EC8094 - SATELLITE COMMUNICATION

L T P C
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OBJECTIVES:

The student should be made to:

- Understand the basics of satellite orbits
- Understand the satellite segment and earth segment
- Analyze the various methods of satellite access
- Understand the applications of satellites
- Understand the basics of satellite Networks

UNIT I SATELLITE ORBITS 9

Kepler's Laws, Newton's Law, Orbital Parameters, Orbital Perturbations, Station Keeping, Geo Stationary and Non Geo-Stationary Orbits – Look Angle Determination - Limits of Visibility – Eclipse - Sub Satellite Point – Sun Transit Outage - Launching Procedures - Launch Vehicles and Propulsion.

UNIT II SPACE SEGMENT 9

Spacecraft Technology - Structure, Primary Power, Attitude and Orbit Control, Thermal Control and Propulsion, Communication Payload and Supporting Subsystems, Telemetry, Tracking and Command – Transponders - The Antenna Subsystem.

UNIT III SATELLITE LINK DESIGN 9

Basic Link Analysis, Interference Analysis, Rain Induced Attenuation and Interference, Ionospheric Characteristics, Link Design with and without Frequency Reuse.

UNIT IV SATELLITE ACCESS AND CODING METHODS 9

Modulation and Multiplexing: Voice, Data, Video, and Analog – Digital Transmission System, Digital Video Broadcast, Multiple Access: FDMA, TDMA, CDMA, DAMA Assignment Methods, Compression – Encryption, Coding Schemes.

UNIT V SATELLITE APPLICATIONS 9

INTELSAT Series, INSAT, VSAT, Mobile Satellite Services: GSM, GPS, INMARSAT, LEO, MEO, Satellite Navigational System. GPS Position Location Principles, Differential GPS, Direct Broadcast satellites (DBS/DTH).

TOTAL:45 PERIODS

OUTCOMES:

At the end of the course, the student would be able to:

- Analyze the satellite orbits
- Analyze the earth segment and space segment
- Analyze the satellite Link design
- Design various satellite applications

TEXT BOOKS:

1. Dennis Roddy, "Satellite Communication", 4th Edition, Mc Graw Hill International, 2006.
2. Timothy,Pratt,Charles,W.Bostain,JeremyE.Allnutt,"SatelliteCommunication",2ndEdition, Wiley Publications,2002

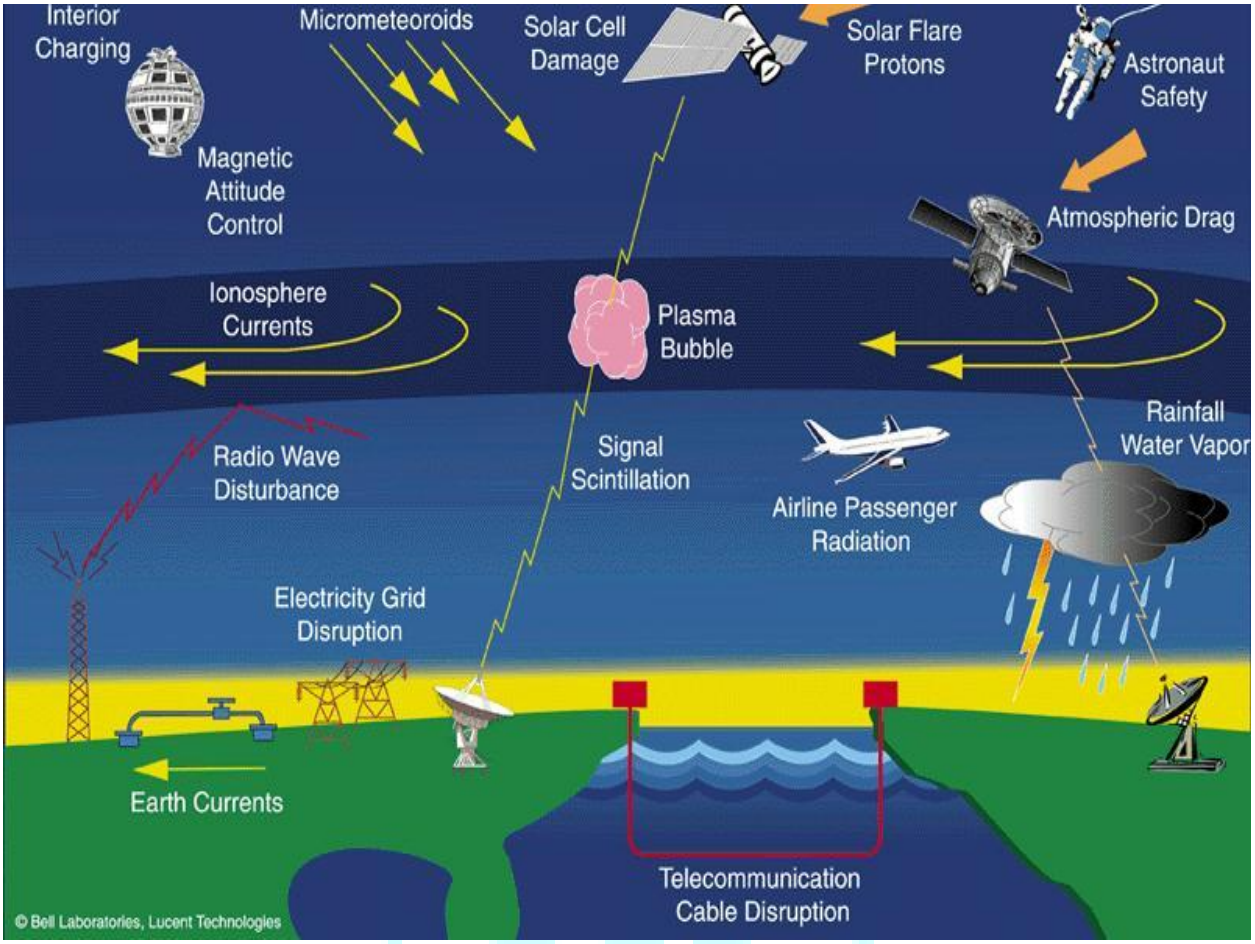
REFERENCES:

1. Wilbur L.Pritchard, Hendri G. Suyderhoud, Robert A. Nelson, "Satellite Communication Systems Engineering", Prentice Hall/Pearson, 2007.
2. N.Agarwal, "Design of Geosynchronous Space Craft", Prentice Hall, 1986.
3. Bruce R. Elbert, "The Satellite Communication Applications", Hand Book, Artech House Bostan London, 1997.
4. Tri T. Ha, "Digital Satellite Communication", II nd edition, 1990.
5. Emanuel Fthenakis, "Manual of Satellite Communications", Mc Graw Hill Book Co.,1984.
6. Robert G. Winch, "Telecommunication Trans Mission Systems", Mc Graw-Hill Book Co., 1983.
7. Brian Ackroyd, "World Satellite Communication and earth station Design", BSP professional Books, 1990.
8. G.B.Bleazard, "Introducing Satellite communications", NCC Publication, 1985.
9. M.Richharia, "Satellite Communication Systems-Design Principles", Macmillan 2003.

UNIT I

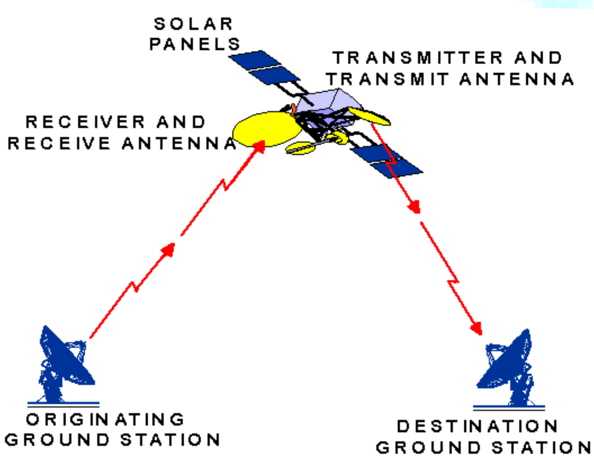
SATELLITE ORBITS

Kepler's Laws, Newton's Law, Orbital Parameters, Orbital Perturbations, Station Keeping, Geo Stationary and Non Geo-Stationary Orbits – Look Angle Determination - Limits of Visibility – Eclipse - Sub Satellite Point – Sun Transit Outage - Launching Procedures - Launch Vehicles and Propulsion.



Satellite

The word satellite originated from the Latin word "**Satellit**"- meaning an attendant, one who is constantly covering around & attending to a "master" or big man



- Earth Stations – antenna systems on earth
- Uplink – transmission from an earth station to a satellite
- Downlink – transmission from a satellite to an earth station
- Transponder – electronics in the satellite that convert uplink signals to downlink signals

First satellite launched into space: Sputnik (1957).

Motion of Space Objects

1473 -1543 Copernicus - Heliocentric (sun in the center) Orbit

1546 – 1601 Tycho Brahe
Before telescope followed the planets (acquired quality data)

1571 – 1630 Johannes Kepler
Discovered orbital path to be elliptical around focus point
Keplers 3 laws of planetary motion

1642 – 1727 Sir Isaac Newton
Physical Principals – Universal law of Gravitation

NEWTON'S LAWS

Newton's First law: Law of Inertia

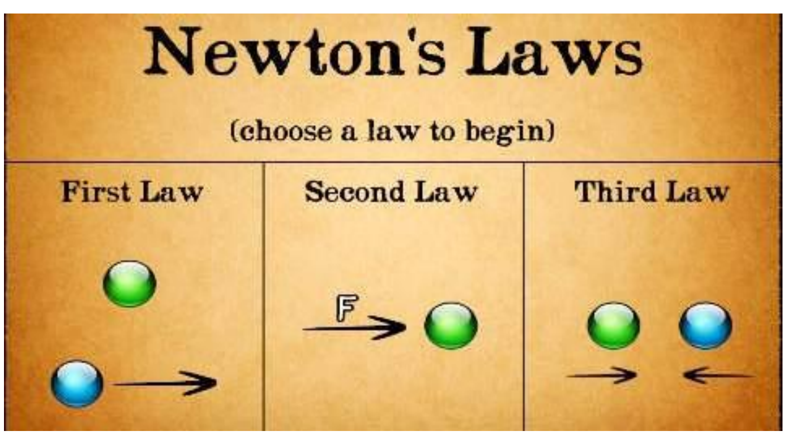
Everybody continues in a state of uniform motion unless it is compelled to change that state by a force imposed upon it.

Newton's Second law: Law of Momentum

Change in momentum is proportional to and in the direction of the force applied. Momentum equals mass x velocity. Change in momentum gives: F=ma.

Newton's Third law: Action – Reaction

For every action, there is an equal and opposite reaction. Hints at conservation of momentum



Origin of orbital mechanics

Kepler's laws:

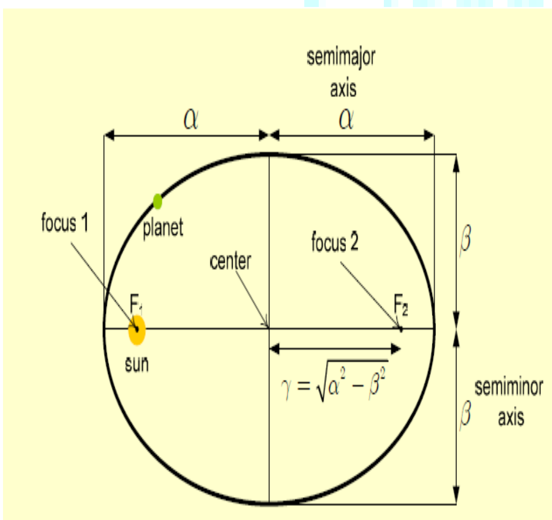
Kepler's laws apply quite generally to any two bodies in space which interact through gravitation. The more massive of the two bodies is referred to as the *primary,* the other, the *secondary* or *satellite*.

