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.

Kepler's Laws, Newton's law

Orbital Parameters

Orbital Perturbations

Station Keeping

GeoStationary And Non Geo-Stationary Orbits

Near GeoStationary orbit

Look Angle Determination

Limits of Visibility

Eclipse and Sun transit outage

Subsatellite point

Launching Procedures

Launching Orbits

Launch vehicles

Launching of geostationary satellite

Launch Vehicles And Propulsion

Principle of Rocket Population

The Rocket Equation

Thrust

Launch from an expandable Launcher

Launch from an expandable Launcher

Launch from a Reusable Launcher

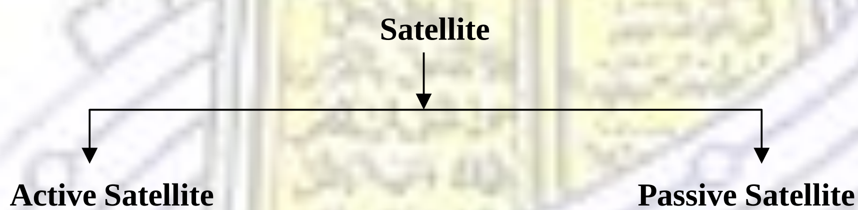
Launching of geostationary satellite

Introduction:-

- **Satellite:-**
  - ✓ In astronomical terms, a satellite is a celestial body that orbits a round a planet.
    Example: The moon is a satellite of Earth.
  - ✓ In aerospace terms, a satellite a space vehicle launched by humans and orbits earth (or) another celestial body.

- **Communications Satellite:** It is a microwave repeater in the sky that consists of a diverse combination of one or more components including transmitter, receiver, amplifier, regenerator, filter onboard computer, multiplexer, demultiplexer, antenna, waveguide etc.
- A satellite radio repeater is also called ***transponder***. This is usually a combination of transmitter and receiver.

- **Types:-**
  Generally satellite is classified into types.
  (a) Natural satellite Moon is an example for Natural Satellite
  (b) Artificial satellite Man-made satellite



- **Active Satellite:-**
  - ✓ Active Satellite employ ‘Regenerative technology’ which consists of demodulation, reshaping, regenerating, processing, frequency translation, switching and amplification processes.
  - ✓ Block used for this purpose is called transponder.
  - ✓ It is also called as Microwave repeaters
    Example: Commutation Satellite

- **Passive Satellite:-**
  - ✓ Passive Satellite is a passive reflector which reflects signals back to earth as there are no gain devices on based to amplify the signals.
  - ✓ They are just used to link two stations through space and they are not useful for communication application.

- **Satellite system:-**
  Satellite system consists of one (or) more satellites, a ground – based station to control the operation of the system and uses network earth stations that presides the interface facilities for the transmission and reception of terrestrial communications traffic.

- **How a satellite works?**
  - ✓ One Earth station transmits the signals to the satellite at ***Uplink frequency***. Up link frequency is the frequency at which Earth station is communicating with a satellite.
  - ✓ The satellite transponder process the signal and sends it to the second Earth station at another frequency called ***downlink frequency***.

- **Advantages:-**

The advantages of satellite communication over terrestrial communication are

1. Large coverage area.
2. Higher Bandwidths are available for use.
3. Transmission cost of a satellite is independent of the distance from the center of the coverage area.
4. Satellite to satellite communication is very precise.

- **Disadvantages:-**

1. Launching satellite into orbit is costly.
2. Satellite Bandwidth is gradually becoming used up.
3. Larger propagation delay.

- **Frequency Allocations for satellite services:-**

According to ITU – International Telecommunication Union world is divided into three regions.

Region 1	:	Europe, Africa, Soviet Union, Mongolia.
Region 2	:	North and South America, Greenland
Region 3	:	Asia, Australia, South West Pacific.

- **Services provided by satellite:-**

1. Fixed satellite services (FSS) – for telephone Network and television.
2. Broadcasting satellite services (BSS) – Direct broadcast to home (DBS (or) DTH)
3. Mobile satellite services – Land Mobile, Maritime Mobile and Aeronautical Mobile.
4. Navigational satellite service – Service to Global positioning systems (GPS)
5. Meteorological satellite service – Search and receive operations.

TABLE 1. 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

1.1 Kepler's Laws, Newton's law

1. **Explain how Kepler's and Newton's laws are used to describe the orbit.(April 2014,Nov/Dec 2011,Nov/Dec 2010,April/May 2010,Nov/Dec 2009,May/June 2009)**
2. **Explain the three laws of Kepler with diagrams. [Dec 2021]**
3. **An ophthalmology department is planning to perform CATARACT surgery for patients through experts using a satellite link. How kepler's law of planetary motion support in launching a satellite for such applications? Discuss the conceptual view. [May 2022]**

Kepler's Laws:-

- In the early seven tenth century, German Astronomer Johannes kepler (1571 – 1630), discovered the laws that govern satellite motion.
- The laws of planetary motion describe the shape of the orbit, the velocities of the planet and the distance a planet is with respect to the sun.
- Kepler's laws can be ***applied to any two bodies in space that interact through gravitation***.
- The larger of the 2 bodies is called "Primary" and the smaller is called "Secondary (or) Satellite".

- **Kepler First Law:-**

Kepler's first law states that the path followed by a satellite around the primary (like earth) will be an ellipse.

✓ Ellipse as two focal points F_1 & F_2.

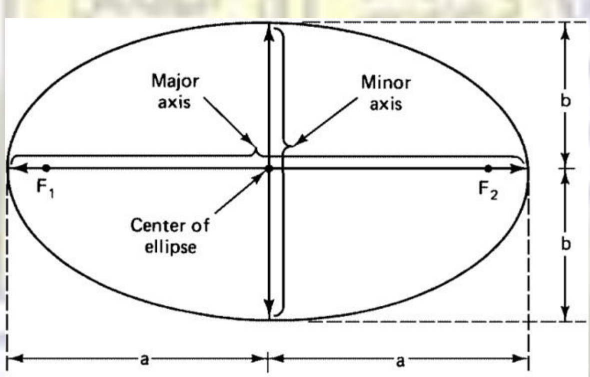


Figure1.1. The foci $\mathbf{F_1}$ and $\mathbf{F_2}$, the semimajor axis ***a***, and the semiminor axis ***b*** of an ellipse.

