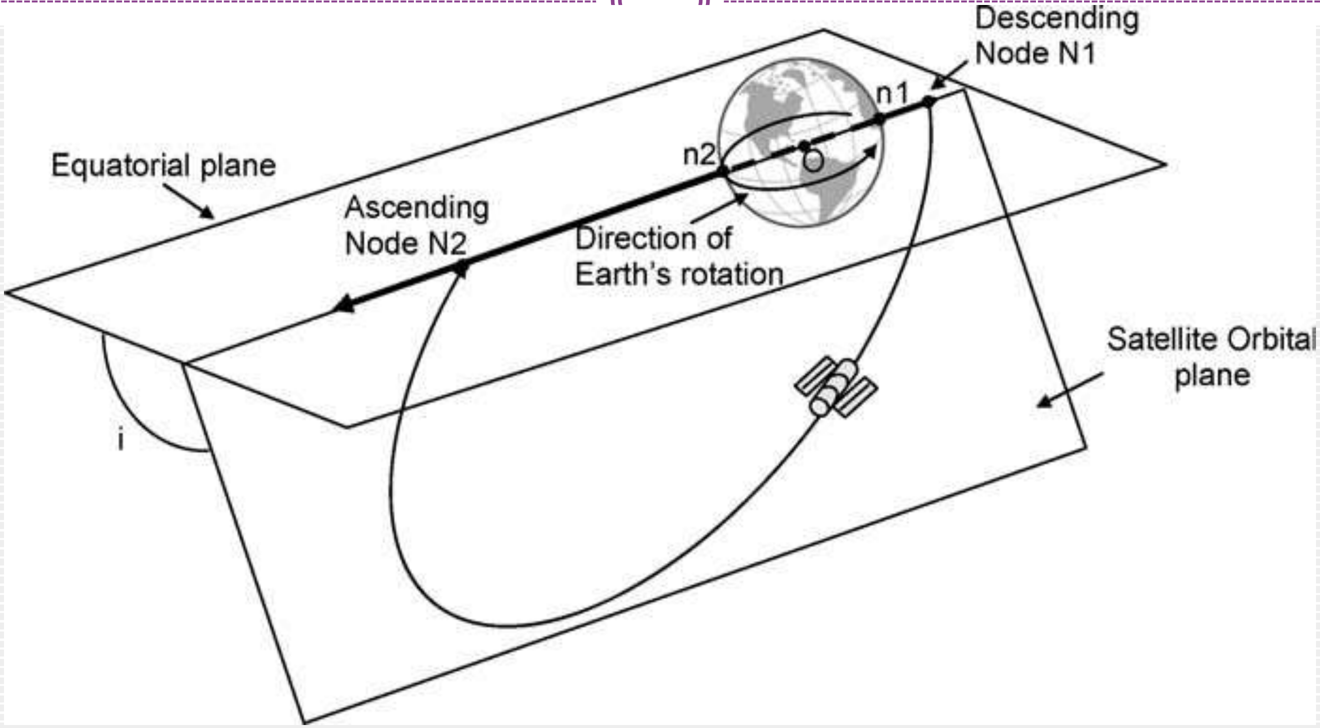
Inclination

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

➢ The line of nodes divides both the Earth's equatorial plane as well as the satellite's orbital plane into two halves.

➢ Inclination is measured as the angle from that half of the satellite's orbital plane containing the trajectory of the satellite from the descending node to the ascending node to that half of the Earth's equatorial plane containing the trajectory of a point on the equator from n1 to n2, where n1 and n2 are respectively the points vertically below the descending and ascending nodes (Figure in the next slide).

Fig.: Angle of Inclination

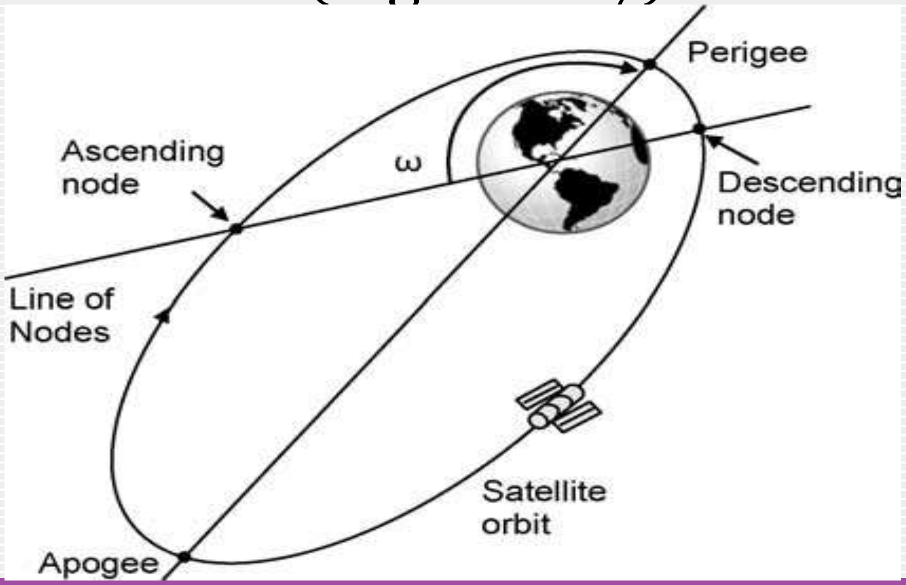


The inclination angle can be determined from the latitude *l at the injection point and the angle Az between* the projection of the satellite's velocity vector on the local horizontal and north. It is given by $\cos i = \sin Az \cos l$

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 (Figure 2.17).

Figure 2.17 : Argument of perigee



True anomaly of the satellite

- This parameter is used to indicate the <u>position of the satellite in its orbit.</u>
- This is done by defining an angle θ, *called the true anomaly of the satellite*, formed by the <u>line joining the perigee and the centre of the Earth</u> with the <u>line joining the satellite and the centre of the Earth</u> (Figure 2.18).

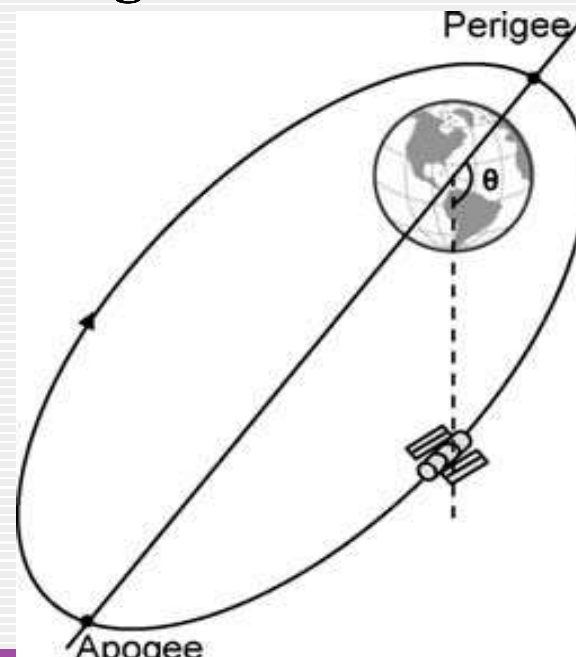


Figure 2.18:
True anomaly of a satellite

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 *Az between* the north and the projection of the satellite's velocity vector on the local horizontal (Figure 2.19).

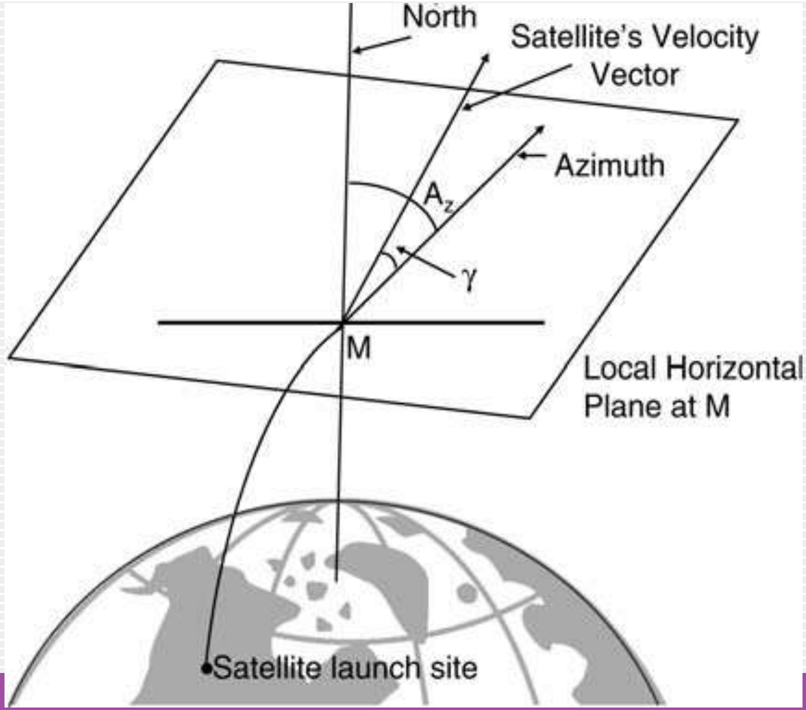


Figure 2.19: Angles defining the direction of the satellite

Injection Velocity and Resulting Satellite Trajectories

- The satellite trajectory directly depends on horizontal velocity with which a satellite is injected into space by the launch vehicle.
- The phenomenon is best explained in terms of the three cosmic velocities.
- General expression for Injection velocity at perigee

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

- Where A = *apogee distance; P = perigee distance; μ = GM = constant* = 3.986 013 × 10^5 km^3/s^2 = 3.986 013 ×10^{14} Nm^2/kg

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

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

If the injection velocity is equal to the first cosmic velocity, (first orbital velocity), the satellite follows a circular orbit (Figure 2.27)) and moves with a uniform velocity. (Above eqn.)

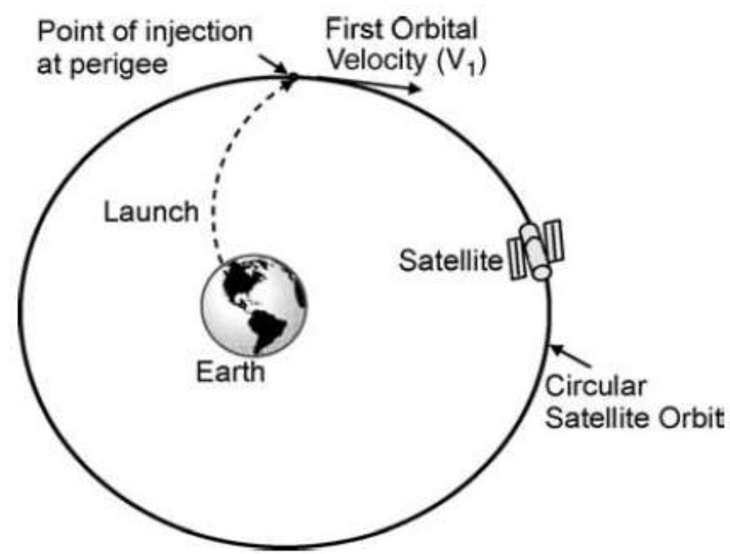


Figure 2.27
Satellite's path where the injection velocity is equal to the first orbital velocity

▪ If the injection velocity is less than the first cosmic velocity, the satellite follows a ballistic trajectory and falls back to Earth.