Keplers 3 (empirical) laws of Planetary Motion

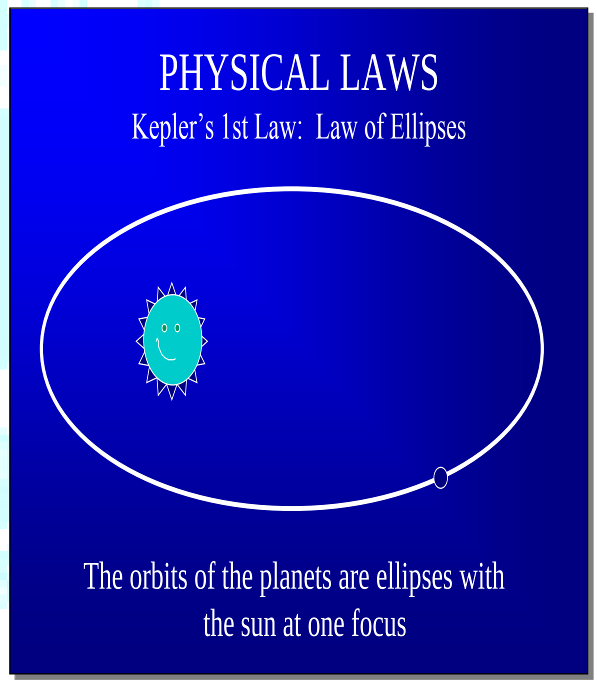
1. **Kepler's first law** states that, the path followed by a satellite around a primary is elliptical with the center of masses at one of the foci.

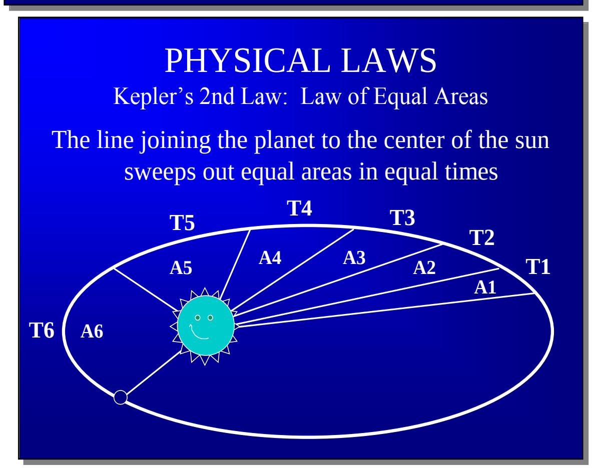
   (OR)

   "The orbital path of a planet takes the shape of an ellipse, with the Sun located at one of its focal points."



$$\text{eccentricity} = \varepsilon = \frac{\sqrt{\alpha^2 - \beta^2}}{\alpha} = \frac{\gamma}{\alpha}$$





2. **Kepler's second law** states that, for equal time intervals, a satellite will sweep equal areas in its orbital plane.

   (OR)

   The line from the sun to a planet sweeps out equal areas in equal time intervals.

3. **Kepler's third law** states that, the square of the periodic time of orbit is proportional to the cube of the mean distance between the two bodies.

$$a^3 = \frac{\mu}{n^2}$$

Where,

T is the orbital period

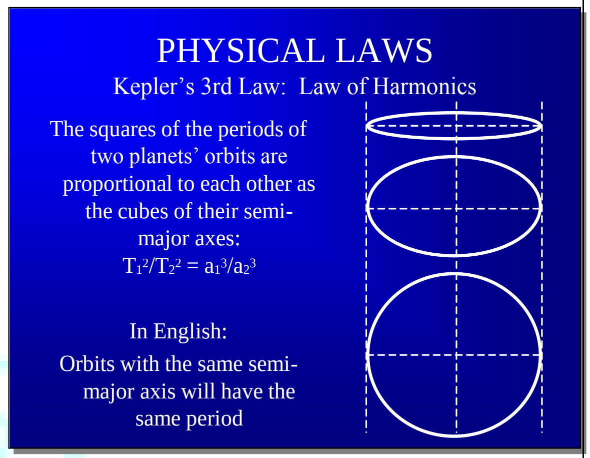
a is the semi major axis of the orbital ellipse

μ is the Kepler's constant = $3.986005*10^{14}$

(OR)

The ratio of the square of the planet's orbital period and the cube of the mean distance from the Sun is constant

$$\mathbf{(D_1/D_2)^3 = (P_1/P_2)^2}$$



Example:

Calculate the radius of a circular orbit for which the period is 1 day.

Solution There are 86,400 seconds in 1 day, and therefore the mean motion is

$$orbit\ period\ p = \frac{2\pi}{n}$$

$$\text{Mean motion of the satellite } n = \frac{2\pi}{86400}$$

n =7.272 * 10^{-5} rad/s

From Kepler's third law:

$$a^3 = \frac{\mu}{n^2}$$

$$n = \left(\frac{\mu}{a^3}\right)^{\frac{1}{3}}$$

$$a = \left[\frac{3.986005*10^{14}}{(7.272*10^{-5})^2}\right]^{\frac{1}{3}} = 42{,}241 \text{ km}$$

Since the orbit is circular the semi major axis is also the radius.

Frequency Allocation and Regulatory Aspects

- Frequency band for satellite services are shared with terrestrial services.
- Satellite signal strength is constrained to avoid interference by it to others.
- Thus a large antenna and sensitive receiver are needed at the earth station.
- Frequency sharing techniques are important study area.
- Many satellites have to share a limited frequency band (and limited orbital arc) thus coordination in frequency and orbital location is important.
- Frequency allocations are done by international agreements.

Frequency Allocation for Satellite Communication

Band	Frequency Range	Total Bandwidth	General Applications
L	1 TO 2 GHz	1 GHz	Mobile Satellite Services (MSS)
S	2 TO 4 GHz	2 GHz	MSS, NASA, Deep Space Research
C	4 TO 8 GHz	4 GHz	Fixed Satellite Services (FSS)
X	8 TO 12.5 GHz	4.5 GHz	FSS Military, Terrestrial Explosion and Metrological Satellites
Ku	12.5 TO 18 GHz	5.5 GHz	FSS, Broadcast Satellite Services (BSS)
K	18 TO 26.5 GHz	8.5 GHz	BSS, FSS
Ka	26.5 TO 40 GHz	13.5 GHz	FSS

Regulatory Aspects:

Domestic: Example:

Federal Communication Commission (FCC)
National Telecommunication and Information Administration (NITA)
In Pakistan, PTA (Pakistan Telecommunication Authority)

International: Example: International Telecommunication Union (ITU)

- Formed in 1932 from the International Telegraph Union
- Consists of over 150 members nations

- World Administrative radio Conference (WARC)
- International Radio consultative committee (CCIR) consists of 13 group.
- World is divided into three regions:
  Region 1 – Europe, Africa, Soviet Union on Mongolia
  Region 2 – North and South America, Greenland
  Region 3 – Asia, Australia and South West Pacific

Frequency Bands Available for Satellite Communications

Band	Uplink (GHz)	Downlink (GHz)
C	6	4
Ku	14	12
Ka	30	20
X	8.2	7.5
S	40	20
Q	44	21
L	1.525 to 1.559	1.626 to 1.660

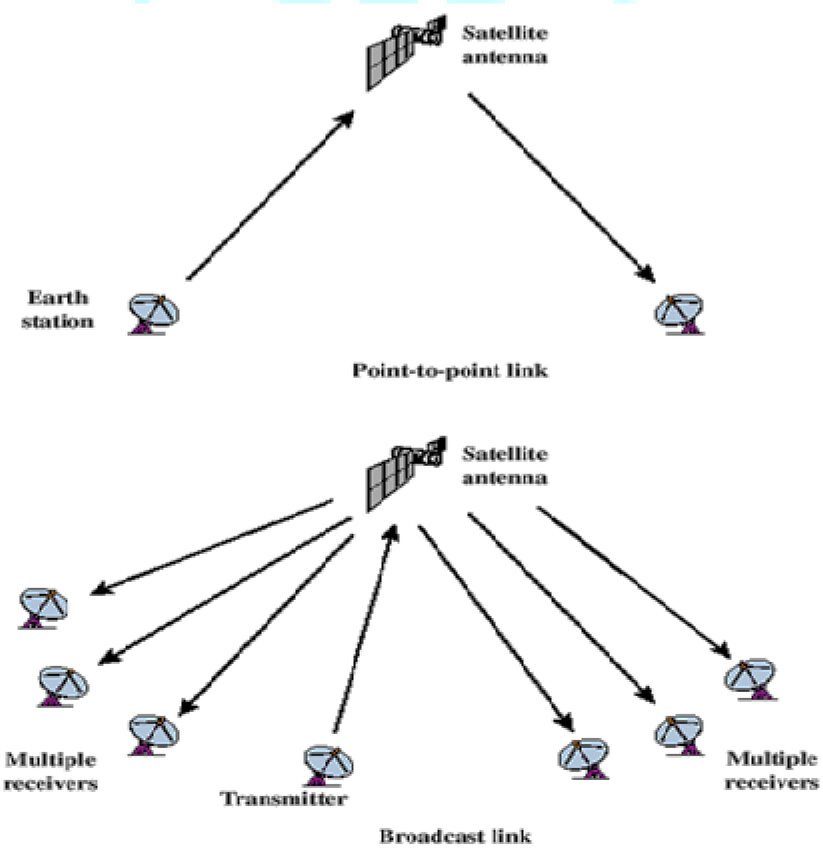
APPLICATIONS

1. Global Telecommunications : Land, Sea, Air
2. Broadcasting : Sound, TV, Multimedia, Cable TV, DTH, DTU, DBS, DVB
3. Navigation : Global Positioning System (GPS)
4. Remote Sensing – Earth Observation
5. Weather
6. Emergency Communication Services – Disasters
7. Mobile Communication Services
8. Corporate Communications – Virtual Private Network (VPN), VSAT Technology
9. Military Communications etc.,

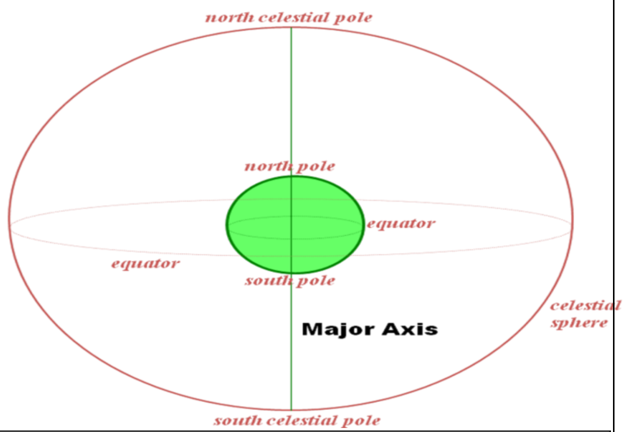
Different services of satellite systems

- **Fixed satellite services**
- **Broadcasting satellite services**
- **Mobile satellite services**
- **Navigational satellite services**
- **Meteorological satellite services.**

Satellite Communication Configurations



Satellite Orbits
An orbit is the path that a satellite follows as it revolves around Earth



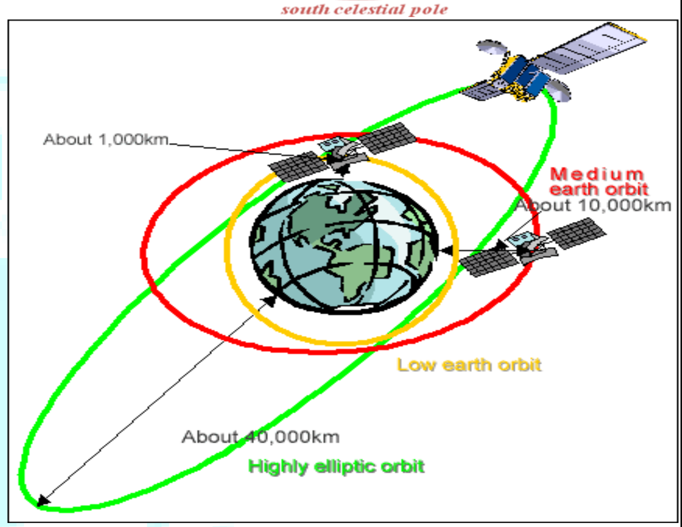
Satellites orbits vary depending on:

1) Altitude 2) Inclination 3) Orbital Period

Three classes of Satellite orbits:

1) **Low Earth Orbit (LEO)**

Up to 2,000km altitude
Remote sensing satellites, altimeter satellites



2) **Medium Earth Orbit (MEO)**

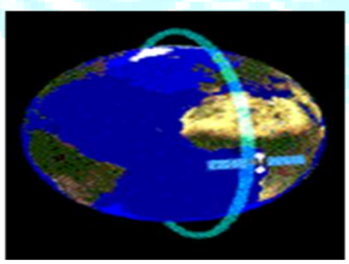
Altitudes between 5,000km – 20,000km
GPS satellites (12 hr periods – twice a day)

3) **Geostationary Earth Orbit (GEO)**

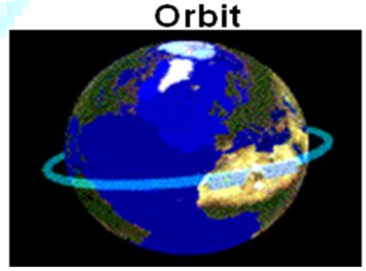
24hrs period appears fixed
Altitudes of 36,000km
Communication satellites

Molniya Orbit

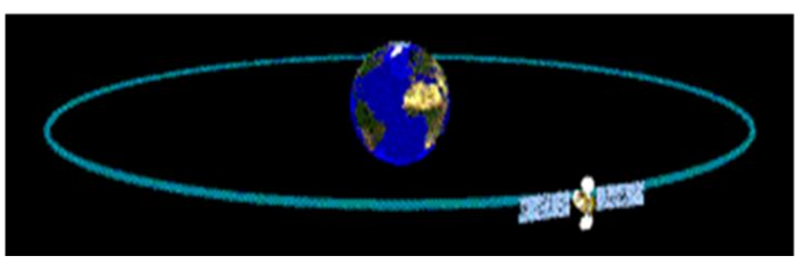
Polar Orbit



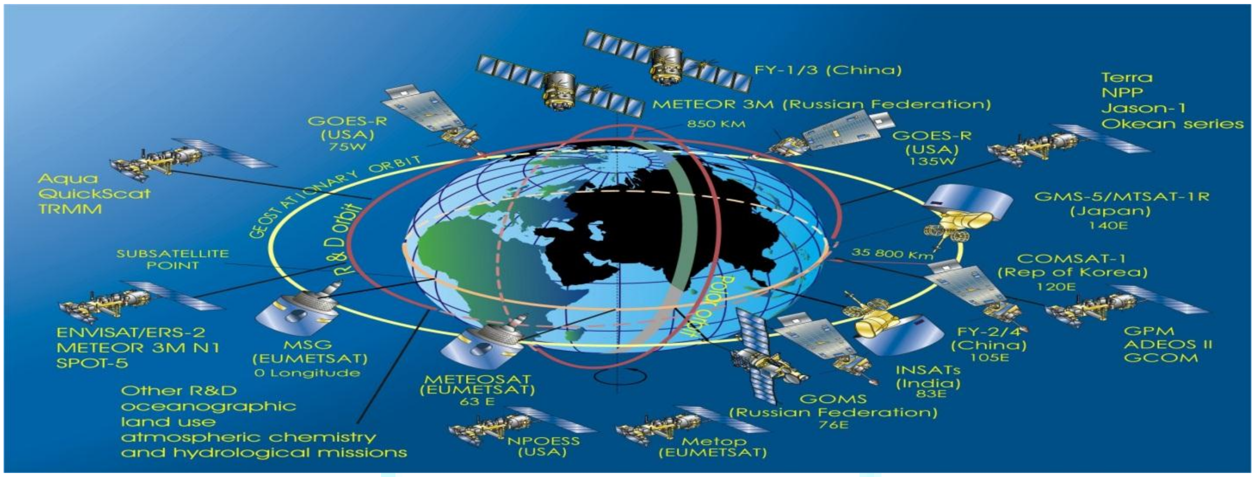
Low Earth Equatorial Orbit



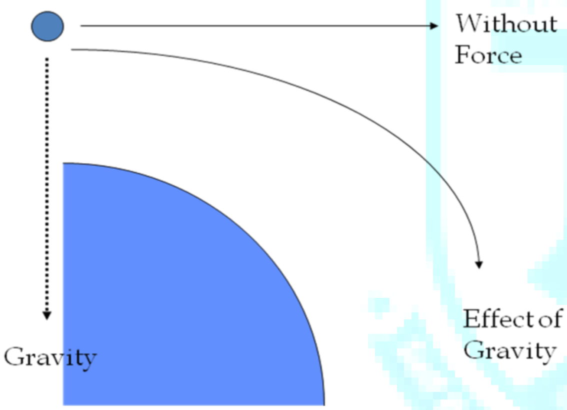
Geostationary Orbit



Motion of Space Objects



Orbital Mechanics



- Gravity depends on the mass of the earth, the mass of the satellite, and the distance between the center of the earth and the satellite

- For a satellite traveling in a circle, the speed of the satellite and the radius of the circle determine the force (of gravity) needed to maintain the orbit

- The radius of the orbit is also the distance from the center of the earth.
- For each orbit the amount of gravity available is therefore fixed
- That in turn means that the speed at which the satellite travels is determined by the orbit
- From what we have deduced so far, there has to be an equation that relates the orbit and the speed of the satellite:

$$T = 2\pi\sqrt{\frac{r^3}{4 \cdot 10^{14}}}$$

T is the time for one full revolution around the orbit, in seconds
r is the radius of the orbit in meters, including the radius of the earth (6.38×10^6m).

Common Example

- "Height" of the orbit = 22,300 mile
- That is 36,000km = 3.6×10^7m
- The radius of the orbit is 3.6×10^7m + 6.38×10^6m = 4.2×10^7m

LEO Satellite Characteristics

- Circular/slightly elliptical orbit under 2000 km
- Orbit period ranges from 1.5 to 2 hours
- Diameter of coverage is about 8000 km
- Round-trip signal propagation delay less than 20 ms
- Maximum satellite visible time up to 20 min
- System must cope with large Doppler shifts
- Atmospheric drag results in orbital deterioration

MEO Satellite Characteristics

- Circular orbit at an altitude in the range of 5000 to 12,000 km
- Orbit period of 6 hours
- Diameter of coverage is 10,000 to 15,000 km
- Round trip signal propagation delay less than 50 ms
- Maximum satellite visible time is a few hours

GEO Satellite Characteristics

- **Advantages of the GEO orbit**
    - No problem with frequency changes
    - Tracking of the satellite is simplified
    - High coverage area
- **Disadvantages of the GEO orbit**
    - Weak signal after traveling over 35,000 km
    - Polar regions are poorly served
    - Signal sending delay is substantial

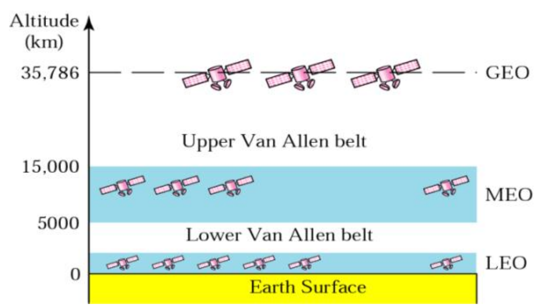
The Geosynchronous Orbit

- The answer is T = 86,000 sec (rounded)
- 86,000 sec = 1,433 min = 24hours (rounded)
- The satellite needs 1 day to complete an orbit
- Since the earth turns once per day, the satellite moves with the surface of the earth.

Comparison Chart

Features	GEO	MEO	LEO
Height (Km's)	36000	6000 - 12000	200 - 3000
Time per orbit (Hrs)	24	5 – 12	1.5
Speed (Km's / hr)	11000	19000	27000
Time Delay (ms)	250	80	10
Time in Site of Gateway	Always	2 – 4 hrs	< 15 min
Satellite for Global Coverage	3	10 – 12	50 - 70

Satellite orbit altitudes



Definitions & orbital parameters

Some of the important terms required in order to understand the orbital elements are given below:

1. **Sub Satellite Path:** This is the path traced out on the earth's surface directly below the satellite.

2. **Apogee:** The point farthest from the earth. (Apogee height -h_a)

3. **Perigee:** The point of closest approach to earth. (Perigee height -h_a)

4. **Line of apsides:** The line joining the perigee and apogee through the center of the earth.

5. **Ascending node:** The point where the orbit crosses the equatorial plane going from south to north.

6. **Descending node:** The point where the orbit crosses the equatorial plane going from north to south.

7. **Line of nodes:** The line joining the ascending and descending nodes through the center of the earth.

8. **Inclination**: The angle between orbital plane and the earth's equatorial plane. It is measured at the ascending node from equator to the orbit, going from east to north.

9. **Declination:** The angle of tilt is often referred to as the declination which must not be confused with the magnetic declination used in correcting compass readings.

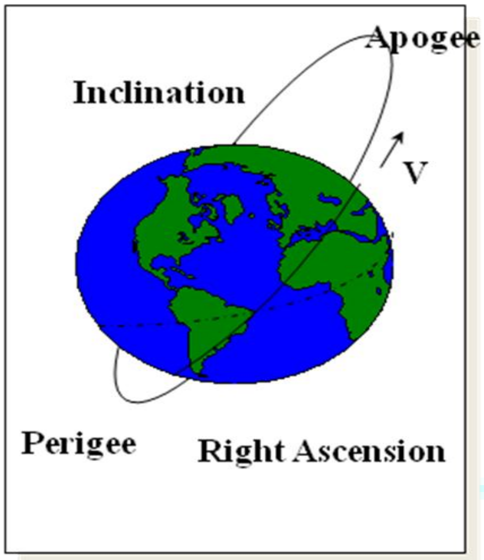
10. **Prograde orbit:** An orbit in which the satellite moves in the same direction as the earth's rotation (West to East). The prograde orbit is also known as the **direct orbit.** The inclination of a prograde orbit always lies between 0 and 90 degrees. Most satellites are launched in a prograde orbit because the earth's rotational velocity provides part of the orbital velocity with a consequent saving in launch energy.

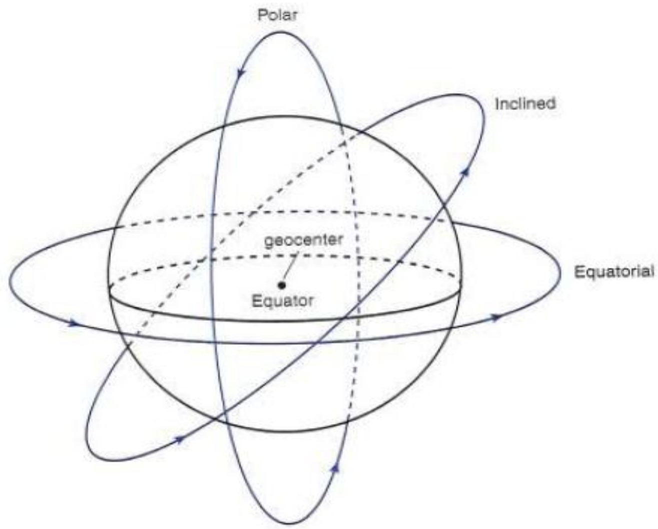
11. **Retrograde orbit:** An orbit in which the satellite moves in a direction of satellite motion.