✓ The Centre of mass of two body system called 'barycentre' is always centered on one of the foci.

✓ Since from two body, earth has enormous mass, centre is located at centre of earth itself (one of the foci)

Where, a → Semi major axis

b→ Semi minor axis

$$\therefore \text{Eccentricity, } e = \frac{\sqrt{a^2 - b^2}}{a} \rightarrow (1.1)$$

- for elliptical orbit, 0 < e < 1
- if e = 0, then the orbit is a circular

- **Kelper's second law:-**

✓ Kelper's second law is known as the "Law of areas".

- ***Kelper's second law*** states that, "for equal intervals of time, a satellite will sweep out equal areas in the orbital plane, focused at the bary center".

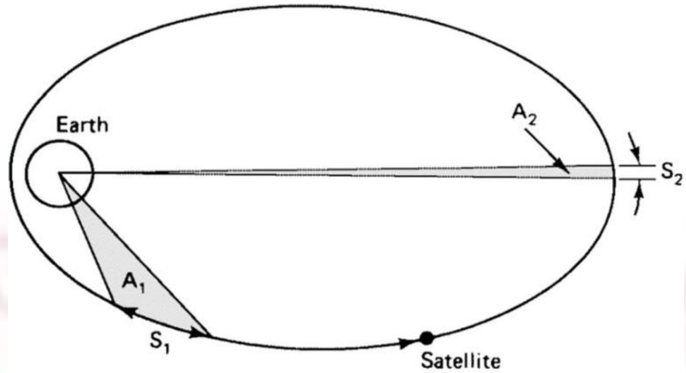


Figure1.2 Kepler's second law. The areas A_1 and A_2 swept out in unit time are equal.

- Referring to *Fig.1.2*, assuming the satellite travels distances S_1 and S_2 meters in 1s, then the areas A_1 and A_2 will be equal.
- The average velocity in each case is S_1 and S_2 meters per second, and because of the equal area law, it follows that the velocity at S_2 is less than that at S_1.
- An important consequence of this is that the satellite takes longer to travel a given distance when it is farther away from earth.

- Use is made of this property to *increase the length of time a satellite can be seen from particular geographic regions* of the earth.

- **Kelper's third law:-**
  - 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.
  - The mean distance is equal to the semimajor axis a. For the artificial satellites orbiting the earth, Kepler's third law can be written in the form

$$a^3 = \frac{\mu}{n^2} \quad \rightarrow (1.2)$$

where

n-the mean motion of the satellite in radians per second

μ- the earth's geocentric gravitational constant.

- With ***a*** in meters, the value of ***μ*** is

$$\mu = 3.986005 \times 10^{14} \text{ m}^3/\text{sec}^2 \quad \rightarrow (1.3)$$

- This Equation applies only to the ideal situation of a satellite orbiting a perfectly spherical earth of uniform mass, with no perturbing forces acting, such as atmospheric drag.
- With '*n'* in radians per second, the orbital period in seconds is given by

$$P = \frac{2\pi}{n} \quad \rightarrow (1.4)$$

- The importance of Kepler's third law is that, it shows there is a fixed relationship between period and size.

- **Newton's Universal Laws and Gravitation**

Sir Iysac Newton (1642 – 1727) derived kelper law from his own laws of mechanics and theory of Gravitation.

Newton law of gravitation sates that "gravitational force of attraction between two bodies varies as the product of their masses 'M_1 and M_2' and inversely as the square of the distance between them and is directed along a line connecting their centers.

$$\mathbf{F = G\, M_1 M_2 / d^2} \rightarrow (1.5)$$

where, G □ gravitational constant

$$\mathbf{G = 6.67 \times 10^{-11} m^3/kg^2} \rightarrow (1.6)$$

****

1.2 Orbital Parameters

Definitions of Terms for Earth-Orbiting Satellites

5. **Explain in detail about the definitions and terms related to Earth orbiting satellites.**
6. **How do you describe the orbit of a satellite? Explain with necessary equations and figure. [Nov/Dec 2022]**

- For the Earth-orbiting satellites, certain terms are used ***to describe the position of the orbit*** with respect to the earth.

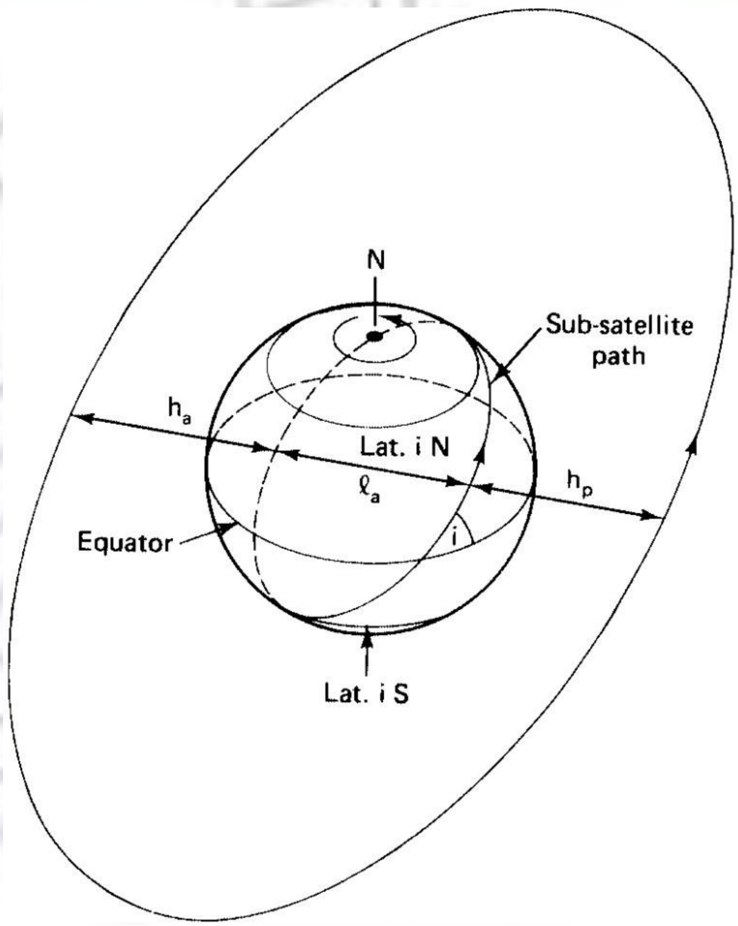


Figure 1.3 Apogee height h_a, perigee height h_p, and inclination i. l_a is the line of apsides.