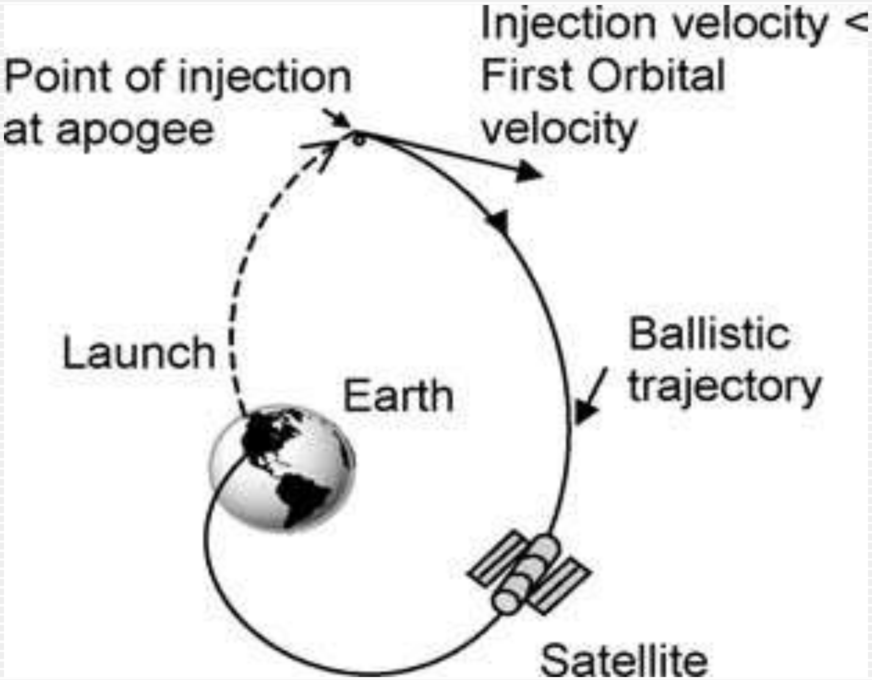


Figure 2.28: Satellite's path where the injection velocity is less than the first orbital velocity

▪ 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 higher the injection velocity, the greater is the apogee distance. Eccentricity is between 0 and 1.

The velocity v *at any other point in* the orbit distant d *from the centre of the Earth using*

$$V_p = \sqrt{\left[(2\mu/P) - \left(\frac{2\mu}{A+P}\right)\right]} = \frac{vd\ \cos\gamma}{P}$$

- When the injection velocity equals √(2μ/P), the apogee distance A becomes infinite and the orbit takes the shape of a parabola (Fig.2.29) and the orbit eccentricity is 1.
- This velocity √(2μ/P) is called the second cosmic velocity.
- 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

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

where V_t is the speed of Earth's revolution around the sun.

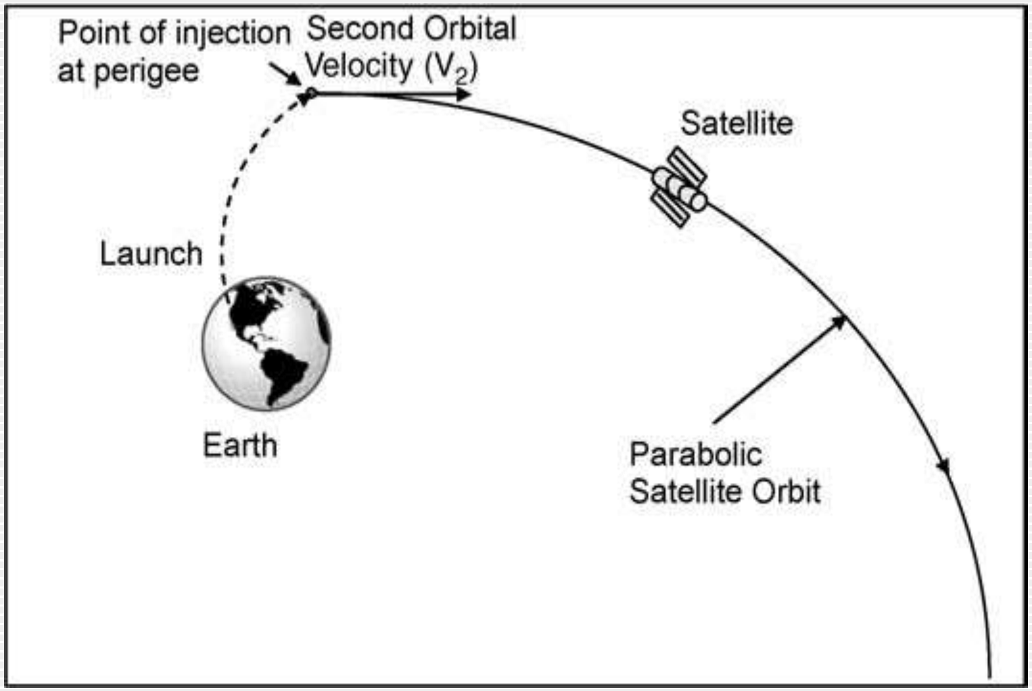


Figure 2.29: Satellite's path where the injection velocity is equal to the second orbital velocity.

Beyond the third cosmic velocity, there is a region of hyperbolic flights outside the solar system.

The greater the injection velocity from the first cosmic velocity, the greater is the apogee distance. This is evident from the generalized expression for the velocity of the satellite in elliptical orbits according to which

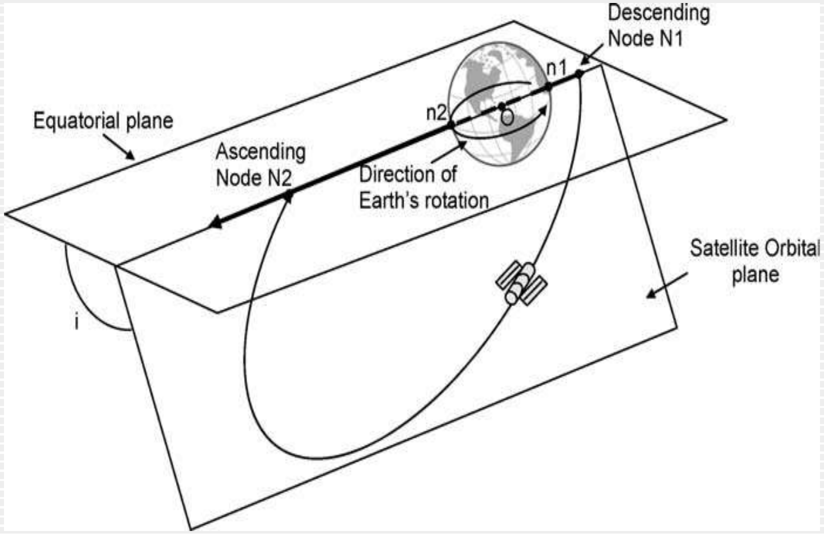
Vp = velocity at the perigee point

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

$$\left(\frac{v_2}{v_1}\right)^2 = \frac{1 + P/A_1}{1 + P/A_2}$$

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



1. 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.

- Based on these inclination, the orbits can be classified as
  - equatorial orbits,
  - polar orbits and
  - inclined orbits.
- **Equatorial orbit**: 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).

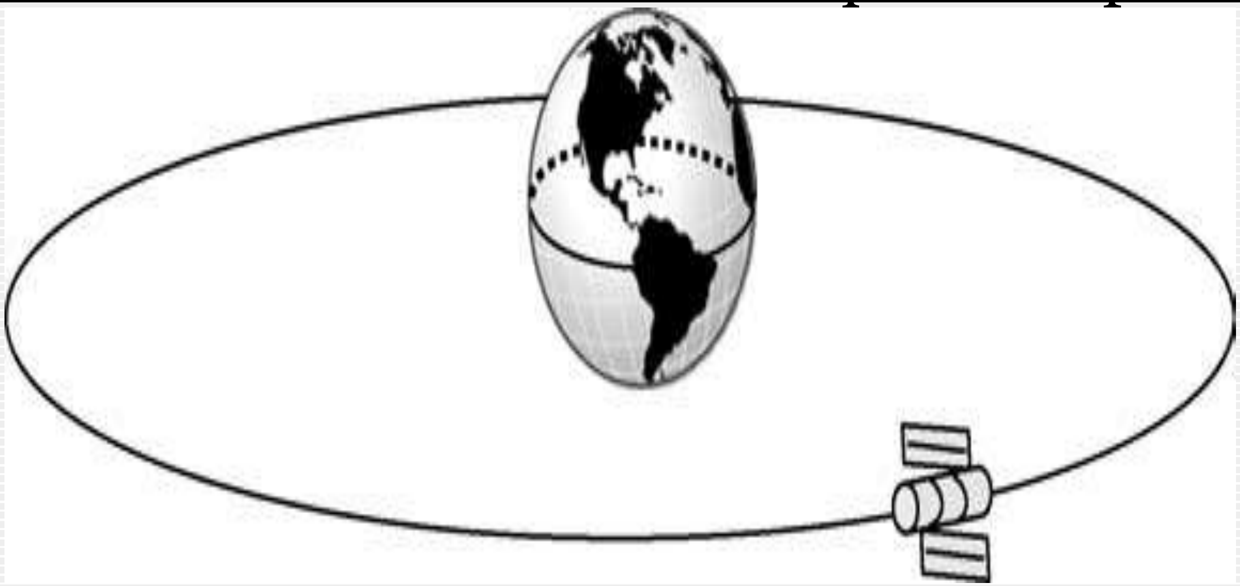


Figure 2.32 : Equatorial orbit

- 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**.

Figure 2.33: Polar orbit

- For inclinations between 0° and 90°, the satellite travels in the same direction as the direction of rotation of the Earth. The orbit in this case is referred to as a direct or **prograde orbit** (Figure 2.34).

- For inclinations between 90° and 180°, the satellite orbits in a direction opposite to the direction of rotation of the Earth and the orbit in this case is called a **retrograde orbit** (Figure 2.35).

Figure 2.34 : Prograde orbit **Figure 2.35: Retrograde orbit**

Eccentricity of the Orbit

- On the basis of eccentricity, the orbits are classified as elliptical and circular 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.

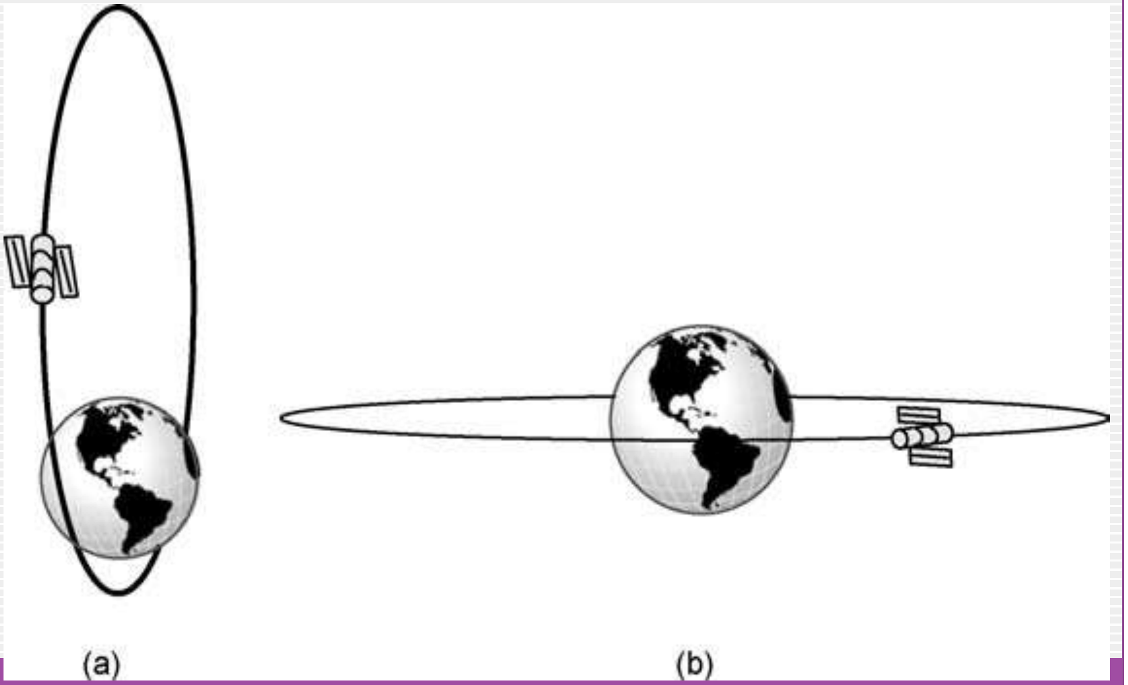


Figure 2.36 (a) Elliptical orbit and (b) circular orbit

- Example: the eccentricity of orbit of geostationary satellite INSAT-3B, an Indian satellite in the INSAT series providing communication and meteorological services, is 0.000 252 6.
- Eccentricity figures for orbits of GOES-9 and Meteosat-7 geostationary satellites, both offering weather forecasting services, are 0.000 423 3 and 0.000 252 6 respectively.