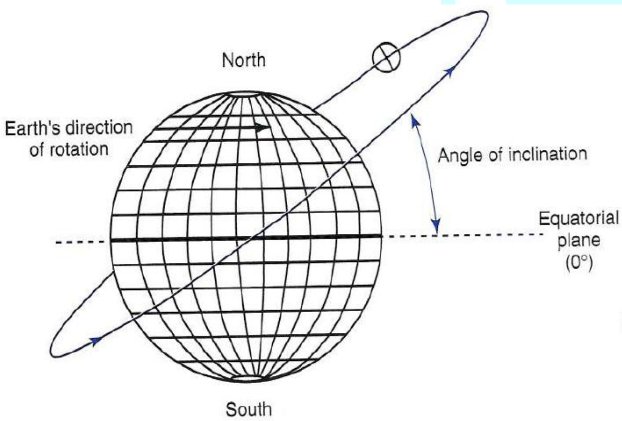
12. **Argument of perigee (w):** The angle from ascending node to perigee, measured in the orbital plane at the earth's center, in the direction of satellite motion.

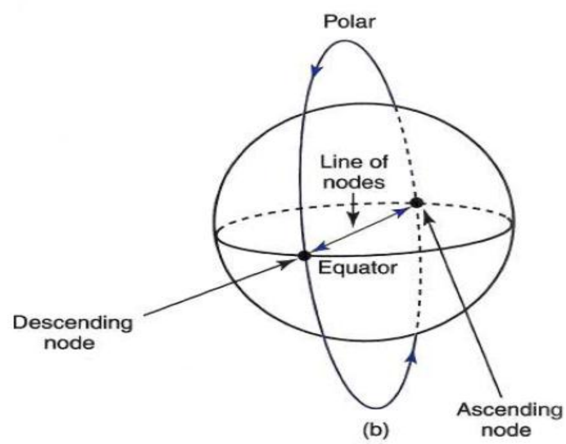
13. **Right ascension of the ascending node (Ω)** is the angle measured eastward in the equatorial plane from the line of Aries/ First point of Aries (♈) to the ascending node.

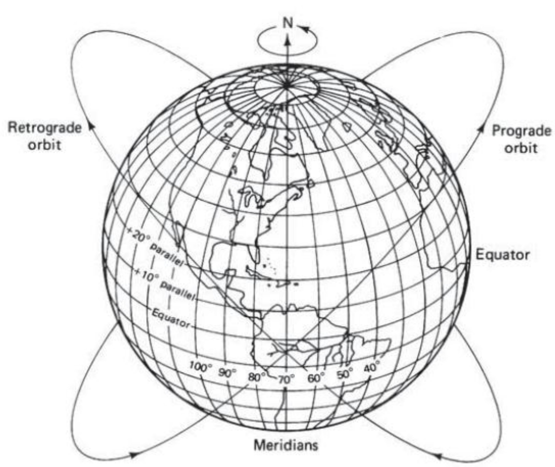
14. **Mean Anomaly** (δ_m) is an average value of the angular position of the satellite with reference to the perigee.
15. **True Anomaly** is the angle from perigee to the satellite position measured at the earth's centre.

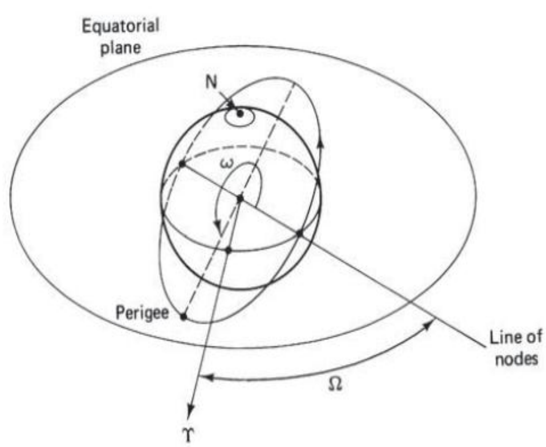

Apogee
Inclination
V
Perigee
Right Ascension


Polar
Inclined
geocenter
Equator
Equatorial


North
Earth's direction
of rotation
Angle of inclination
Equatorial
plane
(0°)
South


Polar
Line of
nodes
Equator
Descending
node
Ascending
node
(b)


N
Retrograde
orbit
Prograde
orbit
Equator
+20° parallel
+10° parallel
Equator
100° 90° 80° 70° 60° 50° 40°
Meridians


Equatorial
plane
N
ω
Perigee
Line of
nodes
Ω
ϒ

Orbital Elements

Orbital elements are the six quantities required to determine the absolute coordinates of a satellite at time t. The orbital elements are:

1. **<u>The semi major axis (a):</u>** It represents half the length of the major axis of the elliptical orbit described by the satellite.

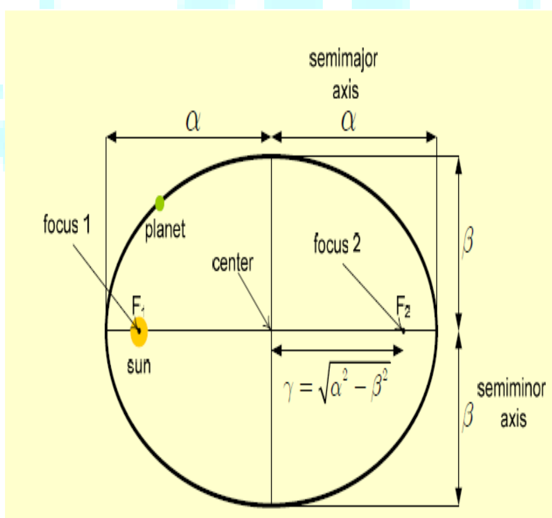
2. **<u>The eccentricity (e):</u>** Eccentricity of the ellipse.

3. **<u>The mean anomaly (M):</u>** Anomaly is the term for angle. Mean anomaly is the angle which describes the position of the satellite in its orbit relative to the perigee. At perigee, mean anomaly is zero. It increases to 180 degrees at apogee and back to perigee at 360 degrees.

4. **<u>The argument of perigee:</u>** It is the angle between the perigee points on the semi-major axis to the semi lotus rectum which joins the satellite to the centre of the earth.

5. **<u>The inclination:</u>** The angle that the orbital plane makes with the equatorial plane is called the inclination.

6. **<u>The right ascension of the ascending node:</u>** It is the angle between the vernal equinox line and the line of nodes measured in the equatorial plane.



$$\text{eccentricity} = \varepsilon = \frac{\sqrt{\alpha^2 - \beta^2}}{\alpha} = \frac{\gamma}{\alpha}$$

For an elliptical orbit, $0 < e < 1$. When $e = 0$, the orbit becomes circular.

Orbital Location:
The location of a geostationary satellite is referred to as its orbital location. International satellites are normally measured in terms of longitudinal degrees East (° E) from the Prime Meridian of 0°

Footprint
The geographic area of the Earth's surface over which a satellite can transmit to, or receive from, is called the satellite's "footprint."

Height of Apogee & Perigee:
The apogee & Perigee heights are often required. The length of the radius vectors at apogee and perigee can be obtained from the geometry of the eclipse.

$$r_a = a\,(1+e)$$
$$r_p = a\,(1-e)$$

Orbital perturbations

The orbit discussed so far is referred to as Keplerian orbit, is elliptical for the special case of an artificial satellite orbiting the earth. However, the Keplerian orbit is ideal in the sense that it assumes that the earth is a uniform spherical mass, and the only force acting is the centrifugal force, resulting from satellite motion balancing the gravitational pull of the earth.

In practice, other forces which can be significant are the gravitational forces of the sun and the moon, atmospheric drag and earth's oblate

The **gravitational pull of sun and moon** has negligible effect on low orbiting satellites, but they do affect satellites in the geo stationary orbit.

Atmospheric drag, on the other hand has negligible effect on geostationary satellites, but does affect low orbiting satellites below 1000 k.m.

Effect of a Non Spherical Earth
For a spherical earth of uniform mass, kepler's third law gives the nominal n_o as,

$$a^3 = \frac{\mu}{n^2}$$

$$n_o = \sqrt{\frac{\mu}{a^3}}$$

The 'o' subscript is included to show for perfectly spherical earth of uniform mass.

However, earth is not perfectly spherical (**Oblate Spheroid**) taken into account and mean motion n is modified as,

$$n = n_o\left[1 + \frac{k_1(1 - 1.5\sin^2 i)}{a^2(1-e^2)^{1.5}}\right]$$

k1 is constant – 66,063.1704 km^2. The earth's oblateness has negligible effect on 'a' if a is known. The mean motion is readily calculated. The orbital period taking into account the earth's oblateness is **anamalistic period** (Perigee to Perigee)

The anamalistic period is,

$$p_A = \frac{2\pi}{n}$$

n – is radians/sec. If n is known quantity we can solve for

$$n - \sqrt{\frac{\mu}{a^3}}\left[1 + \frac{k_1(1 - 1.5\sin^2 i)}{a^2(1 - e^2)^{1.5}}\right] = 0$$

The **oblateness of earth also produces two rotations** of the orbital plane is known as regression of nodes, is where the nodes appear to slide along the equator.

In effect, the line of nodes, which is in the equatorial plane, rotates about the center of the earth. Thus ̓Ω, the right ascension of the ascending node shifts its position.

If the orbit is prograde, the nodes slide westward, and if retrograde, they slide eastward. As seen from the ascending node, a satellite in prograde orbit moves eastward, and in a retrograde orbit, westward. **The nodes therefore move in a direction opposite to the direction of satellite motion, hence the term *regression of the nodes.***

The second effect is **rotation of apsides in the orbital plane**, described below. Both effects depend on the mean motion *n*, the semi major axis *a*, and the eccentricity *e*. These factors can be grouped into one factor *K* given by

$$k = \frac{nK_1}{a^2(1 - e^2)^2}$$

K will have the same units as *n*. Thus, with *n* in rad/day, *K* will be in rad/day, and with *n* in degrees/day, *K* will be in degrees/day.

An approximate expression for the rate of change of _ with respect to time is

$$\frac{d'\Omega}{dt} = -k\cos i$$

Where *i* is the inclination. The rate of regression of the nodes will have the same units as *n*.

When the rate of change given by above Equation is negative, the regression is westward, and when the rate is positive, the regression is eastward. It will be seen, therefore that for eastward regression, *i* must be greater than 90^o, or the orbit must be retrograde. It is possible to choose values of *a*, *e*, and *i* such that the rate of rotation is 0.9856°/day eastward.

The other major effect produced by the equatorial bulge is a rotation of the line of apsides. This line rotates in the orbital plane, resulting in the argument of perigee changing with time. The rate of change is given by

$$\frac{d\omega}{dt} = k(2 - 2.5\sin^2 i)$$

Again, the units for the rate of rotation of the line of apsides will be the same as those for *n* (incorporated in *K*). When the inclination *i* is equal to 63.435°, the term within the parentheses is equal to zero, and hence no rotation takes place. Use is made of this fact in the orbit chosen for the Russian Molniya satellites

Denoting the epoch time by t_0, the right ascension of the ascending node by Ω_0, and the argument of perigee by w_0 at epoch gives the new values for Ω and w at time t as

$$\Omega = \Omega_0 + \frac{d\Omega}{dt}\,(t - t_0)$$

$$\omega = \omega_0 + \frac{d\omega}{dt}\,(t - t_0)$$

Keep in mind that the orbit is not a physical entity, and it is the forces resulting from an oblate earth, which act on the satellite to produce the changes in the orbital parameters. Thus, rather than follow a closed elliptical path in a fixed plane, the satellite drifts as a result of the regression of the nodes, and the latitude of the point of closest approach (the perigee) changes as a result of the rotation of the line of apsides.

With this in mind, it is permissible to visualize the satellite as following a closed elliptical orbit but with the orbit itself moving relative to the earth as a result of the changes in Ω and w. Thus, as stated earlier, the period *PA* is the time required to go around the orbital path from perigee to perigee, even though the perigee has moved relative to the earth.