- **Sub satellite path**
  - ✓ It is the path traced out on the earth's surface directly below the satellite
- **Apogee**
  - ✓ The point farthest from earth. Apogee height is shown as h_a in Fig. 1.3.
- **Perigee**
  - ✓ The point of closest approach to earth. The perigee height is shown as h_p in Fig. 1.3.
- **Line of apsides**
  - ✓ The line joining the perigee and apogee through the center of the earth.
- **Ascending node**

- ✓ The point where the orbit crosses the equatorial plane going from south to north.

- ➢ **Descending node**
  - ✓ The point where the orbit crosses the equatorial plane going from north to south.
- ➢ **Line of nodes**
  - ✓ The line joining the ascending and descending nodes through the center of the earth.
- ➢ **Inclination**
  - ✓ The angle between the orbital plane and the earth's equatorial plane.
  - ✓ It is measured at the ascending node from the equator to the orbit, going from east to north.
  - ✓ The inclination is shown as ***i*** in Fig. 1.3.
  - ✓ Itwill be seen that the greatest latitude, north or south, is equal to the inclination.

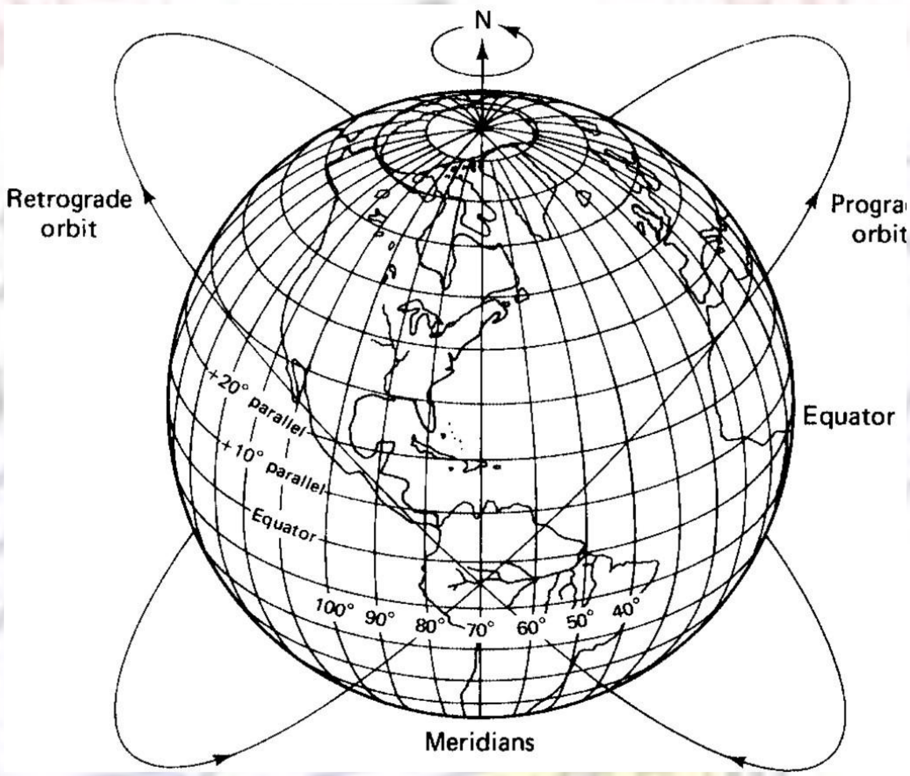


Figure 1.4 Prograde and retrograde orbits.

- ➢ **Prograde Orbit**
  - ✓ An orbit in which the satellite move in the same direction as the earth's rotation. Also called direct orbit.
  - ✓ Inclination of about 0° to 90°.
  - ✓ Most satellite are launched in this orbit, because, same direction as of earth, man a saving in launch energy.

- ➢ **Retrograde Orbit**
  - ✓ An orbit in which the satellite moves in a direction counter (opposite) to earth's rotation.
  - ✓ Inclination lies between 90° and 180°.