Molniya Orbit:

- Highly eccentric, inclined and elliptical orbits are used to cover higher latitudes, which are not covered by geostationary orbits. It is widely used by Russia and other countries of the former Soviet Union to provide communication services.

- Typical eccentricity and orbit inclination figures for the Molniya orbit are 0.75 and 65° respectively. The apogee and perigee points are about 40,000km and 400km respectively from the surface of the Earth.
- The Molniya orbit serves the purpose of a geosynchronous orbit for high latitude regions

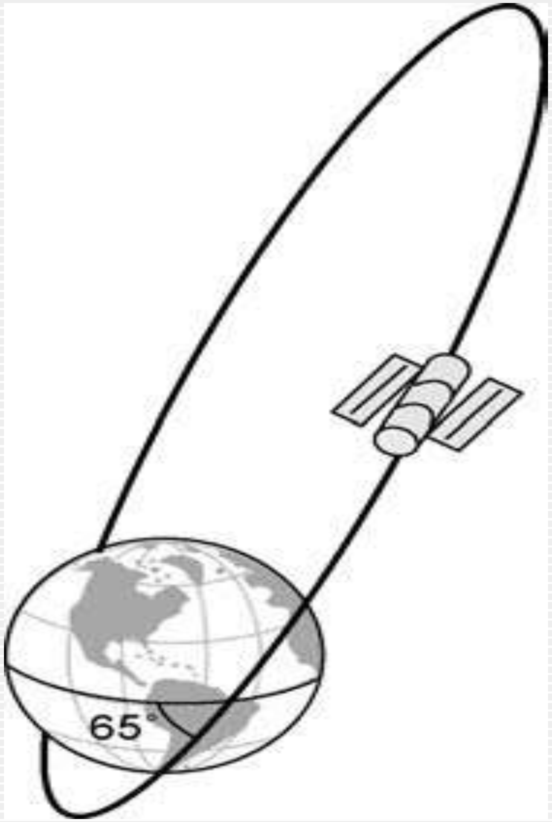


Figure 2.37 : Molniya orbit

Distance from Earth

- Satellites are 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), as shown in Figure 2.38.
- **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.

- The power required for signal transmission is also less, they are of small in physical size and are inexpensive to build.

- One important application of LEO satellites for communication is the project Iridium, which is a global communication system conceived by Motorola. Communications solutions ranging from satellite phones to broadband.

- A total of 66 satellites are arranged in a distributed architecture, with each satellite carrying 1/66 of the total system capacity.

- LEO satellites can be put to use are surveillance, weather forecasting, remote sensing and scientific studies.

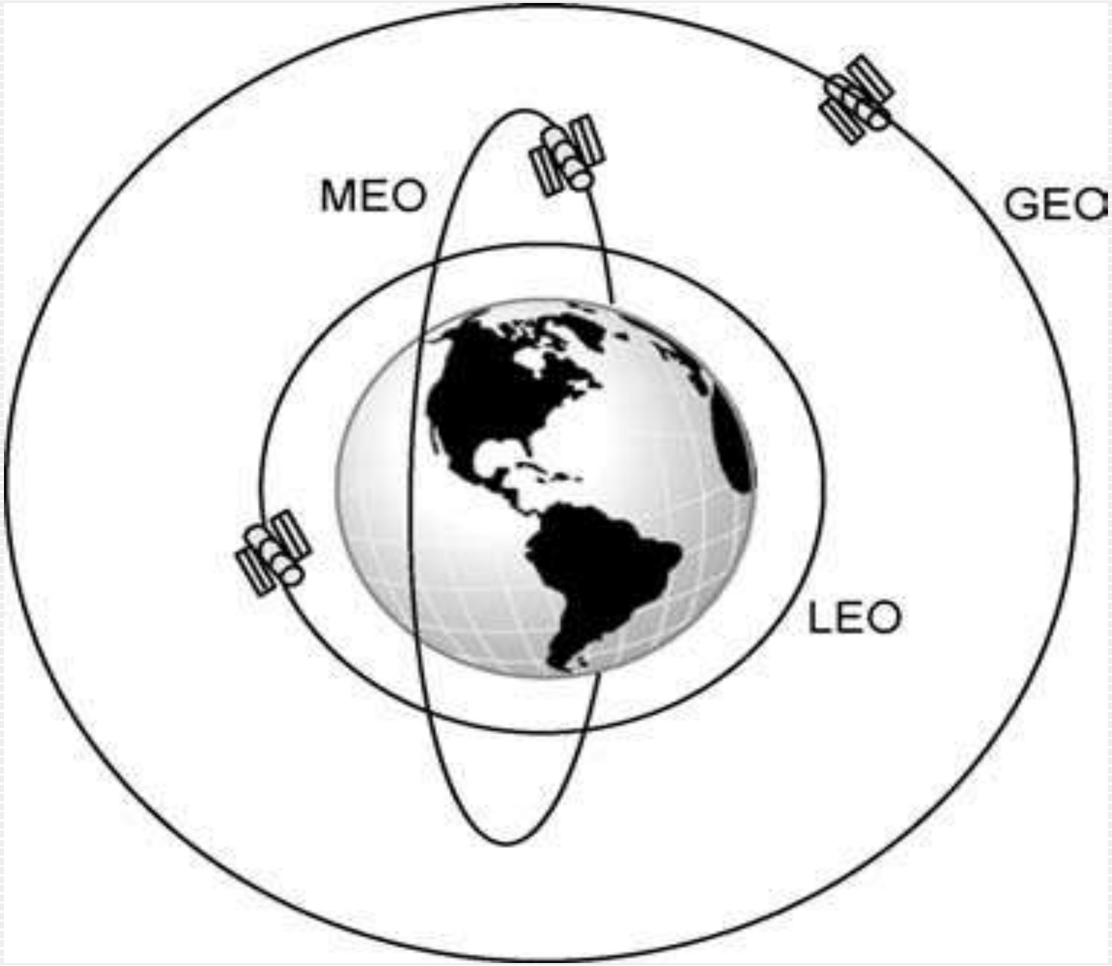


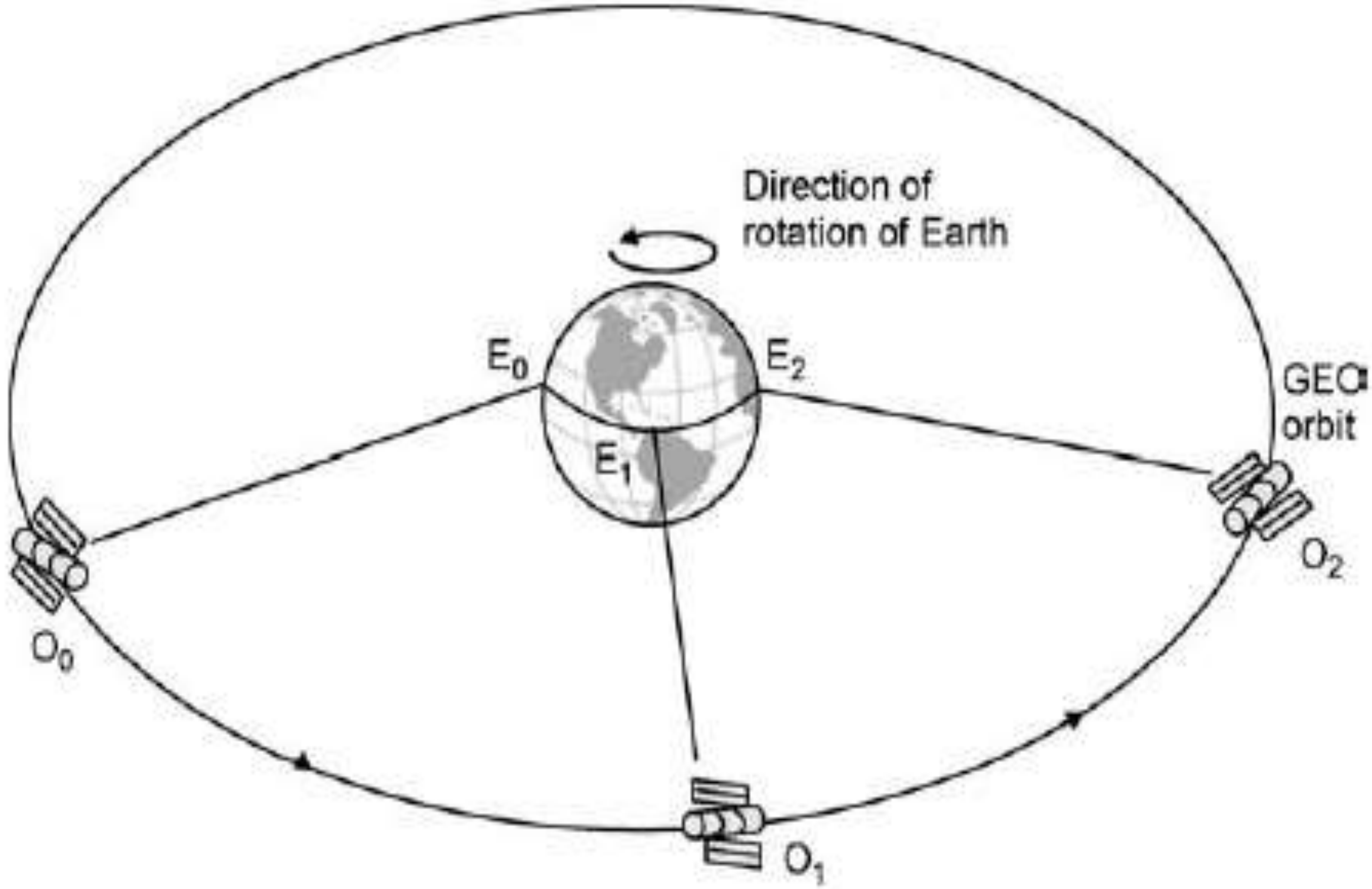
Figure 2.38: LEO, MEO and GEO

- **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 are mainly used for communication and navigation applications.

Geostationary Earth orbits (GEOs): It needs to be at a height of about 36 000 km, 35 786 km to be precise, above the surface of the Earth.

In order to remain in geostationary earth orbit, a satellite must fulfill the following conditions.

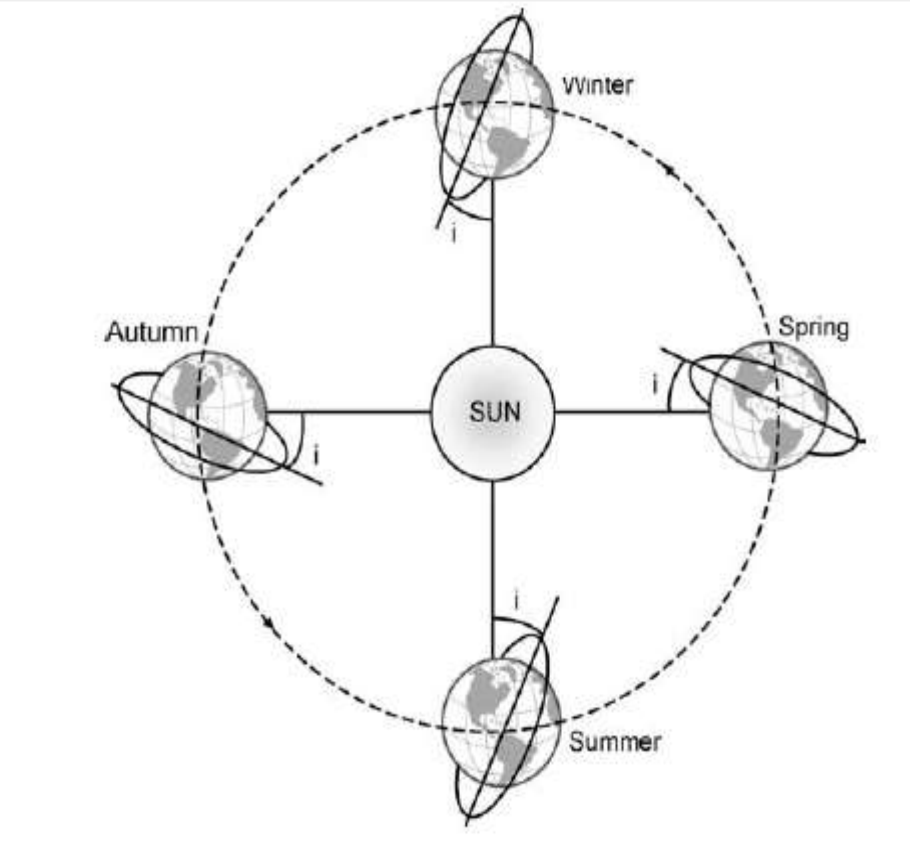
1. It must have a constant latitude, which is possible only at 0° latitude.
2. The orbit inclination should be zero.
3. It should have a constant longitude and thus have a uniform angular velocity, which is possible when the orbit is circular.
4. The orbital period should be equal to 23 hours 56 minutes,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