***Equatorial Ellipticity*:** In addition to the equatorial bulge, the earth is not perfectly circular in the equatorial plane; it has a small eccentricity of the order of 10^{-5}. This is referred to as the *equatorial ellipticity*. The effect of the equatorial In addition to the equatorial bulge, the earth is not perfectly circular ellipticity is to set up a gravity gradient, which has a pronounced effect on satellites in geostationary orbit. Very briefly, a satellite in geostationary orbit ideally should remain fixed relative to the earth. The gravity gradient resulting from the equatorial Ellipticity causes the satellites in geostationary orbit to drift to one of two stable points, which coincide with the minor axis of the equatorial ellipse. These two points are separated by 180°on the equator and are at approximately **75° E** longitude and **105°W** longitude. Satellites in service are prevented from drifting to these points through station-keeping maneuvers. Because old, out-of-service satellites eventually do drift to these points, they are referred to as "**satellite graveyards**." It may be noted that the effect of equatorial ellipticity is negligible on most other satellite orbits.

Atmospheric drag: For near-earth satellites, **below about 1000 km**, the effects of atmospheric drag are significant. Because the drag is greatest at the perigee, the drag acts to reduce the velocity at this point, with the result that the satellite does not reach the same apogee height on successive revolutions. The result is that the semi major axis and the eccentricity are both reduced. Drag does not noticeably change the other orbital parameters, including perigee height. An approximate expression for the change of major axis is

$$a \cong a_0 \left[\frac{n_0}{n_0 + n'_0(t - t_0)}\right]^{\frac{2}{3}}$$

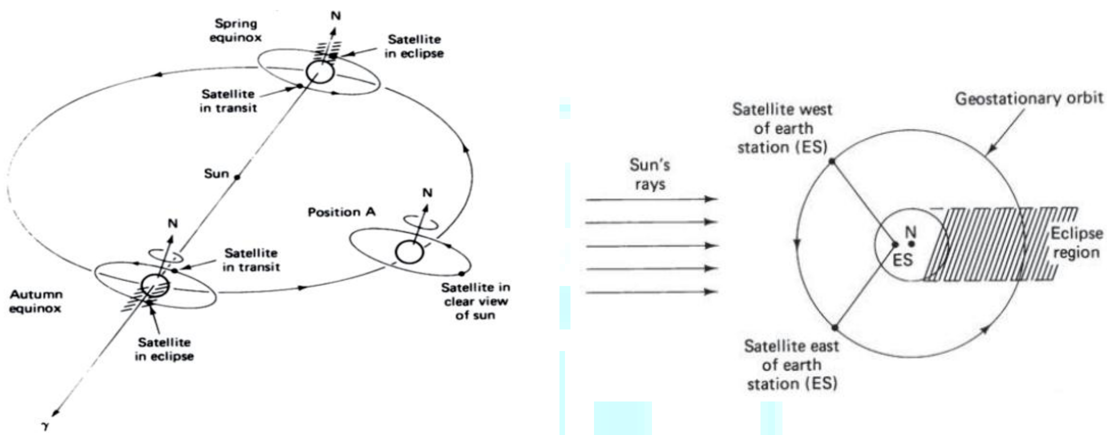
Where the "0" subscripts denote values at the reference time t_0, and n_0 is the first derivative of the mean motion. The mean anomaly is also changed, an approximate value for the change being:

$$\delta M = \frac{n'_0}{2}\,(t - t_0)^2$$

Earth Eclipse of Satellite

If the earth's equatorial plane coincided with the plane of the earth's orbit around the sun (the ecliptic plane), geostationary satellites would be eclipsed by the earth once each day. As it is, the equatorial plane is tilted at an angle of 23.4° to the ecliptic plane, and this keeps the satellite in full view of the sun for most days of the year, as illustrated by position *A* in Figure.

Around the spring and autumnal equinoxes, when the sun is crossing the equator, the satellite does pass into the earth's shadow at certain periods, these being periods of eclipse as illustrated in Figure. The spring equinox is the first day of spring, and the autumnal equinox is the first day of autumn.



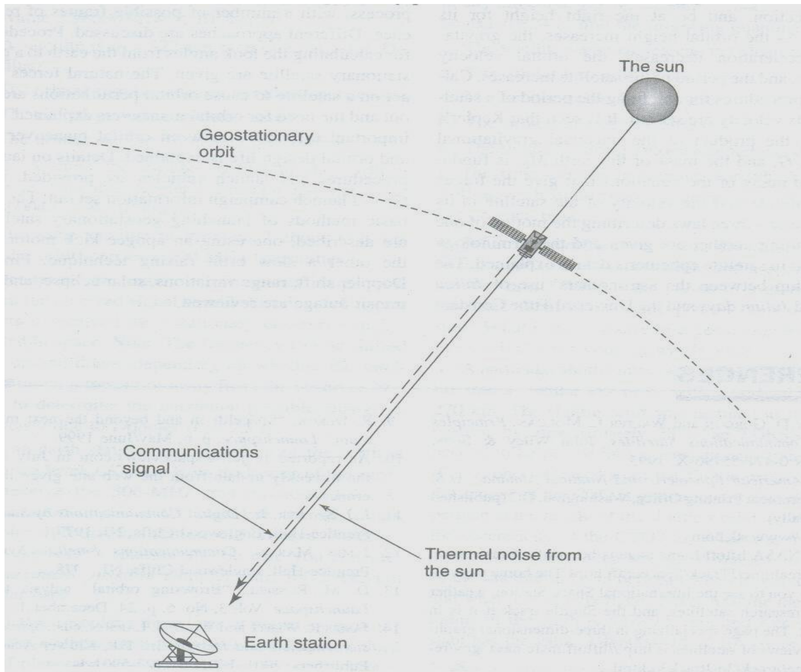
Eclipses begin 23 days before equinox and end 23 days after equinox. The eclipse lasts about 10 min at the beginning and end of the eclipse period and increases to a maximum duration of about 72 min at full eclipse (Spilker, 1977).

During an eclipse, the solar cells do not function, and operating power must be supplied from batteries. Where the satellite longitude is east of the earth station, the satellite enters eclipse during daylight (and early evening) hours for the earth station, as illustrated in Fig. This can be undesirable if the satellite has to operate on reduced battery power.

Where the satellite longitude is west of the earth station, eclipse does not occur until the earth station is in darkness, (or early morning) when usage is likely to be low. Thus satellite longitudes which are west, rather than east, of the earth station are more desirable.

Sun transit outage

During the equinox periods, the orbit of the satellite will also pass directly in front of the sun on the sunlit side of the earth. The sun is a hot microwave source with an equivalent temperature of about 6000 to 10000 K, depending on the time within the 11- year sunspot cycle, at the frequencies used by the communication satellites (4 to 50 GHz). The earth station will therefore receive not only the signal from the satellite but also the noise temperature transmitted by the sun. The added noise temperature will cause the fade margin of the receiver to be exceeded and an outage will occur. This is termed as Sun transit outage.



Sun transits occur when the sun crosses the earth's equatorial plane during the spring and fall equinoxes (late February or early March; September or October). At these times, the sun aligns directly behind the satellites for a few minutes each day. When the sun moves directly behind the satellite to your receive antenna, the satellite signal can be overwhelmed by the enormous amount of thermally generated radio frequency (RF) noise radiated by the sun. This can cause reception interference for a few minutes every day during this occurrence.

The time of occurrence depends both on the geographic location of the earth station and the location of the satellite. The sun may degrade the signal for several minutes depending on the antenna size and available link margin, although it is not unusual for the effect to go unnoticed.

The number of outages, outage duration and the time of outage depend on the radio emission activity of the sun, the movement of the earth with respect to the sun, the pointing and location of receive antenna, and characteristics of the communication system. Those characteristics, in turn, include the operating receive radio frequency, the receive antenna gain pattern, the clear sky operating carrier-to-noise ratio (C/N), the clear sky equivalent system noise temperature and the minimum acceptable C/N.

When the sun transits occur, the antenna noise temperature varies depending on the antenna size, the elevation angle, location and environment.

Computer generated predictions revealed that sun transits peak times of all days associated with an equinox for a given receive earth station do not differ from each other by more than a minute, and they do not vary much with respect to the year considered. The variations are less than one minute for at least the next decade. Thus for practical purposes, they can be considered the same and invariant with respect to the year they are considered.

Antenna Look Angles

The *look angles* for the ground station antenna are the azimuth and elevation angles required at the antenna so that it points directly at the satellite. The look angles were determined in the general case of an elliptical orbit, and there the angles had to change in order to track the satellite. With the geostationary orbit, the situation is much simpler because the satellite is stationary with respect to the earth. Although in general no tracking should be necessary, with the large earth stations used for commercial communications, the antenna beamwidth is very narrow, and a tracking mechanism is required

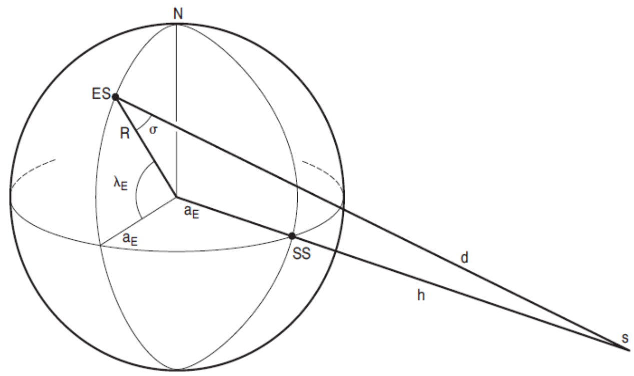
To compensate for the movement of the satellite about the nominal geostationary position. With the types of antennas used for home reception, the antenna beamwidth is quite broad, and no tracking is necessary. This allows the antenna to be fixed in position, as evidenced by the small antennas used for reception of satellite TV that can be seen fixed to the sides of homes.

The three pieces of information that are needed to determine the look angles for the geostationary orbit are

1. The earth-station latitude, denoted here by λ_E
2. The earth-station longitude, denoted here by Φ_E
3. The longitude of the subsatellite point, denoted here by Φ_{SS} (often this is just referred to as the satellite longitude)

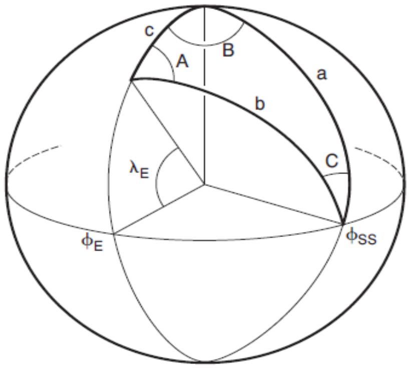
Latitudes north will be taken as positive angles, and latitudes south, as negative angles. Longitudes east of the Greenwich meridian will be taken as positive angles, and longitudes west, as negative angles. For example, if latitude of 40° S is specified, this will be taken as - 40°, and if a longitude of 35° W is specified, this will be taken as -35°.

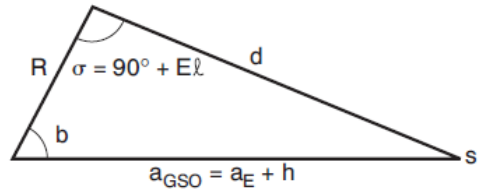
When calculating the look angles for *low-earth-orbit* (LEO) satellites, it was necessary to take into account the variation in earth's radius. With the geostationary orbit, this variation has negligible effect on the look angles, and the average radius of the earth will be used. Denoting this by *R*: R = 6371 km



The geometry involving these quantities is shown in above Figure. Here, ES denotes the position of the earth station, SS the sub satellite point, *S* the satellite, and *d* is the range from the earth station to the satellite. The angle *is* an angle to be determined.

There are two types of triangles involved in the geometry of above Figure, the spherical triangle and the plane triangle shown in heavy outline in below Figures.





Considering first the spherical triangle, the sides are all arcs of great circles, and these sides are defined by the angles subtended by them at the center of the earth. **Side *a*** is the angle between the radius to the North Pole and the radius to the subsatellite point, and it is seen that a= 90°. A spherical triangle in which one side is 90° is called a ***quadrantal triangle***. **Angle *b*** is the angle between the radius to the earth station and the radius to the subsatellite point. **Angle *c*** is the angle between the radius to the earth station and the radius to the north pole. From above Figure it is seen that c= 90°- λ_E.