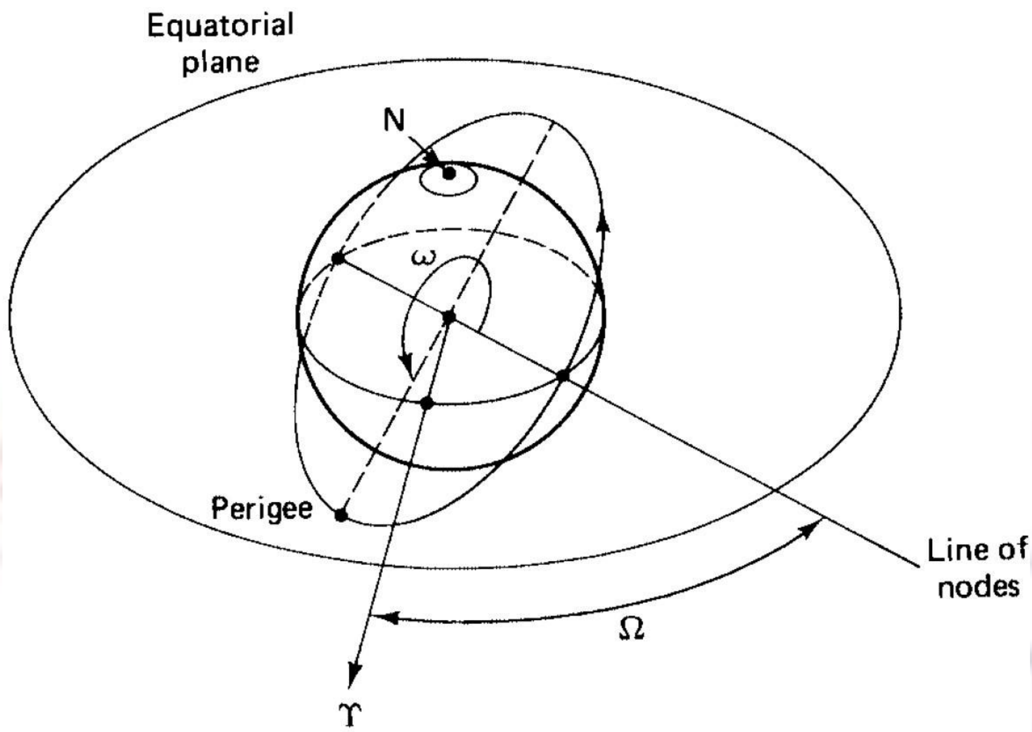


Figure. 1.5 The argument of perigee ω and the right ascension of the ascending node Ω

- **Argument of perigee**
  - ✓ The angle from ascending node to perigee measured in the orbital plane at the earth's center, in the direction of satellite motion.

- **Right ascension of descending order**
  - ✓ For practical determination of an orbit, the longitude and time of coring of the ascending node are frequently used.
  - ✓ For an absolute measurement, a fixed reference in space is required called as 'First point of Aries' also called as 'equinox'.
  - ✓ The vernal equinox occurs when the sun corner the equator going from south to north and an imaginary line draw from this equatorial corning through the center of the sun points to the first point of Aries '?' called 'line of Aries'.
  - ✓ Right accession of the ascending node is then the angle measured eastward in the equatorial plane, from the '?' line to the ascending node, shown as Ω in Figure 1.5.
- **Mean anomaly(M)**
  - ✓ It gives an average value of the angular position of the satellite with reference to the perigee.
  - ✓ For a circular orbit, M gives the angular position of the satellite in the orbit.
  - ✓ For elliptical orbit, the position is much more difficult to calculate, and M is used as an intermediate step in the calculation.
- **True anomaly**
  - ✓ It is angle from perigee to the satellite position measured at the earth's center.
  - ✓ This gives the true angular position of the satellite in the orbit as a function of time.

- **Orbital Elements**
  - ✓ Earth-orbiting artificial satellites are defined by six orbital elements referred to as the *keplerian element set*.
  - ✓ Two of this Semi major axis ***a*** and eccentricity ***e*** ⃞ gives shape of ellipse.
  - ✓ Third Mean anomaly M_0 gives the position of satellite in its orbit at a reference time known as ***epoch***.
  - ✓ Fourth Argument of perigee ***ω*** gives rotation of orbits perigee relative to the orbit's line of nodes in the earth's equatorial plane
  - ✓ Remaining two Inclination ***i*** and Right ascensions of the ascending node ⃞ relate the orbital plane's position to the earth.

- **Apogee and Perigee Heights:-**
  - ✓ These are not specified as orbital elements
  - ✓ But the apogee height and perigee height are often required.
  - ✓ The length of the radius vectors at apogee and perigee can be obtained from the geometry of the ellipse:

Length of radius vector at apogee; $r_a = a(1 + e)$

Length of radius vector at perigee; $r_p = a(1 - e)$

Apogee height: $h_a = r_a - R$.

Perigee height: $h_p = r_p - R$

where,

a⃞ semi major axis; e ⃞ eccentricity, R → mean earth radius of 6371 km.

Problem: Calculate the apogee and perigee heights for the orbital parameters for the eccentricity of .0011501 and semi major axis of 7192.3km. Assume a mean earth radius of 6371 km.

Given data:

e : = .0011501, a : =7192.3km.

R : = 6371 _ km

Solution

Length of radius vector at apogee r_a : = a . (1+ e) =>r_a: =7200.6 km

Length of radius vector at perigee r_p : = a . (1 _ e) =>r_p : = 7184.1 km

Apogee height h_a : = r_a- R =>h_a= 829.6 km

Perigee height h_p : = r_p- R =>h_p= 813.1 km

1.3 : Orbital Perturbations

7. Explain about various Orbital perturbations.*(April 2014,Nov/Dec 2012, Nov/Dec 2011, Nov/Dec 2010, Nov/Dec 2009)
What do you mean by orbital perturbations? Explain in detail. [May 2022]
What are the forces acting for satellite during the powered flight in the atmosphere? Derive the equations of motion for that powered flight stating necessary assumptions. [May 2022]
Describe various types of orbital perturbations an Earth orbiting satellite may experience with illustration. [Nov / Dec 2022]

Orbital Perturbations:-

- ***keplerian orbit:***
  - ✓ The type of orbit described so far, referred to as a keplerian orbit.
  - ✓ It 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
    - the ***only force*** acting is the centrifugal force resulting from satellite motion balancing the gravitational pull of the earth.

- ***In practice, other significant forces***
  - ✓ ***Gravitational forces*** of sun and moon, *but Gravitational forces of sun and moon have negligible affect on law-orbiting satellite but affects geostationary orbits.*
  - ✓ ***Atmospheric drag***, act upon satellite, *but Atmospheric drag have negligible affect on geostationary orbit and affects on low – orbiting satellite (below about 1000 km).*

- **Gravitational Pull of Sun and Moon:-**
  - ✓ The mass of the sun and significantly larger than that of the moon but the moon is considerably close to the earth than the sun.
  - ✓ Hence, the acceleration force induced by the moon on a geostationary satellite is about as larger as that of the sun.
  - ✓ The net effect of the acceleration forces induced by the moon and the sun on a geostationary satellite will change the plane of the orbit, at an initial average rate of change of ***0.85°/ year*** from the equatorial plane.

  - ✓ When both the sun and moon are acting on the same side of the satellite's orbit, the rate of change of the plane of the geostationary satellite's orbit will be higher than average.
  - ✓ When they are on opposite sides of the orbit, the rate of change of plane of the satellite's orbit will be less than the average.

  - ✓ This effect result in a cyclic change in the inclination, going form *0° to 14.67° in 26.6 years and back to zero*, at which the cycle is repeated.