GEO satellites appear stationary wrt to a point on Earth

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

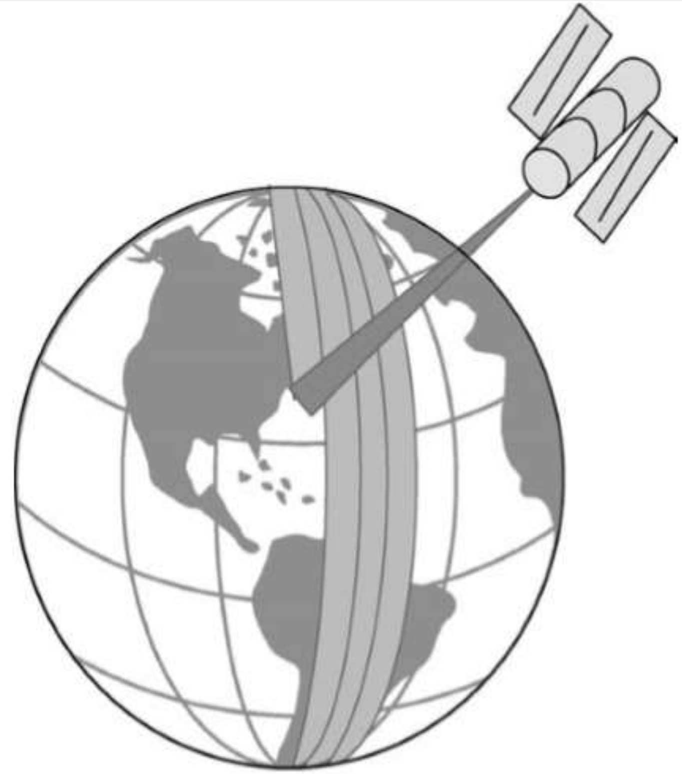


Sun-synchronous orbit

- Applications are
  - passive remote sensing,
  - meteorological,
  - military reconnaissance and
  - atmospheric studies.

1. The satellite passes over a given location on Earth every time at the same local solar time, thereby guaranteeing almost the same illumination conditions, varying only with seasons.
2. The satellite ensures coverage of the whole surface of the Earth, being quasi-polar in nature.

- Every time a sun-synchronous satellite completes one revolution around Earth, it traverses a thin strip on the surface of the Earth.
- During the next revolution it traverses another strip shifted westwards and the process of shift continues with successive revolutions, as shown in Figure 2.42



Earth coverage of sun-synchronous satellites

Fig.2.42

- The number of revolutions required before the satellite repeats the same strip sequence can be calculated, and is called one complete orbital cycle, which is basically the time that elapses before the satellite revisits a given location in the same direction.

- Orbital cycle means the whole number of orbital revolutions that a satellite must describe in order to be once again flying in the same direction over the same point on the Earth's surface.

- In the case of Landsat-1, -2 and -3 satellites each having an orbital cycle of 18 days.

Orbital Perturbations

- Various perturbing torques that cause variations in satellite's 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.

Effects of the disturbing forces:

- Satellite orbit tends to drift
- Orientation changes
- True orbit will be defined , by laws other than Kepler's laws.

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

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

- In the case of a geostationary satellite, a 1° drift in the east or west direction is equivalent to a drift of about 735 km along the orbit.(Fig. 3.27)

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

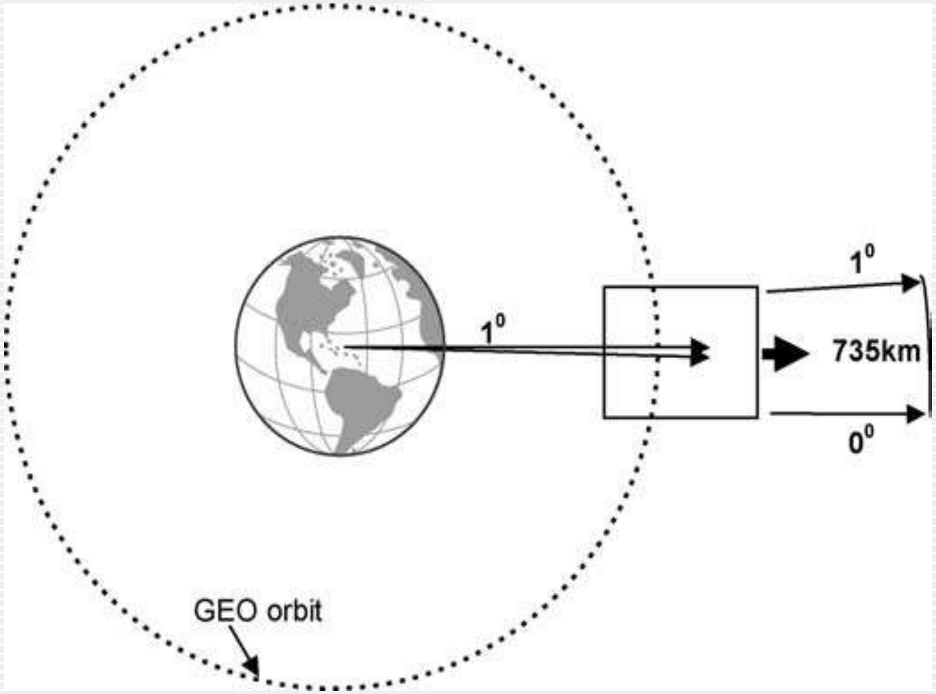


Figure 3.27 : Drift of a geostationary satellite

Reasons for the drift:

1)Non uniform gravitational field around Earth:

This results in acceleration and decceleration component varying with longitudinal location of Satellite.

2)Gravitational pulls of Sun and Moon:

This results in change in inclination of Satellite's orbit. The orbit is not an ellipse anymore.

After drift the Satellite does not return back to the same position as before. The time elapsed between the successive perigee passages is referred to as anomalistic period.

The anomalistic period (t_A) is given by

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

where

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

Here ω_o is the angular velocity for spherical Earth, **K** =66063.1704kmsq, **a** is the semi-major axis, **e** is the eccentricity and **i** = $\cos^{-1} W_Z$ (W_Z is the Z axis component of the orbit normal.

Satellite Stabilization

Stabilization techniques are used for satellite attitude control in the orbit

Types of stabilization:

- Spin stabilization
- Three-axis or body stabilization

Spin Stabilization

- **Spin stabilization** is achieved by **spinning** the exterior of the spacecraft on its axis at a fixed rate.
- 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. 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.

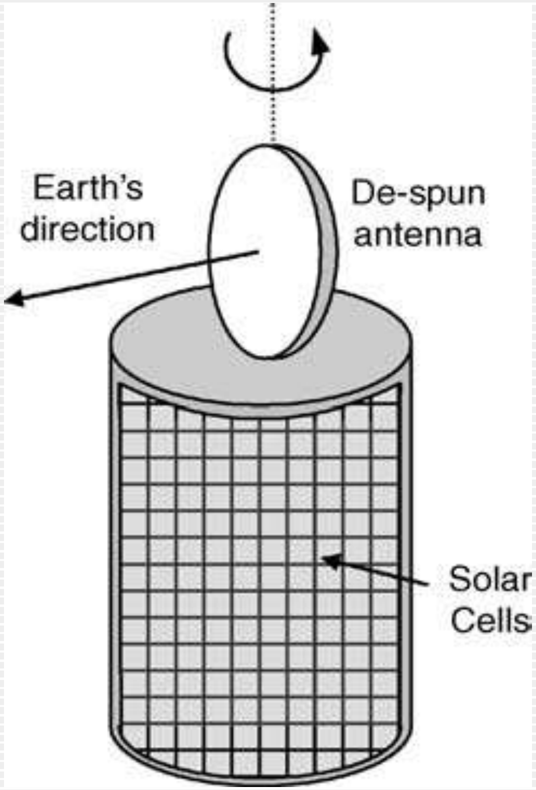


Figure 3.28
Spin stabilized satellite

- To maintain stability, the moment of inertia about the desired spin axis should at least be 10% greater than the moment of inertia about the transverse axis.

Spin configurations for stabilization:

- Simple spinner configuration
- Dual Spinner configuration
- Simple spinner configuration:

Satellite and payload & other subsystems are placed in the spinning section, while the antenna and the feed are placed in despun platform(opposite to spinning Sat body).

- Dual spinner configuration:

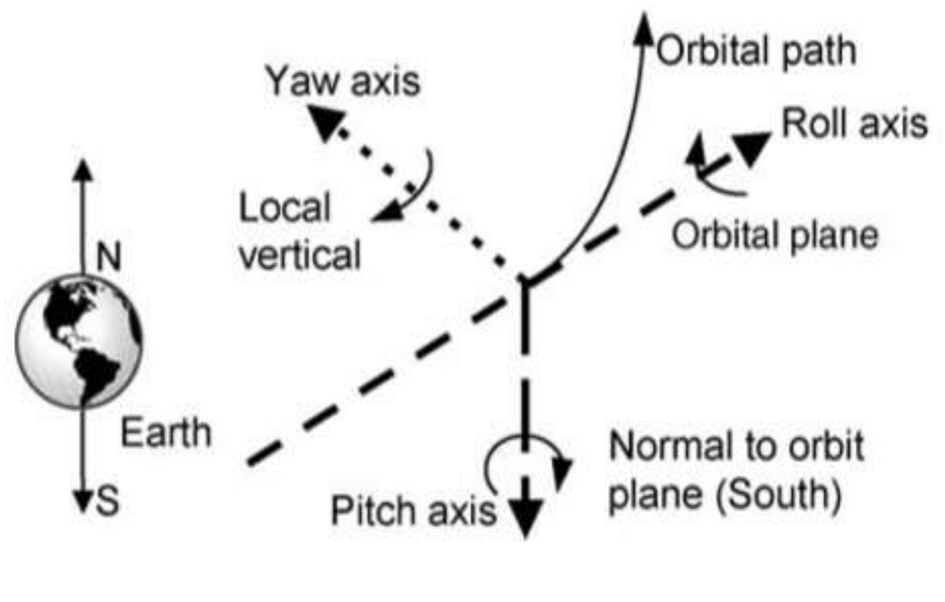
In this configuration the entire payload along with the antenna and the feed is placed on the de-spun platform and the other subsystems are located on the spinning.

Note1: modern spin stabilized Satellites employ dual spinner configuration.

Note2:Antenna mounting on despun platform ensures that antenna has a constant direction pointing.

Three - axis or body stabilization

• In this method 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. (Shown in Figure)



Three-axis stabilization

•The system uses reaction wheels or momentum wheels to correct orbit perturbations.

- 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.
- For example, an increase in speed of the wheel in the clockwise direction will make the satellite to rotate in a counterclockwise direction.

- 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.

Satellites belonging to three axis stabilization:

- Intelsat-5, Intelsat-7, Intelsat-8, GOES-8, GOES-9, TIROS-N and the INSAT series of satellites

Comparison between Spin stabilized and 3-axis stabilized satellites.

Sl no	Spin Stabilization	3-axis stabilization
1	Less power generation capability and less area for complex antenna structures	More power generation capability and more area for complex antenna structures
2	Simpler in design and less expensive	They are complex in design and costly
3	Power is given to Satellite even in transfer orbit	Not possible to give power during transfer, since the solar array is stored inside .