There are six angles in all defining the spherical triangle. The three angles *A*, *B*, and *C* are the angles between the planes. **Angle *A*** is the angle between the plane containing *c* and the plane containing *b*. **Angle *B*** is the angle between the plane containing *c* and the plane containing *a*. From above figure $B = \Phi_E - \Phi_{SS}$. It will be shown shortly that the maximum value of *B* is 81.3°. **Angle *C*** is the angle between the plane containing *b* and the plane containing *a*. To summarize to this point, the information known about the spherical triangle is

$$\mathbf{a = 90°,\ c = 90° - \lambda_E}\ \textit{\textbf{and}}\ \mathbf{B = \Phi_E - \Phi_{SS}}$$

Note that when the earth station is west of the subsatellite point, *B* is negative, and when east, *B* is positive. When the earth-station latitude is north, *c* is less than 90°, and when south, *c* is greater than 90°. Special rules, known as ***Napier's rules***, are used to solve the spherical triangle, and these have been modified here to take into account the signed angles *B* and λ_E. Only the result will be stated here. Napier's rules gives angle *b* as

$$b = \arccos(\cos \boldsymbol{B} \cos \lambda_E)$$

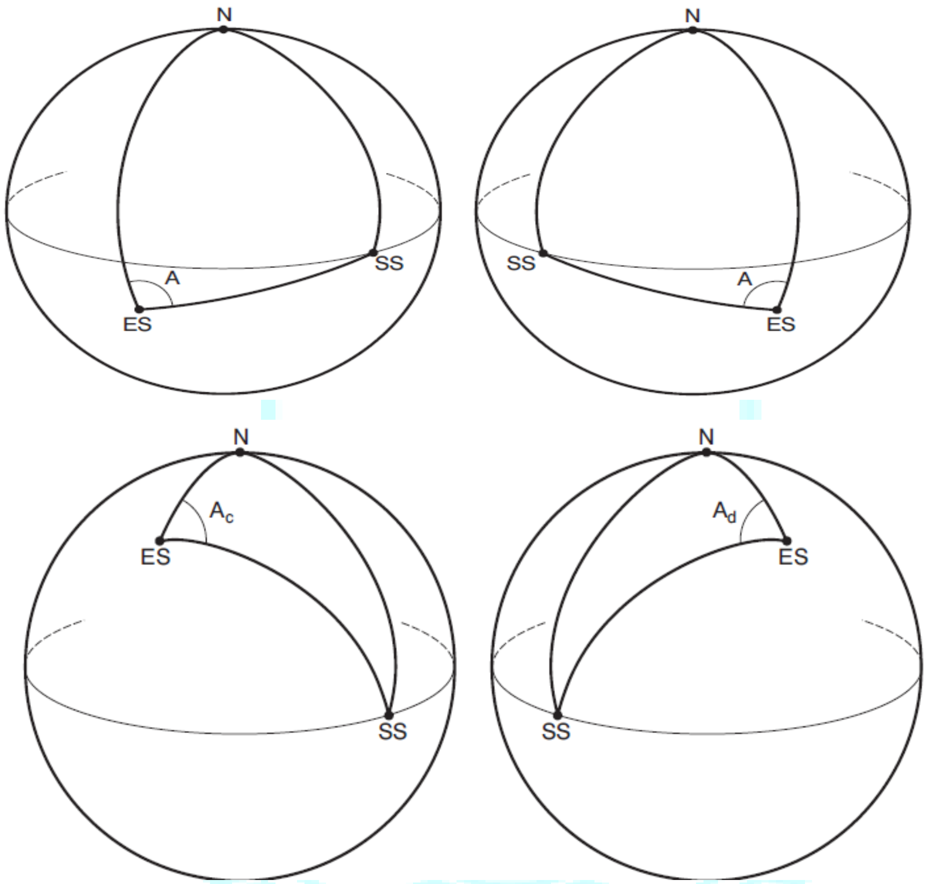
and angle *A* as

$$A = \arcsin\left(\frac{\sin|B|}{\sin b}\right)$$

Two values will satisfy the above Equation, *A* and 180°- *A*, and these must be determined by inspection. These are shown in below figures. Angle *A* is acute (less than 90°), and the azimuth angle is *Az* = *A*.

In second Figure, angle *A* is acute, and the azimuth is, by inspection, *Az* =360°- *A*. In Third Figure, angle *Ac* is obtuse and is given by *Ac*=180° - *A*, where *A* is the acute value obtained from angle A Equation. Again, by inspection, *Az* = *Ac* - 180°- *A*.

In Figure Four, angle *Ad* is obtuse and is given by 180°- *A*, where *A* is the acute value obtained from angle A Equation. By inspection, *Az* = 360° - *Ad* = 180° + *A*.



In all cases, *A* is the acute angle returned by angle A Equation. These conditions are summarized in below Table.

Angle	λ_E	B	A_z, Degrees
a	< 0	< 0	A
b	< 0	> 0	360° - A
c	> 0	< 0	180° - A
d	> 0	> 0	180° + A

Applying the cosine rule for plane triangles to the triangle Figure allows the **range *d*** to be found to a close approximation:

$$d = \sqrt{R^2 + a^2{}_{GSO} - 2Ra_{GSO}\cos b}$$

Applying the sine rule for plane triangles to the triangle Figure allows the **angle of elevation** to be found:

$$El = \arccos\left(\frac{a_{GSO}}{d}\sin b\right)$$

Example Problem:

A geostationary satellite is located at 90°W. Calculate the azimuth angle for an earth station antenna at latitude 35°N and longitude 100°W. And also find the range and antenna elevation angle.

Solution the given quantities are

Φ_{SS} = - 90° (West) **,** λ_E = 35° (North), Φ_E = - 100° (West)

$$B = \Phi_E - \Phi_{SS} = -100 + 90 = -10°$$

angle *b* as

$$b = \arccos(\cos \boldsymbol{B} \cos \lambda_E) = 36.23°$$

and angle *A* as

$$A = \arcsin\left(\frac{\sin|B|}{\sin b}\right) = 17.1°$$

azimuth is, by inspection, $\lambda_E > 0$ *and* $B < 0$*, therefore*

$$Az = 180° - A = 162.9°$$

range *d*

$$d = \sqrt{R^2 + a^2{}_{GSO} - 2Ra_{GSO} \cos b}$$

R- Earth radius = 6371 or 6378
$a_{GSO} - GEO\ Stationary\ Radius = 42164$

$$d = 37215 \text{ km}$$

Angle of elevation

$$El = \arccos\left(\frac{a_{GSO}}{d} \sin b\right) = 48°$$

The Polar Mount Antenna

Where the home antenna has to be steerable, expense usually precludes the use of separate azimuth and elevation actuators. Instead, a single actuator is used which moves the antenna in a circular arc. This is known as a *polar mount antenna.*

$$\textit{Angle of tilt: } \delta = 90° - \mathrm{El}_0 - \lambda_E \textit{ and } b = \lambda_E$$

$$El = \arccos\left(\frac{a_{GSO}}{d} \sin \lambda_E\right)$$

Example:

Determine the angle of tilt required for a polar mount used with an earth station at latitude 49° north. Assume a spherical earth of mean radius 6371 km, and ignore earth-station altitude

range *d*

$$d = \sqrt{R^2 + a^2{}_{GSO} - 2Ra_{GSO}\cos b} = 38287 \text{ km}$$

Angle of elevation

$$El = \arccos\left(\frac{a_{GSO}}{d}\sin\lambda_E\right) = 33.8°$$

Angle of tilt:

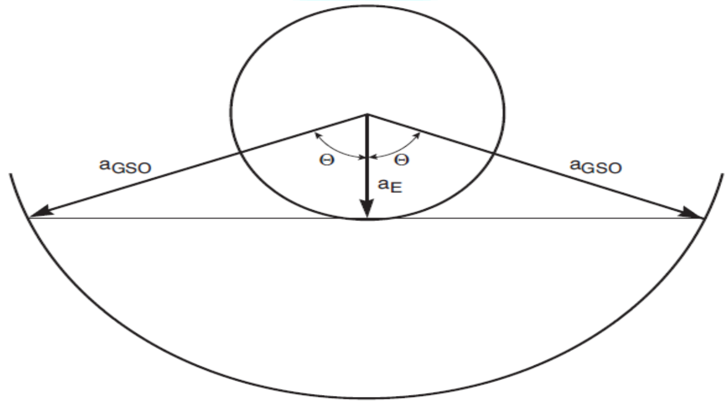
$$\delta = 90° - \mathrm{El}_0 - \lambda_E = 7°$$

Limits of Visibility

There will be east and west limits on the geostationary arc visible from any given earth station. **The limits will be set by the geographic coordinates of the earth station and the antenna elevation.** The lowest elevation in theory is zero, when the antenna is pointing along the horizontal. A quick estimate of the longitudinal limits can be made by considering an earth station at the equator, with the antenna pointing either west or east along the horizontal, as shown in Figure. The limiting angle is given by

$$\theta = \arccos\frac{a_E}{a_{GSO}}$$

$$\theta = \arccos\frac{6378}{42164} = 81.3°$$



Thus, for this situation, an earth station could see satellites over a geostationary arc bounded by $\pm$ 81.3° about the earth-station longitude.

In practice, to avoid reception of excessive noise from the earth, some finite minimum value of elevation is used, which will be denoted here by El_{min}. A typical value is 5°. The limits of visibility will also depend on the earth-station latitude. As in Figure, let S represent the angle **subtended** at the satellite when the angle $\boldsymbol{\sigma_{min} = 90° - El_{min}}$. Applying the sine rule gives

$$S = \arcsin\left(\frac{R}{a_{GSO}}\sin\sigma_{\min}\right)$$

A sufficiently accurate estimate is obtained by assuming a spherical earth of mean radius 6371 km as was done previously. Once angle S is known, angle b is found from

$$b = 180 - \sigma_{\min} - S$$

$$B = \arccos\left(\frac{\cos b}{\cos \lambda_E}\right)$$

Example:

Determine the limits of visibility for an earth station situated at mean sea level, at latitude 48.42° north, and longitude 89.26 degrees west. Assume a minimum angle of elevation of 5°.