- **Effects of a non – spherical earth:-**
  - ✓ For a spherical earth of uniform mass, By kepler's third law,

    Mean motion n_0,

$$n_0 = \sqrt{\frac{\mu}{a^3}} \qquad \rightarrow (3.1)$$

Note: The 0 subscript is included as a reminder that *this result applies for a perfectly spherical earth of uniform mass*.

- ✓ *Oblatespheroid:* It is known that the earth is not perfectly spherical, there being an equatorial bulge and a flattening at the poles. This shape is described as an oblatespheroid.
- ✓ When the earth's oblateness is taken into account, the mean motion, denoted in this case by symbol *n*, is modified to

$$n = n_0 \left[\frac{1 + K_1 (1 - 1.5 \sin^2 i)}{a^2 (1 - e^2)^{1.5}} \right] \qquad \rightarrow (3.2)$$

where, K_1 constant = 66063.1704km^2

- ✓ The earth's oblateness has negligible effect on the semimajor axis *a*, and if *a* is known, the mean motion is readily calculated.
- ✓ *Anomalistic period:* Orbital period taking into account the earth's oblateness is called as *'anomalistic period'* (e.g., from perigee to perigee).
- ✓ The mean motion specified in the NASA bulletins is the reciprocal of the anomalistic period.
- ✓ The anomalistic period is

$$P_A = \frac{2\pi}{n} \text{ sec} \qquad \rightarrow (3.3)$$

Where, *n* is in radians per second.

- ✓ If the known quantity is *n* (as is given in the NASA bulletins, for example), we can solve Eq. (3.2) for *a*, keeping in mind that n_0 is also a function of *a*.
- ✓ Equation (3.2) may be solved for *a* by finding the root of the following equation

$$n - \sqrt{\frac{\mu}{a^3}} \left[1 + \frac{K_1 (1 - 1.5 \sin^2 i)}{a^2 (1 - e^2)^{1.5}} \right] = 0 \qquad \rightarrow (3.4)$$

8. Discuss the effects of non spherical earth and atmospheric drag on satellite communications. (April/May 2011, April/May 2010, May/June 2009)

➢ **Oblateness of earth***(2 effects)*

- ✓ The oblateness of the earth also produces two rotations of the orbital plane.
- ✓ The first effect, known as ***regression of the nodes***
  - 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 (see Fig. 1.4), 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.* For a polar orbit (i= 90°), the regression is zero.

✓ The second effect, known as ***rotation of apsides*** in the orbital plane, described below.

- Both effects depend on the mean motion n, the semimajor axis a, and the eccentricity e.
- These factors can be grouped into one factor K given b
- K will have the same units as n.
- Thus, with n in *rad / day*, K will be in *rad / day*, and with n in °/*day*, K will be in °/*day*. An approximate expression for the rate of change of Ω with respect to time is

$$\frac{d\Omega}{dt} = -K \cos i \quad \rightarrow (3.5)$$

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 Eq. (3.5) 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°, 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. Such an orbit is said to be *sun-synchronous.*
- In the other major effect produced by the equatorial bulge, rotation of the line of apsides in the orbital plane, the argument of perigee changes with time, in effect, the rate of change being given by (Wertz, 1984).

$$\frac{d\omega}{dt} = K(2 - 2.5 \sin^2 i) \quad \rightarrow (3.6)$$

- *Again, the units for the rate of rotation of the line of apsides will be the same as those for n.*
- *When the inclination i is equal to 63.435°, the term within the parentheses (2-2Sin^{2}63.435) is equal to zero, and hence no rotation takes place. This fact is used 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 ω_0 at epoch gives the new values for Ω and at time t as

$$\Omega = \Omega_0 + \frac{d\Omega}{dt}(t - t_0) \quad \rightarrow (3.7)$$

$$\omega = \omega_0 + \frac{d\omega}{dt}(t - t_0 \quad \rightarrow (3.8)$$

✓ 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 ω.

- ✓ Thus, as stated above, the period P_A is the time required to go around the orbital path from perigee to perigee, even though the perigee has moved relative to the earth.

- ✓ Suppose, for example, that the inclination is 90° so that the regression of the nodes is zero (from Eq. 3.5), and the rate of rotation of the line of apsides is $-K/2$ (from Eq. 3.6), and further, imagine the situation where the perigee at the start of observations is exactly over the ascending node.
- ✓ One period later the perigee would be at an angle $-KP_A/2$ relative to the ascending node or, in other words, would be south of the equator. The time between crossings *at the ascending node* would be $P_A(1 + K/2n)$, which would be the period observed from the earth. Recall that K will have the same units as n, e.g., radians per second.

- ✓ 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 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.

(iii) Atmospheric drag*(by earth)***:-**

- ✓ 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, because of that, the satellite will not reach the same apogee height on successive revolutions.
- ✓ The result is that the *semimajor axis and the eccentricity are both reduced*.
- ✓ Drag *does not noticeably change the other orbital parameters*, including perigee height.
- ✓ In the program used for generating the orbital elements given in the NASA bulletins, a “pseudo-drag” term is generated which is equal to one-half the rate of change of mean motion (ADC USAF, 1980).
- ✓ An approximate expression for the change of major axis is

$$a \cong a_0 \left[\frac{n_0}{n_0 + n_0'(t - t} \right. \quad \rightarrow (3.9)$$

- ✓ The mean anomaly is also changed. An approximate expression for the amount by which it changes is

$$\delta M = \frac{n_0'}{2}(t - t_0)^2 \quad \rightarrow (3.10)$$

where,

t_0= reference time
$n_0{}'$ = first derivative of mean motion

****

1.4 : Station Keeping

9. Explain about station keeping.* [Nov/Dec 2022]

Station keeping

A geostationary satellite should be kept in its correct orbital slot, in addition to having its attitude control, due to some factors such as equatorial ellipticity and gravitational pull of sun and moon.

Types of station keeping:-

Depending upon the factors which affects the satellite, two types of station keeping is in use.

- ✓ East – West station keeping maneuvers.
- ✓ North – South station keeping maneuvers.

East – West station keeping maneuvers.