(10) A rocket injects a Satellite at perigee with a horizontal velocity of 8 km/s from a height of 1620 km from the surface of the Earth. what will be the velocity of the Satellite at a point distant 10000 km from the center of the Earth if the direction of Satellite makes an angle of 30° with the local horizontal at that point? Assume radius of Earth to be 6380 km.

SOLUTION: We Know

$$v_p = \sqrt{2\mu\left[\frac{1}{p} - \frac{1}{A+p}\right]}$$

$$= v \cdot d \cdot \cos\gamma / p$$

$v \to$ velocity of satellite

$v_p = 8$ km/s, $p = 1620 + 6380 = 8000$ km

If $d = 10{,}000$ km & $\gamma = 30°$,

$$v = \frac{v_p \times p}{d \cos\gamma} = \frac{(8000) \times 8}{10000 \times \cos 30°}$$

$\cos 30° \to 0.866$

$$= \frac{8 \times 8}{10 \times 0.866} = 7.39 \text{ km/s}.$$

Station keeping

- This is defined as the process of maintainence of satellite's orbit in right position against temporal drift

Reasons for drift:

- Gravitational pulls of Sun and Moon
- Solar radiation pressure
- Earth being imperfect sphere

Remedies :

- The corrections are done by jets of gas /water or firing of small rockets tied to body of satellite

Orbital effects on Satellite's performance

- The satellite is revolving constantly around the Earth, and the motion of the satellite has significant effects on its performance . They are:
- Doppler shift
- Effect due to variation in orbital distance
- Effect of solar eclipse
- Sun's Transit outage.

Doppler shift

- 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.
- 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,

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 (3×10^8 m/s)

Variation in 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 LEO and MEO satellites as compared to the geostationary satellite.

Solar Eclipse

- 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.

Sun's Transit outage

- These are times when the Satellite passes directly between Sun and Earth (Shown in the figure).

- Earth station antenna will receive signals from Satellite as well as microwave radiation emitted by the Sun.

- Sun acts as a noisy source and blocks out signal from Satellite.

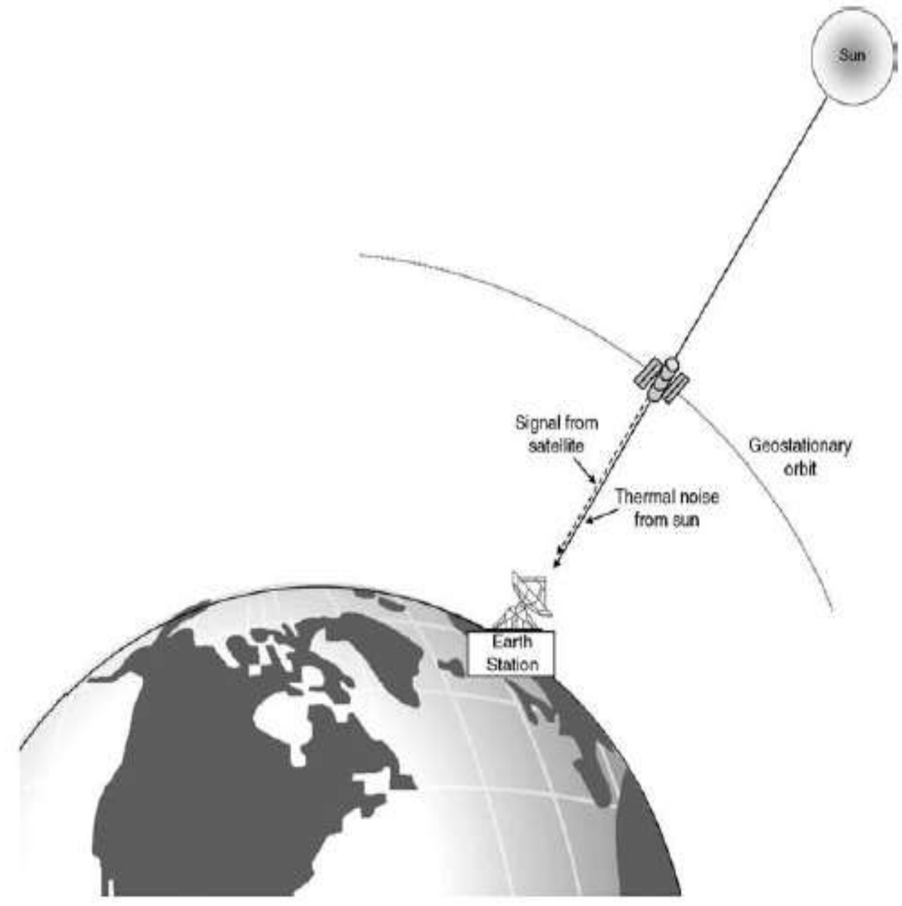


Figure 3.32 Sun outrage conditions

Eclipses

With reference to satellites, an eclipse is said to occur when the sunlight fails to reach the satellite's solar panel due to an obstruction from a celestial body. The frequent source of eclipse is due to the satellite coming in the shadow of the Earth. This is called solar eclipse.

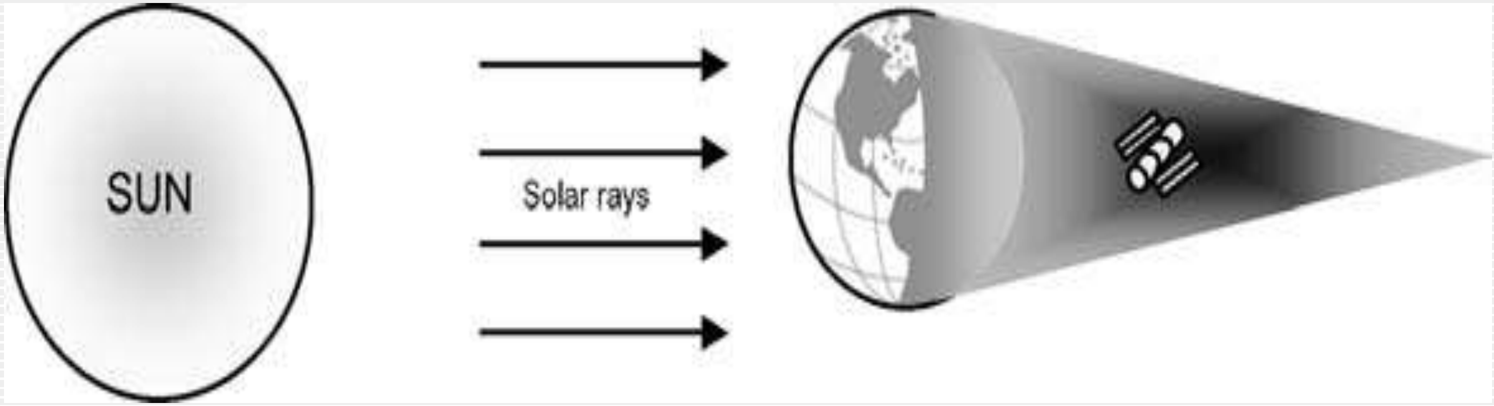


Figure 3.33: Solar eclipse

- The eclipse is total; i.e. the satellite fails to receive any light if it passes through the umbra region, which is the dark central region of the shadow.
- It receives very little light if it passes through the penumbra, which is the less dark region surrounding the umbra (Figure 3.34).

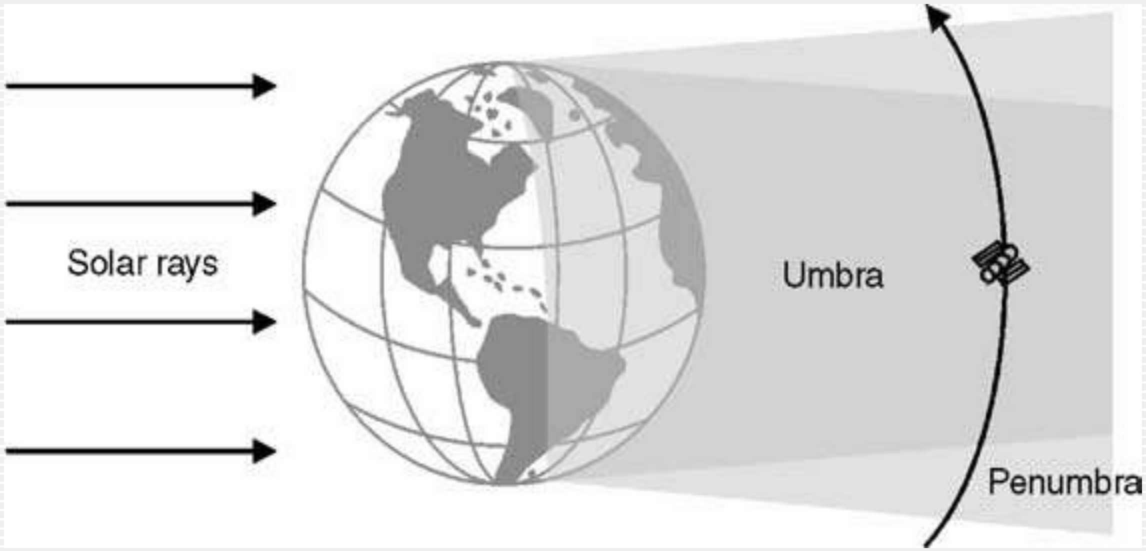


Figure 3.34 : Umbra and penumbra

- The eclipse occurs as the Earth's equatorial plane is inclined at a constant angle of about 23.5° *to its ecliptic plane, which is the plane of the Earth's orbit extended to infinity*.

- *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 of eclipse is the worst during the equinoxes and lasts for about 72 minutes.

- 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, thereby receiving only a part of solar light for that time.

- During the rest of the year, the geostationary satellite orbit passes either above or below the umbral cone. It is at the maximum distance at the time of the solstices, above the umbral cone at the time of the summer solstice (20–21 June) and below it at the time of the winter solstice (21–22 December).
- Figure 3.35 further illustrates the phenomenon.

- 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.

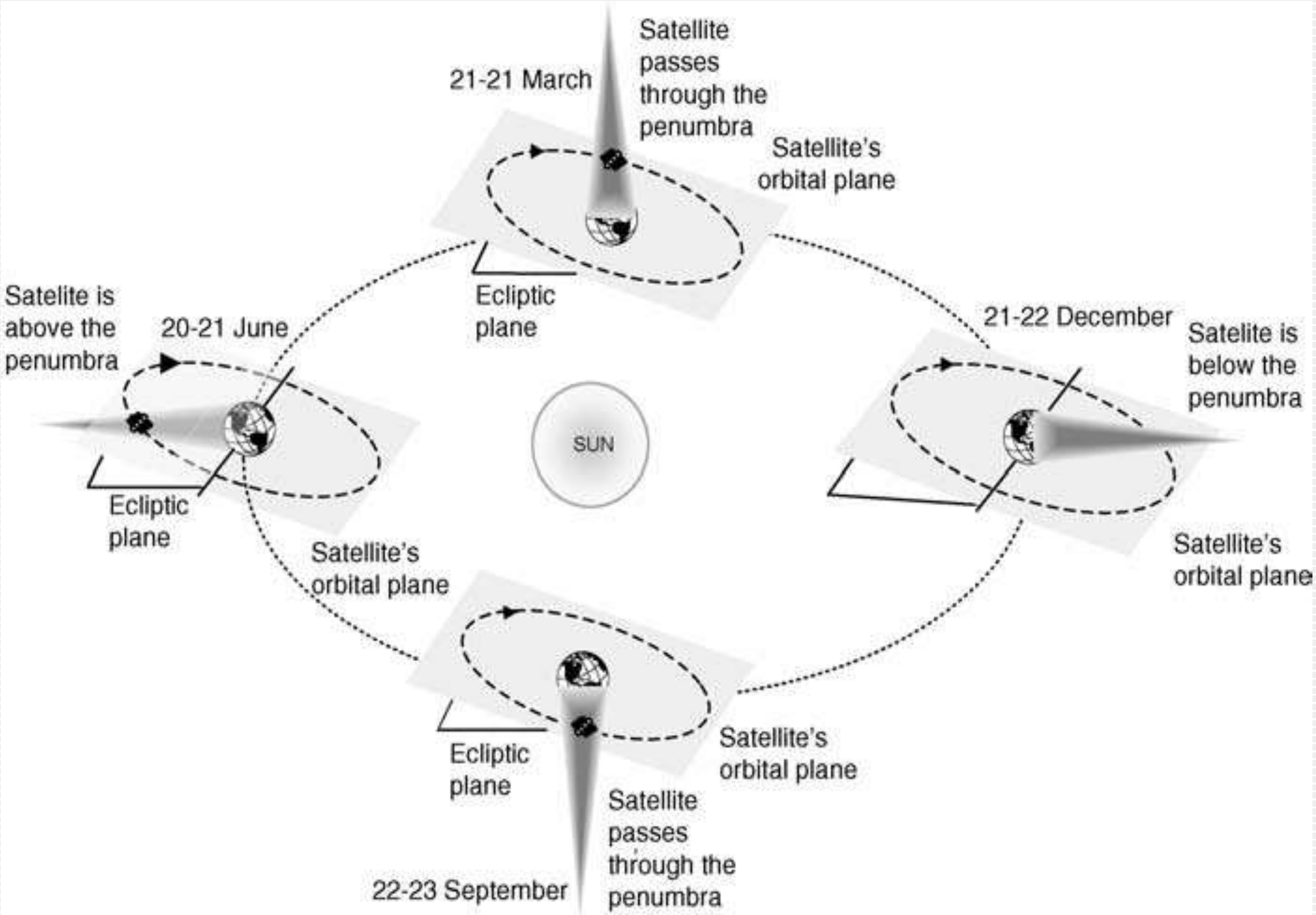


Figure 3.35 : Positions of the geostationary satellite during the equinoxes and solstices

- The duration of an eclipse on a given day around the equinox can be seen from the graph in Figure 3.36.