Given Data:

$$\lambda_E = 48.42°, \Phi E = -89.26°, El\min = 5°$$

R- Earth radius = 6371 or 6378

$a_{GSO} - GEO\ Stationary\ Radius = 42164$

$$\sigma_{min} = 90° - El_{min} = 95°$$

$$S = \arcsin\left(\frac{R}{a_{GSO}}\sin\sigma_{\min}\right) = 8.66°$$

$$b = 180 - \sigma_{\min} - S = 76.34°$$

$$B = \arccos\left(\frac{\cos b}{\cos \lambda_E}\right) = 69.15°$$

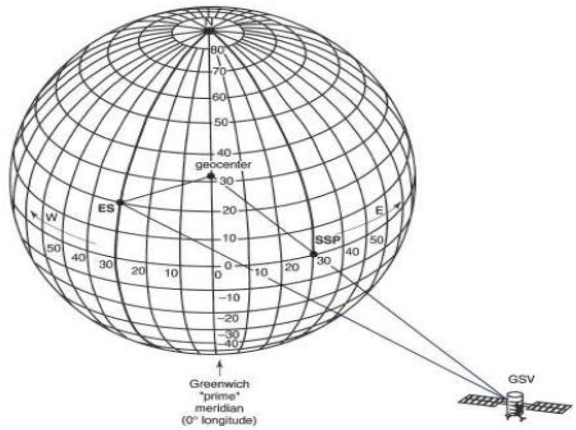
The satellite limit east of the earth station is at

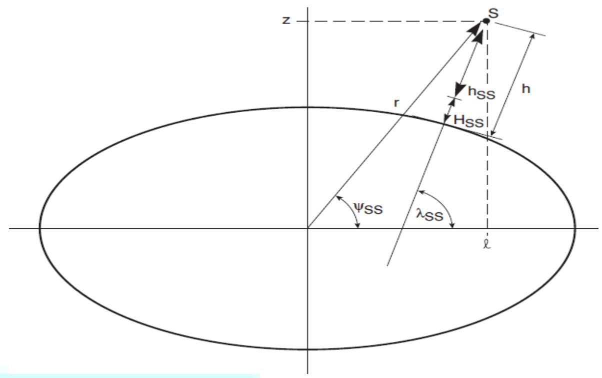
$$\Phi E + B \approx -20°$$

and west of the earth station at

$$\Phi E - B \approx -158°$$

Sub Satellite Point





The point on the earth vertically under the satellite is referred to as the *subsatellite point*. The latitude and longitude of the subsatellite point and the height of the satellite above the subsatellite point can be determined from knowledge of the radius vector **r**. The above Figure shows the meridian plane which cuts the subsatellite point. The height of the terrain above the reference ellipsoid at the subsatellite point is denoted by *H*SS, and the height of the satellite above this, by *h*SS. Thus the total height of the satellite above the reference ellipsoid is

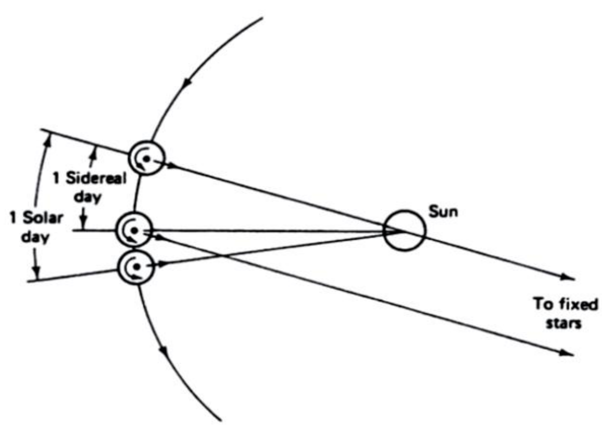
$$h = H_{ss} + h_{ss}$$

Sidereal Time:

1. Sidereal time is the time measured relative to the fixed stars.
2. Sidereal day is defined as one complete rotation of the earth relative to the fixed star.

1 mean solar day = 1.0027379093 mean sidereal days
= $24^h\ 3^m\ 56^s.55536$ sidereal time
= 86,636.55536 mean sidereal seconds.

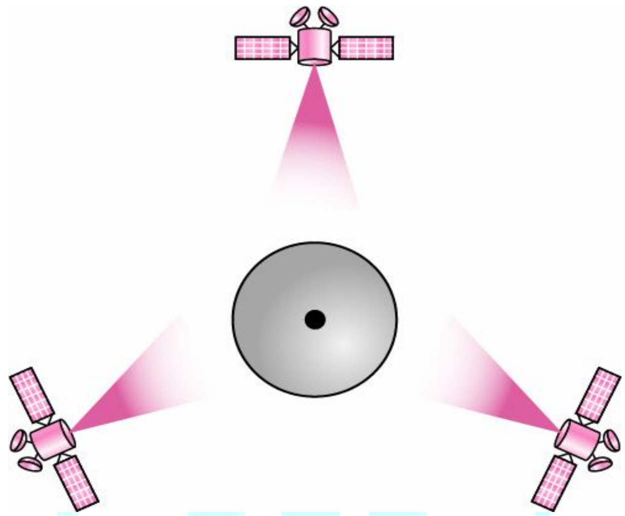
1 mean sidereal day = 0.9972695664 mean solar days
= $23^h\ 56^m\ 04^s.09054$ mean solar time
= 86,164.09054 mean solar seconds.



Geostationary Orbit

1. A satellite in a geostationary orbit appears to be stationary with respect to the earth.
2. Three conditions are required for an orbit to be stationary.
   - The satellite must travel eastward at the same rotational speed as the earth.
   - The orbit must be circular.
   - The inclination of the orbit must be 0.
3. The radius of the geostationary orbit=42164km.

Satellites in geosynchronous orbit



Uses of Geostationary Orbits

Geostationary orbits are primarily used for two functions:

- Weather monitoring
- Telecommunications & Broadcasting
  - Commercial growth is focused on:
    - DTH TV (Direct To Home: Sky TV)
    - Phone, Fax, Video, Data services
    - Mobile Communications
    - VSAT & USAT
    - Digital Radio

How to File for a Geo Position

- Only Allocated to National Governments
- Go to National Government
  - Request Orbital Position (s)
    - US Companies (Non Governmental Entities) work through FCC
    - UK through UK Radiocommunications Agency
- Prepare ITU Paperwork
- File & Coordinate
  - First Come, First Served = Priority!

Characteristics of Geostationary (GEO) Orbit Systems

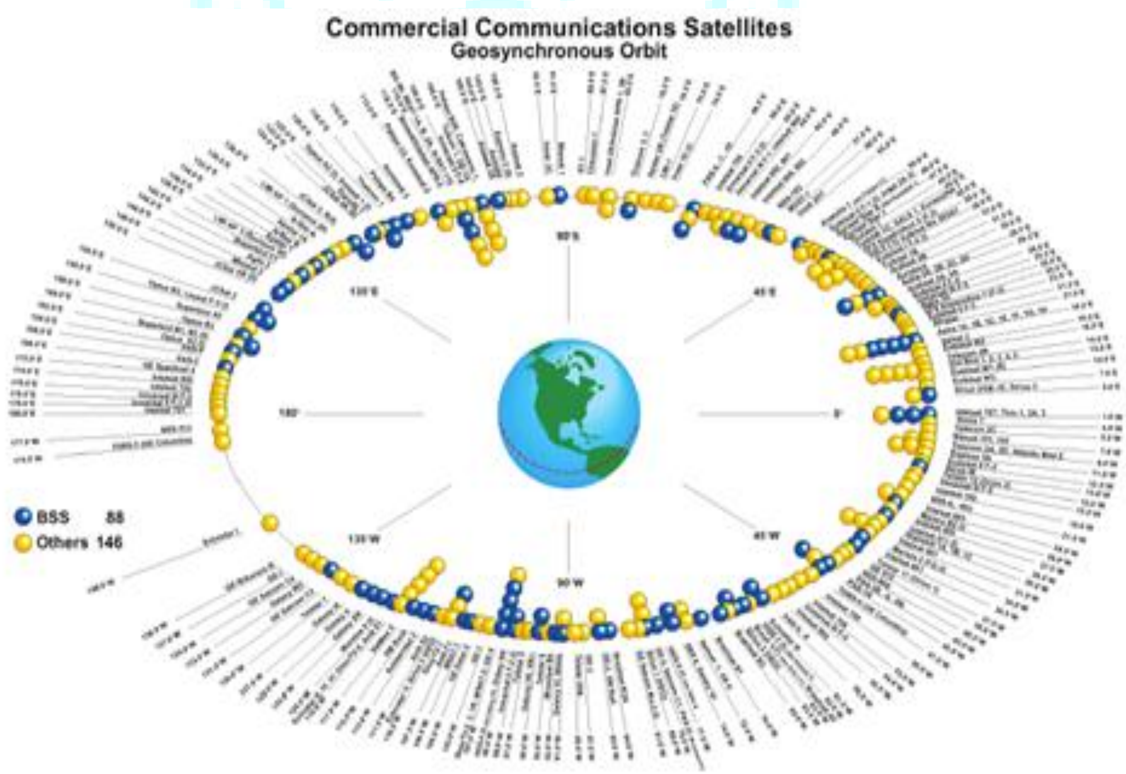
- User terminals do not have to track the satellite
- Only a few satellites can provide global coverage
- Maximum life-time (15 years or more)
- Above Van Allen Belt Radiation
- Often the lowest cost system and simplest in terms of tracking and high speed switching

Features	GEO
Height (Km's)	36000
Time per orbit (Hrs)	24
Speed (Km's / hr)	11000
Time Delay (ms)	250
Time in Site of Gateway	Always
Satellite for Global Coverage	3

Challenges of Geostationary (GEO) Orbit

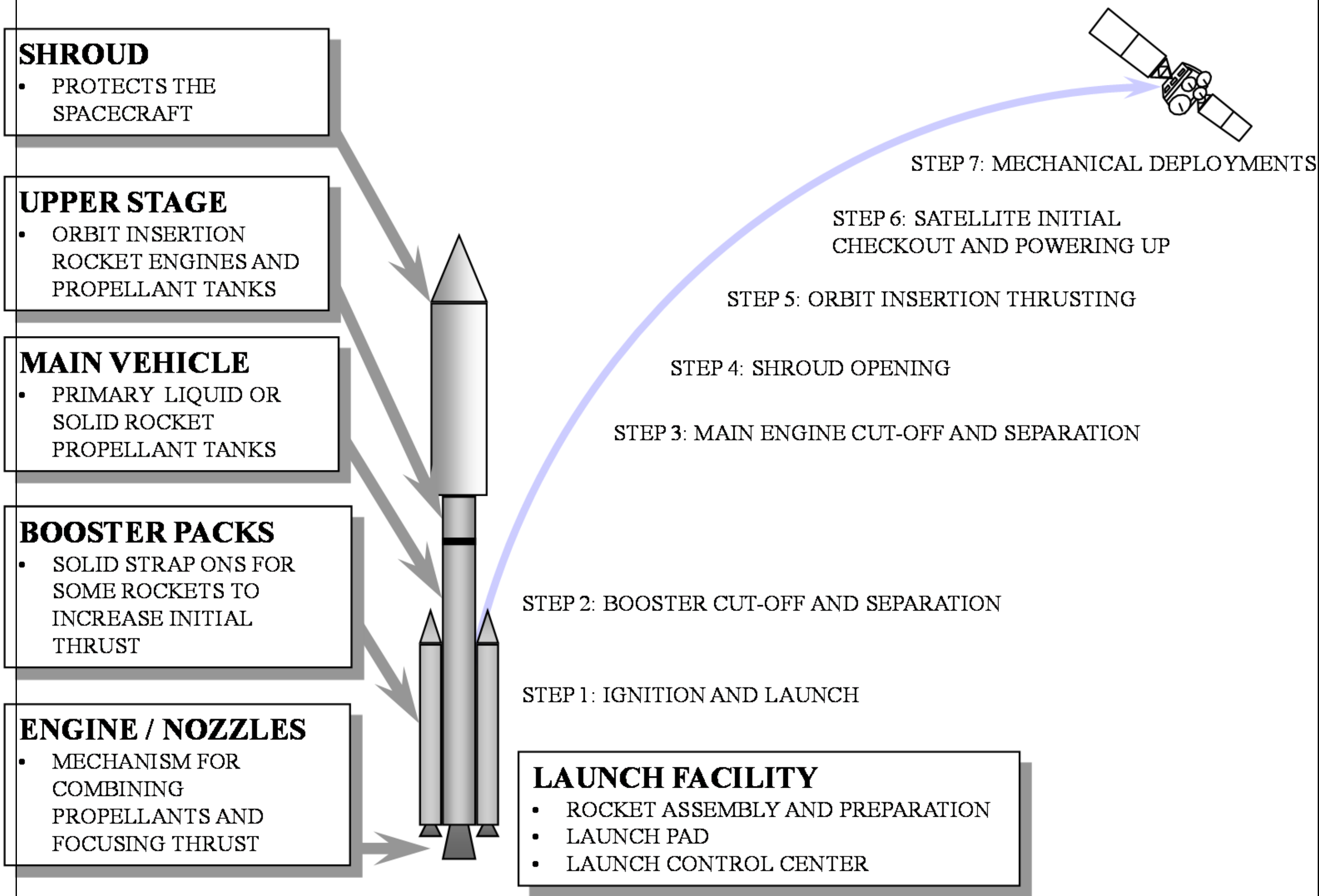
- Transmission latency or delay of 250 millisecond to complete up/down link
- Satellite antennas must be of larger aperture size to concentrate power and to create narrower beams for frequency reuse
- Poor look angle elevations at higher latitudes

Geostationary Orbit Today



LAUNCH SYSTEM CONCEPTS

Satellites may be *directly injected* into low-altitude orbits, up to about 200 km altitude, from a



launch vehicle. Launch vehicles may be classified as *expendable* or *reusable*.