- ➢ The equatorial ellipticity of the earth causes geostationary satellites to drift slowly along the orbit, to one of the two sable points, at 75E and 105w.
- ➢ To counter this drift, an oppositely directed velocity component is imparted to be satellite by means of gets, which are pulsed once every 2 (or) 3 weeks.
- ➢ This results in the satellite drifting back through its domical station position.
- ➢ These maneuvers are termed as east – west station keeping maneuvers

North – South station keeping maneuvers.

- ➢ Due to the gravitational pull of sun and moon, the position of geostationary satellite results a drift on latitude.
- ➢ These forces cause the inclination to change at a rate of about 0.85/year.
- ➢ If left uncorrected, the drift would result in a cyclic change in the inclination, going from0 to 114.67of 6.6 years and back to zero, at which the cycle is repeated.
- ➢ To prevent the shift in inclination from exceeding specified limits, gets may be pulsed at the appropriate time to return the inclination to zero.
- ➢ Counteracting gets must be pulsed when the inclination is at zero, to halt the change in inclination.
- ➢ These maneuvers which are much more expensive in fuel than east –west station keeping maneuvers.
- ➢ The North – South station keeping tolerances are the same as these for east west station keeping, 0.1 in the C – band and 0.05 in the ku band.

- ➢ Orbital correction is carried out by command from the TT & C earth station, which monitors the satellite position.

- ➢ East – West & North – South station keeping maneuvers are usually carried out using the same thrusters as are used for attitude control.
- ➢ Figure 1.6 shows typical latitude and longitude variations for the Canadian Anik-C3satellite which remain after station-keeping corrections are applied.

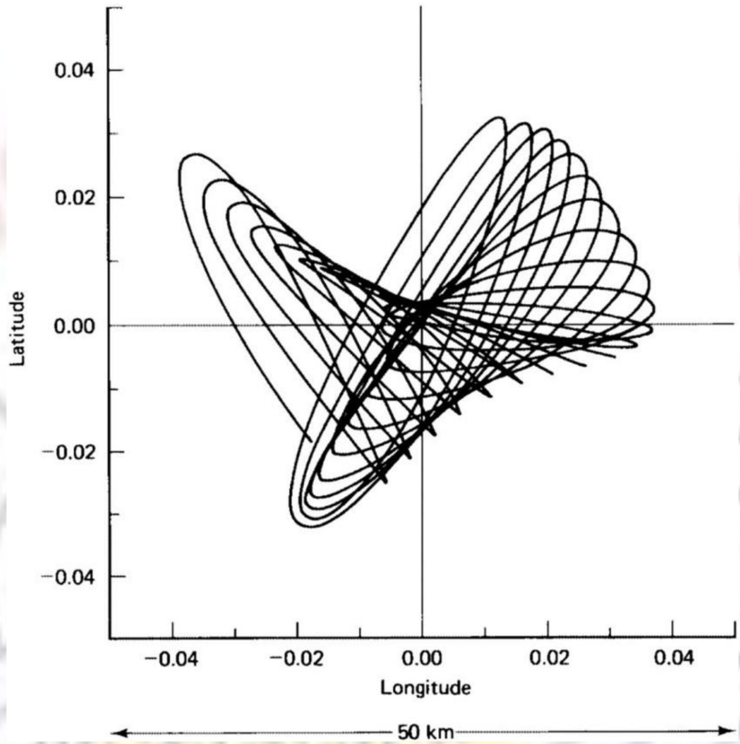


Figure 1.6 Typical satellite motion.
(From Telesat, Canada, 1983; courtesy of Telesat Canada.)

- ➢ Satellite altitude also will show variations of about 0.1 percent of the nominal geostationary height.
- ➢ If, for sake of argument, this is taken as 36,000 km, the total variation in the height is 72 km. A C band satellite therefore can be anywhere within a box bound by this height and the ±0.1° tolerances on latitude and longitude.
- ➢ Approximating the geostationary radius as 42,164 km, an angle of 0.2° subtends an arc of approximately 147 km.

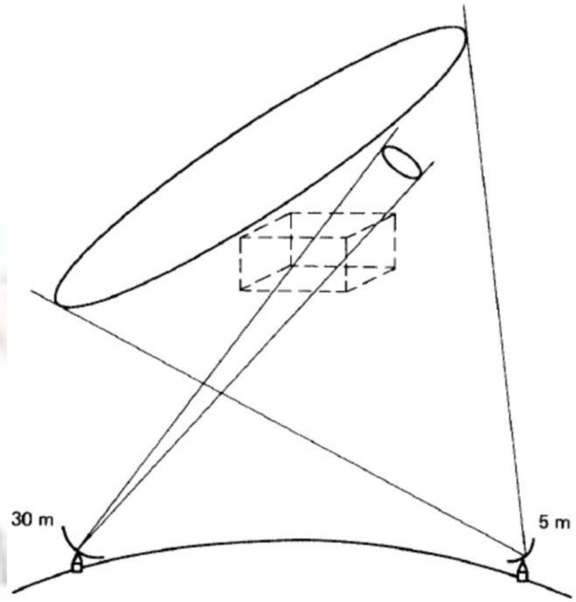


Figure 1.7 The rectangular box shows the positional limits for a satellite in geostationary orbit in relation to beams from a 30-m and a 5-m antenna.

- Thus both the latitude and longitude sides of the box are 147 km. The situation is sketched in Fig. 1.7, which also shows the relative beamwidths of a 30-m and a 5-m antenna.
- The -3dB beamwidth of a 30m antenna is about 0.12°, and of a 5m antenna, about 0.7° at 6 GHz.
- Assuming 38,000 km (typical) for the slant range, the diameter of the 30-m beam at the satellite will be about 80 km.
- This beam does not encompass the whole of the box and therefore could miss the satellite. Such narrow-beam antennas therefore must track the satellite.

- Diameter of 30m beam at the satellite wills about 80km.
- This beam does not encompass the whale satellite and therefore could minimum the satellite, (i.e) narrow beam antennas must track the satellite.
- Diameter of 5m antenna beam at the satellite will be about 464km and this does encompass the box, so tracking is not required.
- Position uncertainty of satellite leads to propagation delay in communication.