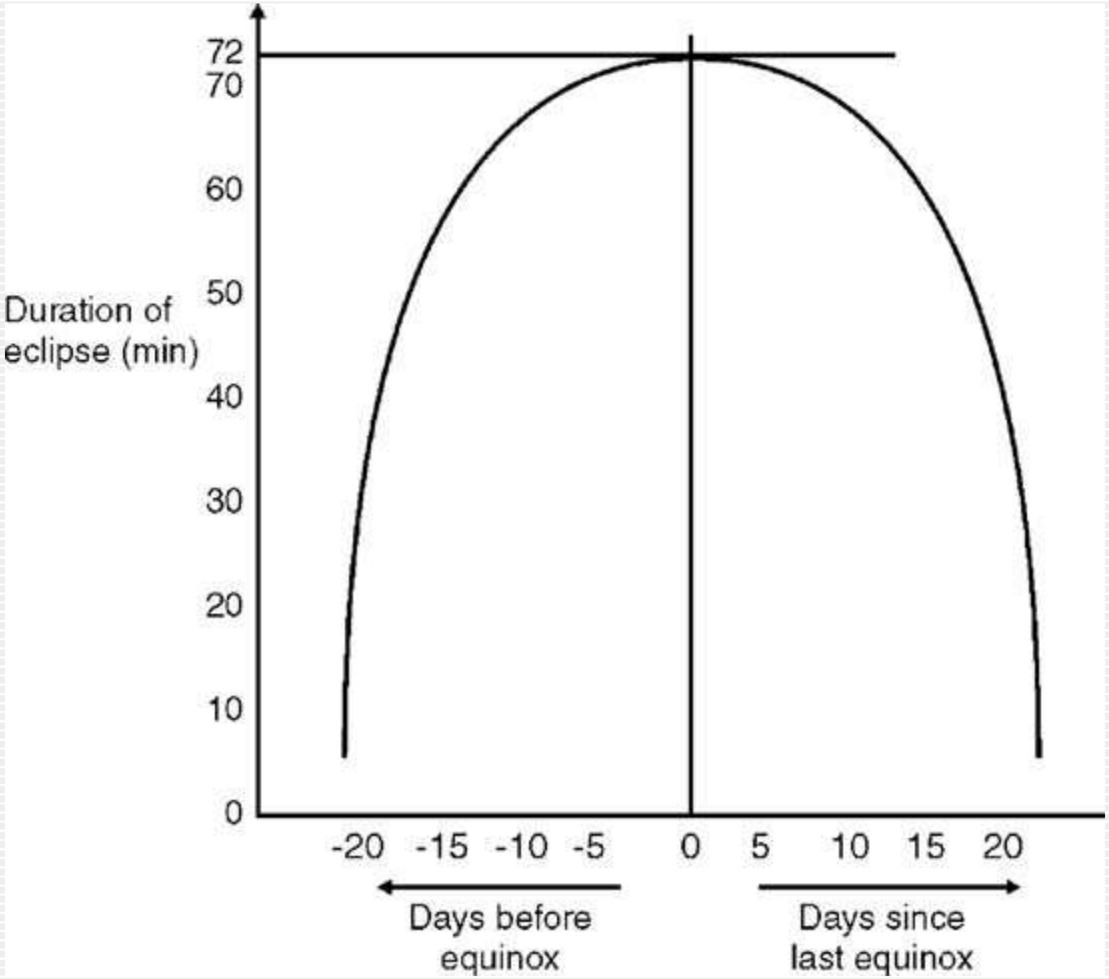


Figure 3.36: Duration of the eclipse before and after the equinox

- Another type of eclipse known as the lunar eclipse occurs when the moon's shadow passes across the satellite (Figure 3.37).

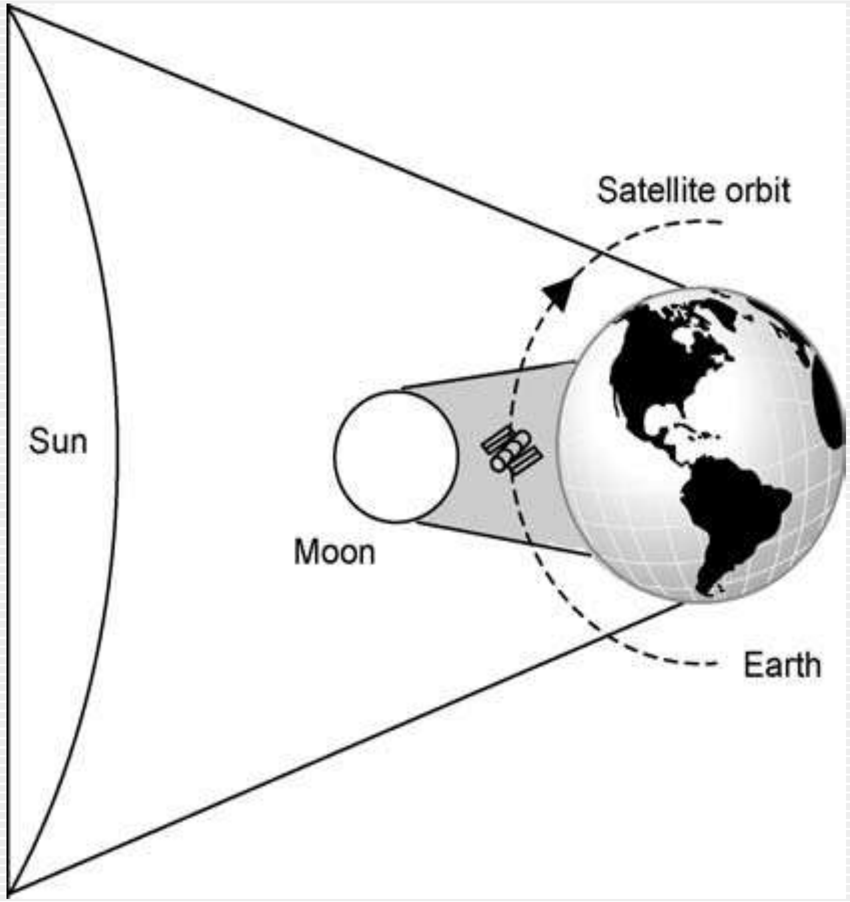


Figure 3.37 : Lunar eclipse

Look angles of a Satellite

- They refer to coordinates to which the earth station antenna must be pointed in order to communicate with the Satellite.
- They are expressed interms 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.

Determination of the look angles of a satellite:

- Its precise location should be known.
- The location of a satellite is 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.

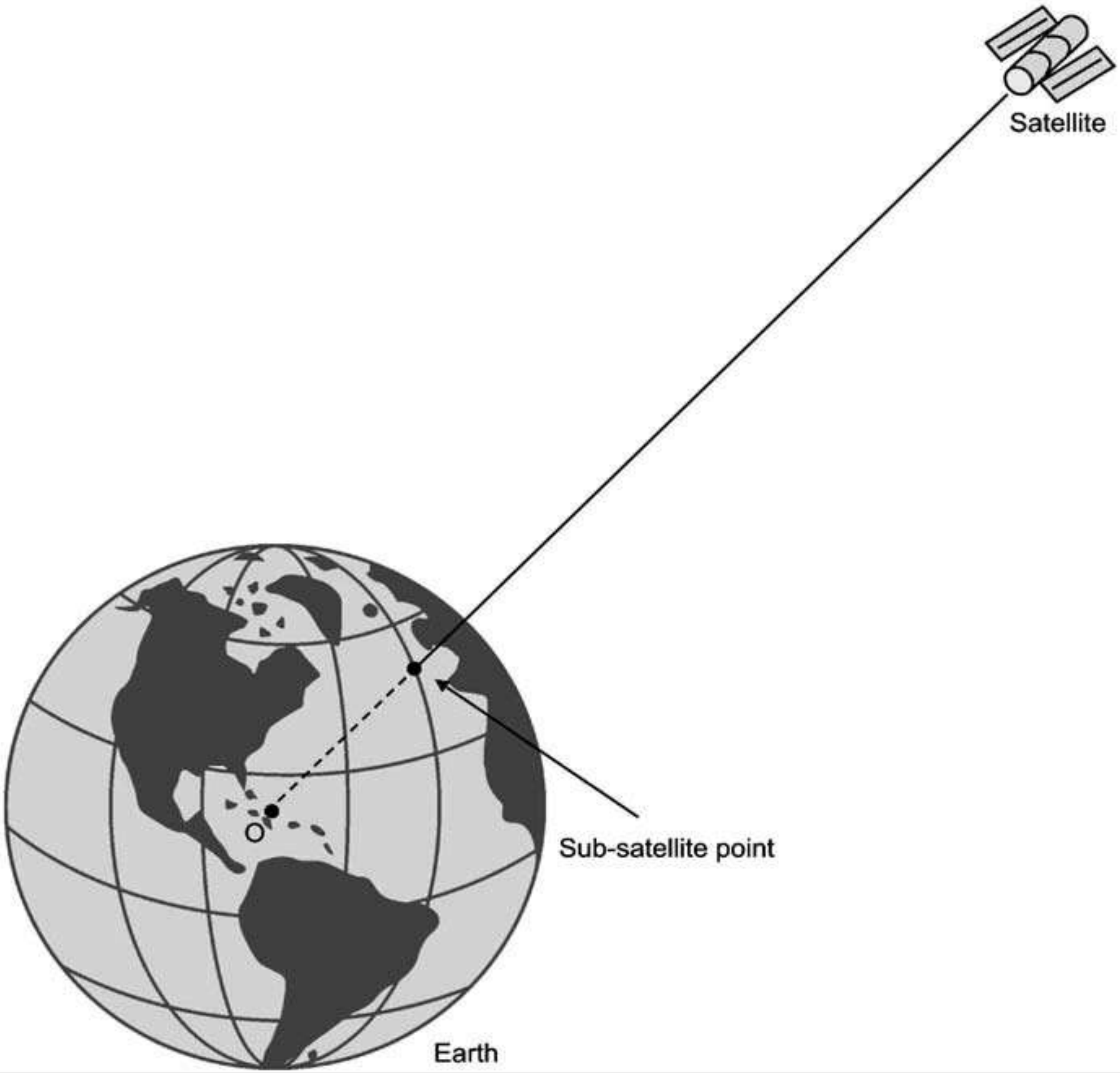


Figure 3.38: Sub-satellite point

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. (Fig. 3.39)

- Calculation of Azimuth angle:
- 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*