In addition shuttle was designed with the capability to retrieve satellite in low orbit. The shuttle consists of a reusable orbiter which injects satellites to a LEO and re-enters the atmosphere, landing as an aircraft. The orbitter itself is launched vertically with the help of two recoverable solid rocket boosters. An expendable liquid hydrogen/ liquid oxygen tank furnishs propellant to the three main engines. This tank is the only part of the shuttle that is not reused.

The shuttle can only launch satellite in LEO and therefore additional propulsion is necessary to inject a satellite into the GEO. The heavy lift capability of the shuttle is effectively used for carrying shuttle upper stages for the extra propulsion. Various types of upper stage have been developed. These may be categorized as perigee stages and Integrated stages.

Typical of the expendable launchers are the U.S. Atlas-Centaur and Delta rockets and the European Space Agency Ariane rocket. Japan, China, and Russia all have their own expendable launch vehicles, and one may expect to see competition for commercial launches among the countries which have these facilities. Until the tragic mishap with the Space Shuttle in 1986, this was to be the primary transportation system for the United States. As a reusable launch vehicle, the shuttle, also referred to as the *Space Transportation System* (STS), was planned to eventually replace expendable launch vehicles for the United States (Mahon and Wild, 1984).

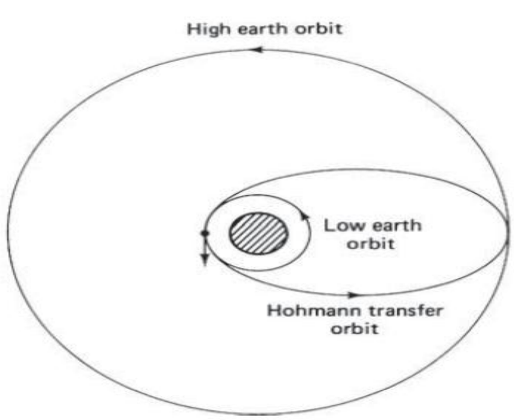
Three methods of launching a satellite: (1) Using apogee kick motor
(2) Using spacecraft thrusters
(3) Direct insertion to GEO

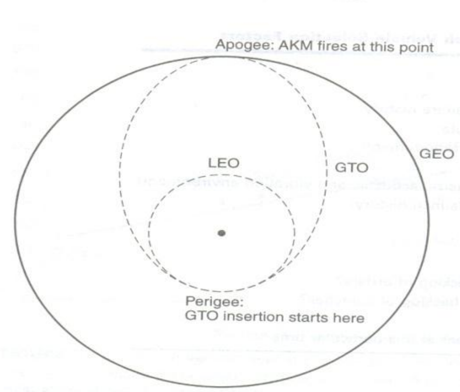
<u>Using spacecraft thrusters</u>
This method is employed since apogee kick motor imparts a vigorous acceleration. Spacecraft thrusters are used to raise the orbit from GTO to GEO over a number of turns. There are two thrusters on a satellite, one for more powerful orbit raising maneuvers and the other for on-orbit maneuvers.

<u>Direct insertion to GEO</u>
Here the final stages of the rocket are used to place the satellite directly into GEO rather than the satellite using its own propulsion system to go from GTO to GEO.

Where an orbital altitude greater than about 200 km is required, it is not economical in terms of launch vehicle power to perform direct injection, and the satellite must be placed into transfer orbit between the initial LEO and the final high-altitude orbit.

In most cases, the transfer orbit is selected to minimize the energy required for transfer, and such an orbit is known as a *Hohmann transfer* orbit. The time required for transfer is longer for this orbit than all other possible transfer obits. Assume for the moment that all orbits are in the same plane and that transfer is required between two circular orbits, as illustrated in Figure. The Hohmann elliptical orbit is seen to be tangent to the low altitude orbit at perigee and to the high-altitude orbit at apogee. At the perigee, in the case of rocket launch, the rocket injects the satellite with the required thrust into the transfer orbit. With the Space Transportation System (STS), the satellite must carry a perigee kick motor which imparts the required thrust at perigee. At apogee, the *apogee kick motor* (AKM) changes the velocity of the satellite to place it into a circular orbit in the same plane. As shown in Figure, it takes 1 to 2 months for the satellite to be fully operational.





The total energy, U of a satellite, for a two body system, is given by

$$U = \frac{1}{2}\, mv^2 - \frac{GmM}{r}$$

Where m and v are the mass and velocity of the satellite respectively, and r is the distance from geocenter, G is the gravitational constant and M the mass of the earth. To achieve a geostationary orbit, the launch vechicle must be able to impart a velocity of 3070m/s at the GEO orbit height about 42165 km from the earth. In an equipotential field, the maximum velocity increment, Δv, which a launch vehicle of total mass m_0 can impact may be given as

$$\Delta v = v_g\, ln\left(\frac{1}{1 - \frac{m_f}{m_0}}\right)$$

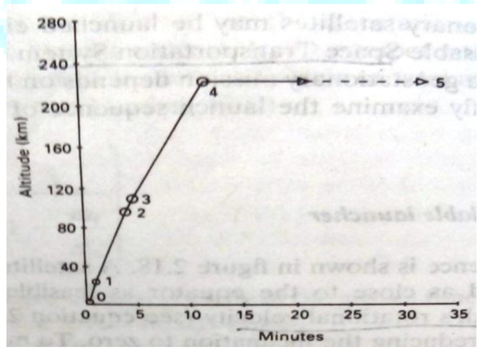
v_g = effective exhaust velocity of the gas, m_f = mass of the expanded fuel

In order to maximize Δv the ratio $\frac{m_f}{m_0}$ should be maximized. Therefore it is usual to launch satrllites by means of multiple stage rockets, each stage being jettisoned after imparting a given thrust. As m_o is progressively reduced, succeeding stages of rockets need to impact progressively lower thrust to achieve the desired orbit. The final velocity of the spacecraft is the sum of the velocity increments of all the stages.

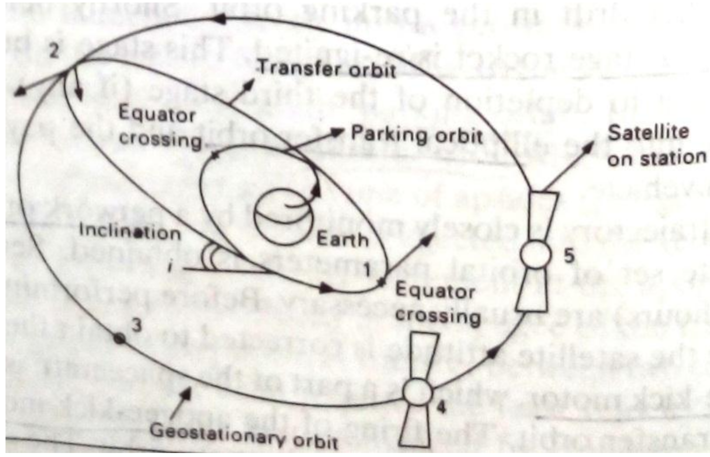
Orbital inclination is determined by the latitude of the launch station and is given by

$$\cos(i) = \sin(\in_1)\cos(\theta_1)$$

Where i – inclination, $\in_1 - azimuth\ of\ launch,\ \theta_1 - latitude\ of\ launch\ site.$



Event Number	Event
0	Vertical lift off
1	Guidance system begins tilting rocket towards east
2	First stage drop off
3	Second stage ignition
4	Horizontal insertion into parking orbit 185 to 250 km
5	Second and third stages fired at equator to acquire transfer orbit



Event Number	Event
1	Velocity increment to acquire transfer orbit, satellite spun for stabilization, attitude maneuvers done before apogee kick motor firing
2	Apogee kick motor fired to give necessary velocity increment, orbit ciculaized and inclination reduced to near zero
3	Satellite despun
4	Three axis stabilization acquired
5	Minor orbit errors in orbit tested and position satellite on station

Throughout the launch and acquisition phases, a network of ground stations, spread across the earth, is required to perform the ***tracking, telemetry, and command*** **(TT&C)** functions. Velocity changes in the same plane change the geometry of the orbit but not its inclination. In order to change the inclination, a velocity change is required normal to the orbital plane. Changes in inclination can be made at either one of the nodes, without affecting the other orbital parameters. Since energy must be expended to make any orbital changes, a geostationary satellite should be launched initially with as low an orbital inclination as possible. It will be shown shortly that the smallest inclination obtainable at initial launch is equal to the latitude of the launch site. Thus the farther away from the equator a launch site is, the less useful it is, since the satellite has to carry extra fuel to effect a change in inclination. Russia does not have launch sites south of 45°N, which makes the launching of geostationary satellites a much more expensive operation for Russia than for other countries which have launch sites closer to the equator.

Prograde (direct) orbits have an easterly component of velocity, so prograde launches gain from the earth's rotational velocity. For a given launcher size, a significantly larger payload can be launched in an easterly direction than is possible with a retrograde (westerly) launch. In particular, easterly launches are used for the initial launch into the geostationary orbit. The relationship between inclination, latitude, and azimuth may be seen as follows [this analysis is based on that given in Bate et al. (1971)].

Launch Window: before the launch of a satellite is necessary to ensure that the launch time falls within a "launch window". This guarantees that the position of a satellite in respect of the sun is favorable thus ensuring adequate power supply and thermal control throughout the mission.

Further, the launch must be so timed that the satellite is that the satellite is visible to the control station during all the critical maneuvers. This set of conditions limits the launch time to certain specified intervals, designed the launch window.

Propellant types:

- Chemical propulsion
- Solid propulsion

Consideration of chemical propulsion

Chamber Condition:

- Mass Balance
- Enthalpy Balance (Conversion of energy)
- Pressure Balance (Daltons Law)
- Chemical Equilibirium (Minimization of Gibbs Formula)

Rocket Performance:

- Equation of state (Ideal Gas law)
- Continuity Equation (Conversion of mass)
- Conversion of momentum
- Conversion of energy (First law of thermodynamics)
- Isotropic Expansion (Second law of thermodynamics)

Launch Vehicles

Arian:

- Arian Space – European consortium with European space agency and french space agency
- Initiated in 1973, launch took place in 1979.
- Payload capacity of 1850 kg
- Ariane 2 & Ariane 3 has initiated in 1980
- Ariane 2 has a 2175 kg capacity into GTO (Geo Transfer orbit) & Ariane 3 has 2700 kg capacity.

ATLAS:

- The atlas launch system is the product of the general dynamics space system division
- Atlas I, II & II A
- Payload capacity of 2500 kg

DELTA:

- Delta I & II
- Managed by McDonnell Douglay for NASA
- Launched in 1960
- 2 stage mission & 3 stage mission

M- I & M – II:

- Launch vechicle of the national space development agency & Japan
- M-I has 3 stages, consisting of two liquid propellant stages and a composite solid propellant 3 stage.
- Operational in 1988
- M – II has 2 stage liquid oxygen & liquid hydrogen rocket augmented by two solid

Orbit	M-I	M-II
GTO	1100	400
LEO 30 degree, 300 km	2900	9200
Sun synchronous, 700 km	1400	5000
Lunar or planetary	1400	2500

Long March:

- Developed by china initiated in 1956
- Long march 1,2,3,4 & 2E vehicles
- Since 1970, launches have placed more than 30 satellite into orbits
- E the most powerful in current production has a payload capability of 93000 kg into LEO and 3370 kg into GTO.

PROTON:

- In Russia proton was developed between 1961 & 1965
- The proton has a capability of 20000 kg into LEO, 5500 kg into GTO, 2800 kg into sun synchronous orbit.

SLV/ GSLV/ PSLV – ASSIGNMENT