1.5 : GeoStationary And Non Geo-Stationary Orbits

10. Discuss about Geostationary orbit.*(May/June 2009)
11. Determine the geostationary Earth Orbit Radius using Kepler's third law. [Nov/Dec 2022]

Geostationary orbit

A satellite in a geostationary orbit appears to be stationary with respect to the earth, hence the name *geostationary*.

Three conditions are required for an orbit to be geostationary:

1. Satellite rotated in eastward with the same speed of earth
2. Orbit is circular
3. Inclination is zero.

- The first condition is apparent.
  - If the satellite is to appear stationary, it must rotate at the same speed as the earth, which is constant.

- The second condition follows from this and from Kepler's second law.
  - Constant speed means that equal areas must be swept out in equal time periods, and this can only occur with a circular orbit (see *Kepler's Second Law*).

- The third condition, that the inclination must be zero:
  - Any inclination would have the satellite moving north and south, and hence it would not be geostationary.
  - Movement north and south can be avoided only with zero inclination, which means that the orbit lies in the earth's equatorial plane.
  - Kepler's third law may be used to find the radius of the orbit (for a circular orbit, the semimajor axis is equal to the radius). Denoting the radius by a_{GSO}, then from Eqs. (1.2) and (1.4),

$$a_{GSO} = \left(\frac{\mu P^2}{4\pi^2}\right)^{\frac{1}{3}} \quad \rightarrow (4.1)$$

  - The period P for the geostationary is 23 h, 56 min, 4 s mean solar time (ordinary clock time).
  - This is the time taken for the earth to complete one revolution about its N-S axis, measured relative to the fixed stars.
  - Substituting this value along with the value for a given by Eq. (1.3) results in

$$a_{GSO} = 42164 \text{ km} \quad \rightarrow (4.2)$$

  - The equatorial radius of the earth, to the nearest kilometer, is

$$a_E = 6378 \text{ km} \quad \rightarrow (4.3)$$

  - Hence the geostationary height is

$$\begin{aligned} h_{GSO} &= a_{GSO} - a_E \\ &= 42{,}164 - 6378 \\ &= 35{,}786 \text{ km} \end{aligned} \quad \rightarrow (4.4)$$

- This value is often rounded up to 36,000 km for approximate calculations.
- In practice, a precise geostationary orbit cannot be attained because of disturbance forces in space and the effects of the earth's equatorial bulge.
- The gravitational fields of the sun and the moon produce a shift of about 0.85°/year in inclination.
- Also, the earth's *equatorial ellipticity* causes the satellite to drift eastward along the orbit.
- In practice, station-keeping maneuvers have to be performed periodically to correct for these shifts.

- An important point to grasp is that there is only one geostationary orbit because there is only one value of a that satisfies Eq. (1.3) for a periodic time of 23 h, 56 min, 4 s.

- Communications authorities throughout the world regard the geostationary orbit as a natural resource, and its use is carefully regulated through national and international agreements.

1.6 Near GeoStationary orbit

12. Discuss about near Geostationary orbit.*(May/June 2009)

Near geostationary orbits:-

- There are a number of perturbing forces that cause an orbit to depart from the ideal keplerian orbit.
- For the geostationary case, the most important of perturbing forces are the gravitational fields of the moon and the sun and the nonspherical shape of the earth.
- Other significant forces are solar radiation pressure and reaction of the satellite itself to motor movement within the satellite.
- As a result, station keeping maneuvers must be carried out to maintain the satellite within set limits of its nominal geostationary position.

- An exact geostationary orbit therefore is not attainable in practice, and the orbital parameters vary with time.
- The two-line orbital elements are published at regular intervals, Fig. 1.8 showing typical values.
- The period for a geostationary satellite is 23 h, 56 min, 4 s, or 86,164 s. The reciprocal of this is 1.00273896 rev/day, which is about the value tabulated for most of the satellites in Fig. 1.8.
- Thus these satellites are *geosynchronous*, in that they rotate in synchronism with the rotation of the earth.
- However, they are not geostationary. The term *geosynchronous satellite* is used in many cases instead of *geostationary* to describe these near-geostationary satellites.
- It should be noted, however, that in general a geosynchronous satellite does not have to be near-geostationary, and there are a number of geosynchronous satellites that are in highly elliptical orbits with comparatively large inclinations (e.g., the Tundra satellites).
- Although in principle the two-line elements could be used to determine orbital motion, the small inclination makes it difficult to locate the position of the ascending node, and the small eccentricity makes it difficult to locate the position of the perigee.
- However, because of the small inclination, the angles ω and Ω are almost in the same plane, and this approximation is used.

Thus the mean longitude of the satellite is given by

$$\phi_{SSmean} = \omega + \Omega + M - GST$$

$$\phi_{SS} = \omega + \Omega + v - GST \qquad \rightarrow (5.1)$$

We can calculate the true anomaly as,

$$v = M + 2e \sin(M) \qquad \rightarrow (5.2)$$

```
COMSTAR 2
1 09047U 76073A 00223.54804866 .00000046 00000-0 10000-3 0 6105
2 09047 12.0422 29.2398 0003118 249.8535 110.1810 0.99970650
89122
MORELOS B
1 16274U 85109B 00223.33258916 -.00000005 00000-0 00000+0 0 9543
2 16274 1.5560 89.5711 0001273 11.0167 218.0190 1.00272326 43202
EUTELSAT II F1
1 20777U 90079B 00224.09931713 .00000112 00000-0 00000+0 6898
2 20777 1.3398 93.0453 0004190 46.4886 264.0253 1.00275097 16856
AS1ASAT 3
1 25126U 97086A 00221.37048611 -.00000288 00000-0 10000-4 0 3326
2 25126 7.1218 291.3069 0048368 338.8396 120.3853 1.00273882 10448
INTELSAT 805
1 25371U 98037A 00223.15300705 -.00000297 00000-0 00000+0 0 2387
2 25371 0.0309 272.5299 0003525 247.9161 158.0516 1.00271603 7893
INTELSAT 806
1 25239U 98014A 00221.20890226 -.00000275 00000-0 00000-0 0 3053
2 25239 0.0360 287.7943 0003595 234.8733 189.0306 1.00270223 9029
```

Figure 1.8 Two-line elements for some geostationary satellites.