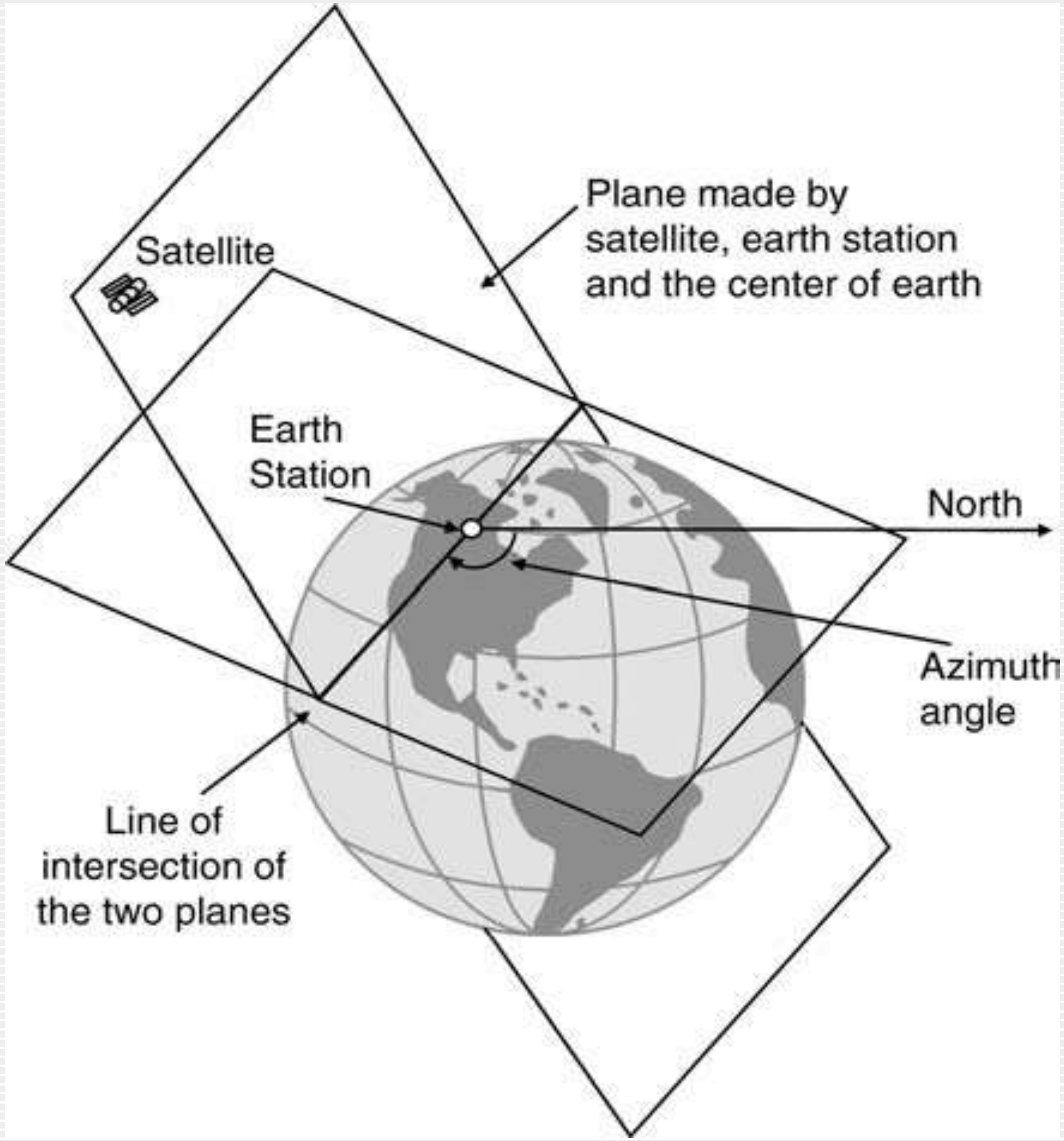


Figure 3.39: Azimuth angle

▪ Earth station in the southern hemisphere:

A = A' . . . when the Earth station is to the west of the satellite.

A = 360° – A' . . . when the Earth station is to the east of the satellite.

where *A'* can be computed from

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

where

θ_s = satellite longitude

θ_L = Earth station longitude

θ_l = Earth station latitude

Elevation Angle:

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

▪Figures 3.40 (a) show the elevation angle for a satellite and Earth station positions.

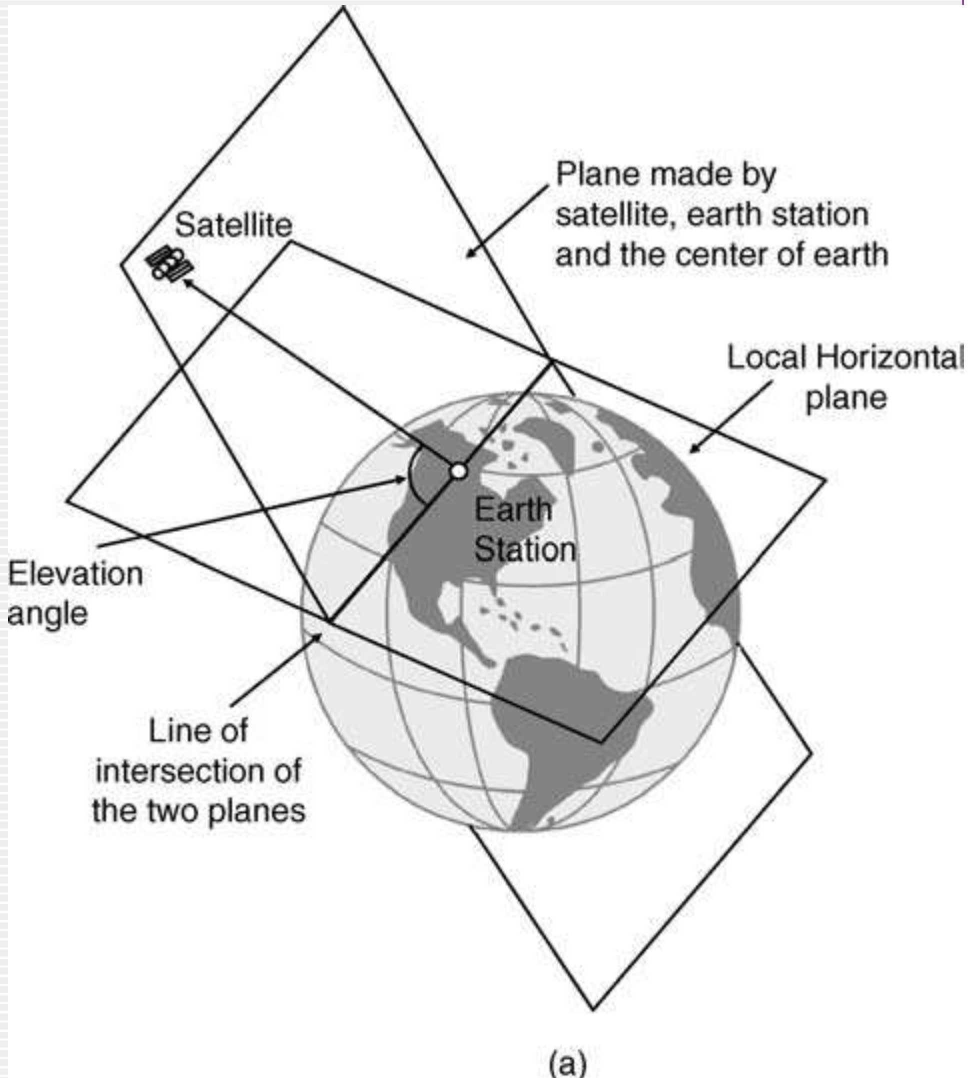


Figure 3.40: Earth station elevation angle.

Elevation angle can be computed from

$$E = \tan^{-1}\left[\frac{r - R\cos\theta_{\mathrm{l}}\cos|\theta_{\mathrm{s}} - \theta_{\mathrm{L}}|}{R\sin\{\cos^{-1}(\cos\theta_{\mathrm{l}}\cos|\theta_{\mathrm{s}} - \theta_{\mathrm{L}}|)\}}\right] - \cos^{-1}(\cos\theta_{\mathrm{l}}\cos|\theta_{\mathrm{s}} - \theta_{\mathrm{L}}|)$$

where

r = orbital radius, R = Earth's radius

θ_s = Satellite longitude, θ_L = Earth station longitude, θ_l = Earth station latitude

Computing the Slant Range

- Slant range of a satellite is defined as the range or the distance of the satellite from the Earth station.
- The elevation angle E, *has a direct bearing on the slant range. The* smaller the elevation angle of the Earth station, the larger is the slant range and the coverage angle.
- The slant range can be computed from

$$\text{Slant range } D = \sqrt{R^2 + (R+H)^2 - 2R(R+H)\sin\left[E + \sin^{-1}\left\{\left(\frac{R}{R+H}\right)\cos E\right\}\right]}$$

$$\text{Coverage angle } \alpha = \sin^{-1}\left\{\left(\frac{R}{R+H}\right)\cos E\right\}$$

Where

R = radius of the Earth
E = angle of elevation
H = height of the satellite above the surface of the Earth

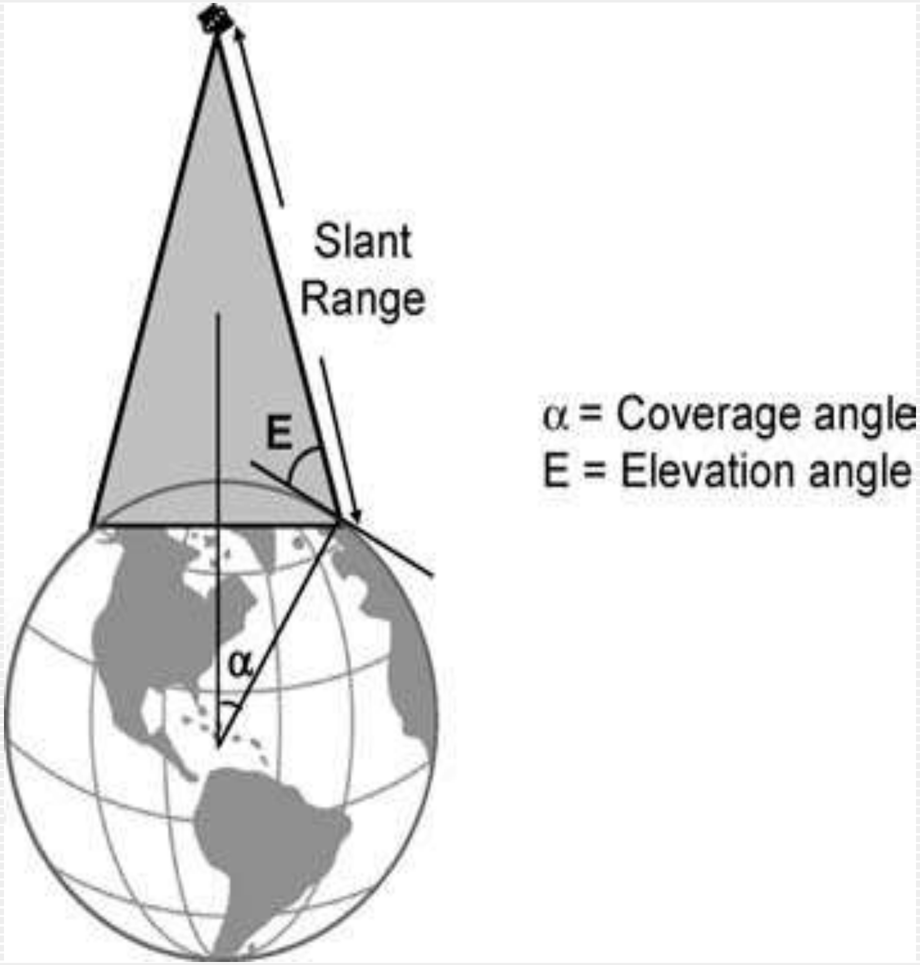


Figure 3.41 Elevation angle, slant range and coverage angle.

- Conclusion from the above expression is that a zero angle of elevation leads to the maximum coverage angle.
- A larger slant range means a longer propagation delay time and a greater impairment of signal quality, as the signal has to travel a greater distance through the Earth’s atmosphere.

Computing the Line-of-Sight Distance between Two Satellites

- The line-of-sight distance between two satellites placed in the same circular orbit can be computed from triangle ABC formed by the points of location of two satellites and the centre of the Earth.
- The line-of-sight distance AB in this case is given by

$$AB = \sqrt{(AC^2 + BC^2 - 2\,AC\,BC\,\cos\theta)}$$

- Angle θ is the angular separation of the longitudes of the two satellites.
- AC and BC : Distances of A and B from the center of earth.

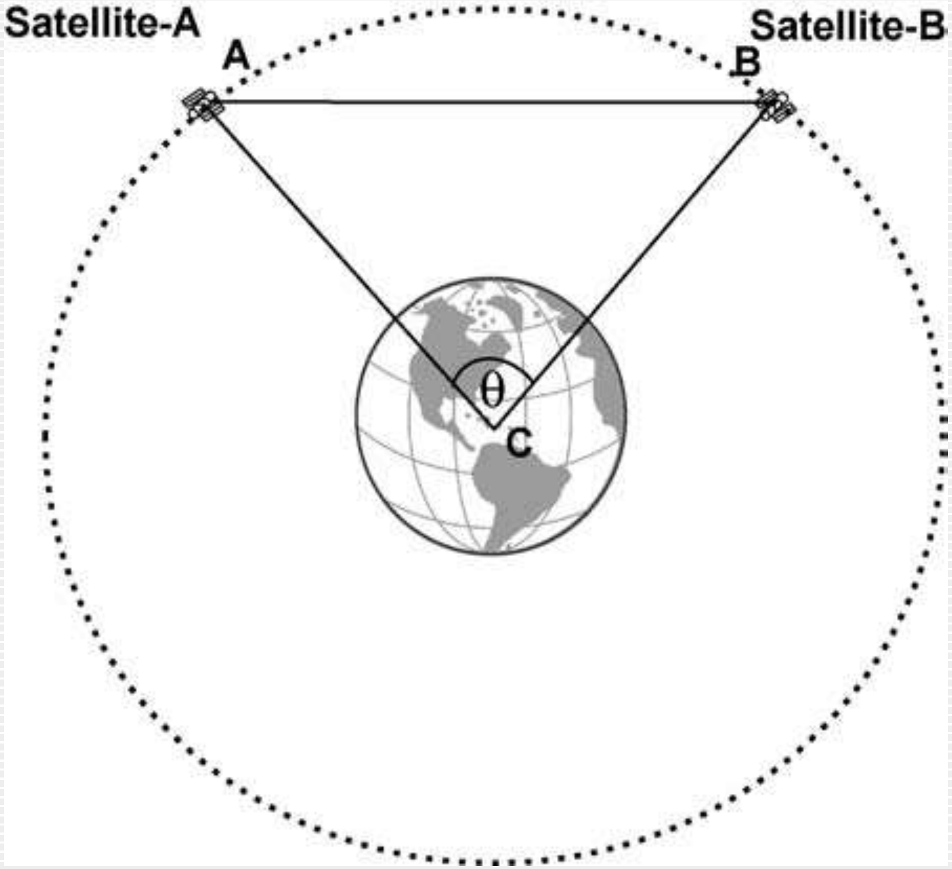


Figure 3.42 Line-of-sight distance between two satellites

The maximum line-of-sight distance (AB) equals OA + OB, which further equals 2OA or 2OB as OA=OB. If *R is the radius of the Earth and H is the height of satellites* above the surface of the Earth, then

$$\mathrm{OA} = \mathrm{AC}\ \sin\beta = (R+H)\ \sin\beta$$

Now

$$\beta = \cos^{-1}\left(\frac{R}{R+H}\right)$$

Therefore

$$\mathrm{OA} = (R+H)\ \sin\left[\cos^{-1}\left(\frac{R}{R+H}\right)\right]$$

and

$$\text{Maximum line-of-sight distance} = 2(R+H)\ \sin\left[\cos^{-1}\left(\frac{R}{R+H}\right)\right]$$

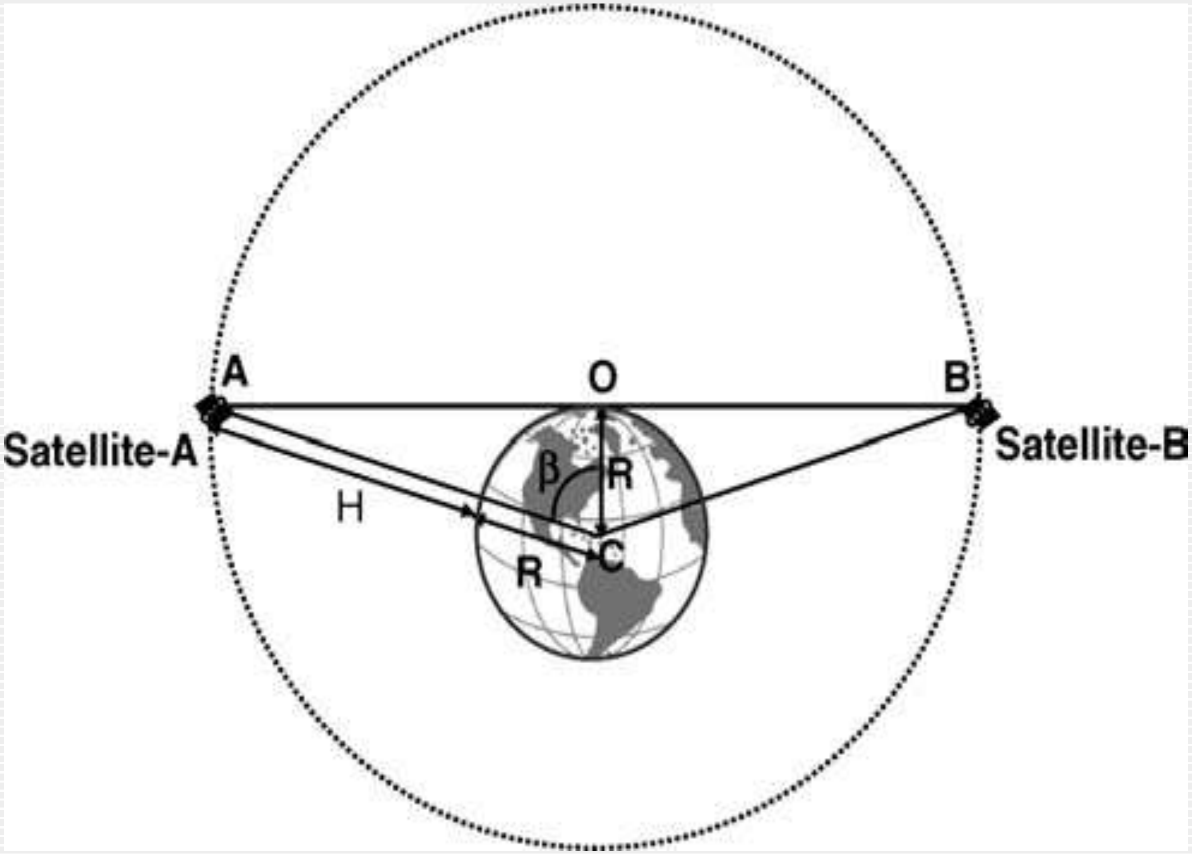


Figure 3.43 Maximum line-of-sight distance between two satellites

VTU: Jan 2020

Problem:

An Earth station is located at 30°W longitude and 60°N latitude. Determine look angle parameters with respect to GEO satellite located at 50°W longitude. The orbital radius is 42 164 km. (Assume the radius of the Earth to be 6378 km.).

Steps in solving the problem:

i) Calculate Azimuth angle

ii) Calculate Elevation angle.

Solution:

Since the Earth station is in the northern hemisphere and is located towards east of the satellite, the azimuth angle A is given by (180◦ + A'), where A' can be computed from

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

where

θ_s = satellite longitude

θ_L = Earth station longitude

θ_l = Earth station latitude

where

θs = satellite longitude = 50◦W

θ_L = Earth station longitude = 30◦W

θ_l = Earth station latitude = 60◦N

$\theta s - \theta_L = 20°$

i) Azimuth angle, *A = 180◦ + A'*

ii) The **Earth station elevation angle** is given by

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

Answer:

$$A' = \tan^{-1}(\tan 20^o/\tan 60^o)$$
$$= \tan^{-1}(0.364/0.866) = \mathbf{22.8^o}$$

Azimuth angle, **A** = $180^\circ + 22.8^\circ$ = $\underline{\mathbf{202.8^\circ}}$

Elevation angle,

$$E = \tan^{-1}[(42164 - 6378 \cos 60^o \cos 20^o)/6378 \sin\{\cos^{-1}(\cos 60^o \cos 20^o)\}] - \cos^{-1}(\cos 60^o \cos 20^o)$$

$$\underline{\mathbf{E = 19.8^o}}$$

PROBLEM:

A typical Molniya orbit has a perigee and apogee heights above the surface of the Earth as 400 km & 40000 km respectively. Verify that the orbit has a 12 hour time period assuming radius of Earth to be 6380 km and $\mu = 39.8 \times 10^{13}\ Nm^2/kg$

SOLUTION: The orbital period of a Satellite in elliptical orbit is given by

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

$a \rightarrow$ Semi-major axis:

$$a = \frac{40000 + 400 + 6380 + 6380}{2}$$

$$= 26580 \text{ km}$$

$$\therefore T = \frac{2 \times 3.14 \times (26580 \times 10^3)^{3/2}}{\sqrt{39.8 \times 10^{13}}}$$

$$= 43137 \text{ s}$$

Note

a is expressed in meters.

Since μ is in Nm^2/kg

(8) The apogee and perigee distance of a certain elliptical satellite orbit are 42000 km & 8000 km respectively. If the velocity at perigee distance is 9.142 km/s what would be the velocity at the apogee point?

If P = perigee distance = 8000 km

A = apogee distance = 42000 km

v_{PA} = Velocity at apogee

v_{PP} = Velocity at perigee = 9.142 km/s.

$$v_P \times P = v_A \times A$$

$$v_A = \frac{v_P \times P}{A} = \frac{8000 \times 9.142}{42000}$$

$$= 1.741 \text{ km/s.}$$

Kepler’s second law - Satellite position at any given time.

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

Problem:

The elliptical eccentric orbit of a satellite has its semi-major and semi-minor axes as 25,000 km and 18,330 km respectively. Determine the apogee and perigee distances.

Solution:

If ra and rp are the apogee and perigee distances respectively, a and b are the semi-major and semi-minor axis respectively and e is the eccentricity, then

$$a = \frac{r_a + r_p}{2} \qquad \text{and} \qquad b = \sqrt{(r_a r_p)}$$

Now $(r_a + r_p)/2 = 25\,000$, which gives $r_a + r_p = 50\,000$

and $\sqrt{(r_a r_p)} = 18\,330$, which gives $r_a r_p = 335\,988\,900$

Substituting the value of r_p from these equations gives

$$r_a(50\,000 - r_a) = 335\,988\,900$$

which gives

$$r_a^2 - 50\,000 r_a + 335\,988\,900 = 0$$

Solving this quadratic equation gives two solutions:

$$r_a = 42\,000\text{ km},\ \ 8000\text{ km}$$

If the semi-major axis is 25 000 km, r_a cannot be 8000 km. Therefore

$$\text{Apogee distance } r_a = 42\,000\text{ km}$$

and

$$\text{Perigee distance } r_p = 50\,000 - 42\,000 = 8000\text{ km}$$

Alternative solution:

$$e = \frac{\sqrt{(a^2 - b^2)}}{a} = e = \frac{\sqrt{[(25\,000)^2 - (18\,330)^2]}}{25\,000} = 0.68$$

This gives

Apogee distance $= a(1 + e) = 25\,000 \times 1.68 = 42000$ km

and Perigee distance $= a(1 - e) = 25\,000 \times 0.32 = 8000$ km

Therefore, from both the methods,

Apogee distance = 42 000 km and Perigee distance = 8000 km

Thank you