Modified inclination and eccentricity parameters can be derived from the specified values of inclination *i*, the eccentricity *e*, and the angles and **Ω**. Details of these will be found in Maral and Bousquet (1998).

1.7 : Look Angle Determination

13. What are known as sun synchronous orbits? How will you determine the look angles for the Geostationary orbit? [Or]
What are look angles and derive the expression for azimuth and elevation*(May/June 2013, , May/June 2012, Nov/Dec 2009, April/May 2008) [Or]
Derive the complete expression for Look Angles, along with intermediate angle in satellite communication. Show that intermediate angle is : [Dec 2020, May 2021]

$$\alpha = \tan^{-1}\left(\frac{\tan|l_s - l_e|}{\sin(L_e)}\right)$$

[Or]
How do you find the elevation and azimuth for an earth station to look at geostationary satellite? [Dec 2021]

- **Antenna Look Angles:-**

Look angles for the ground station antenna are the azimuth and elevation angles required at the antenna to that is point directly at the satellites.

Azimuth – horizontal
Elevation – vertical

To determine look angle, the following information are needed,

(i) Earth –station latitude (⍰$_E$)
(ii) earth – station longitude (ϕ_E)
(iii) longitude of sub satellite point (ϕ_{SS}).

- ✓ North latitude takes as positive, (+ve) angles
- ✓ South latitude taken as Negative (-ve) angles, i.e., 40°S → (-40°)

- ✓ East longitude taken positive (GMT based)
- ✓ West longitude takes as negative. i.e., 35°W → (-35°)

Low – Earth Orbiting (LEO) satellite, variation of earth radius is taken into account, but in geostationary axis is constant;
where, radius R = 6371km.→ (8.1)

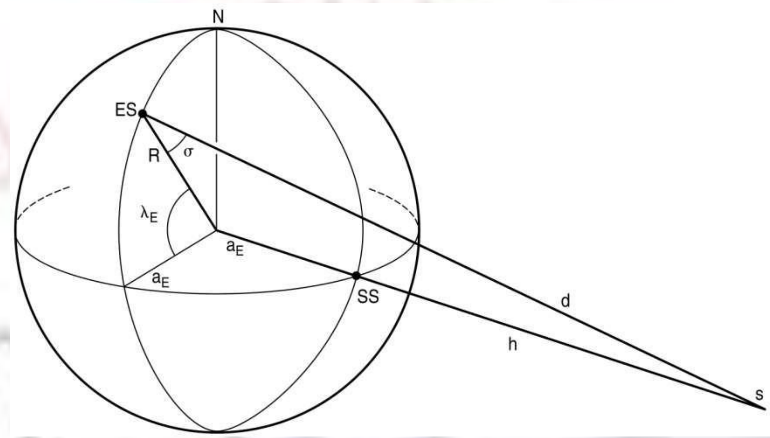


Figure 1.9 The geometry used in determining the look angles for a geostationary satellite.

Where, ES Earth station, S Satellite
SS Sub satellite print,   angle to be determined
d range from earth stations to satellite.

- ➢ The geometry involving these quantities is shown in Fig. 1.9.
  Two types of triangles involved in geometry,
  (i) spherical triangle (*Heavy outline in 1.10 (a)*)
  (ii) plane triangle. (*1.10 (b)*)

- ➢ In spherical triangle, sides are all are of great circle.
  - ✓ Sides are defined by the angles suspended by them at the center of the earth.
  - ✓ Side ‘a’ is the angle between the radius to the north pole and radius in the sub satellite point, (i.e) a = 90 .
  - ✓ It is called as “equatorial triangle”, if one side = 90.

Angle ‘b’ angle between radius of to the earth station and radius to the sub satellite point.
Angle ‘c’ angle between radius of to the earth station and radius to the sub satellite point.

$$c = 90 - _E$$

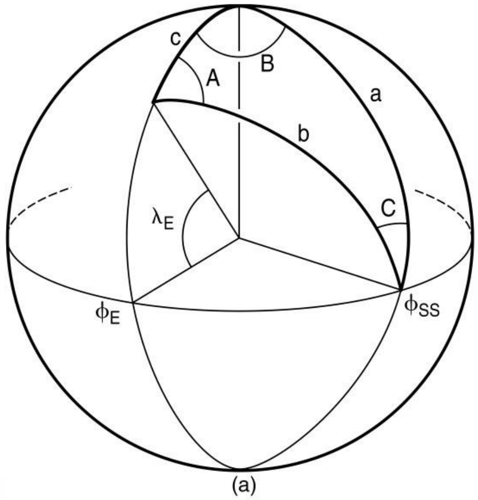


Figure 1.10

(a) The spherical geometry related to Fig. 1.8.

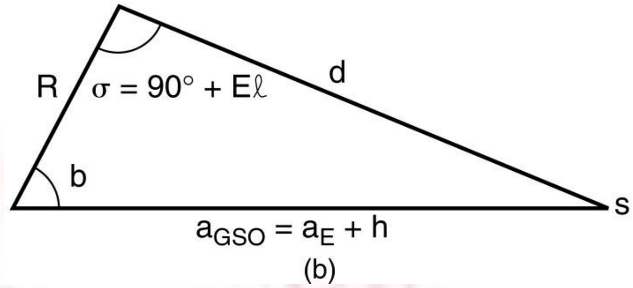


Figure 1.10

(b) The plane triangle obtained from Fig. 1.8.

There are six angles defining spherical triangle.

A, B, C = plane angles.

Angle A □ angle between the plane containing c and the plane containing b.

Angle B □ angle between the plane containing c and the plane containing a. $B = \phi_E - \phi_{SS}$

Angle C □ angle between the plane containing b and the plane containing a.

To summarize to this point, the information known about the spherical triangle is

$$a = 90° \quad \rightarrow (8.2)$$

$$c = 90° - \lambda_E \quad \rightarrow (8.3)$$

$$B = \phi_E - \phi_{SS} \quad \rightarrow (8.4)$$

When earth station is west of sub satellite point, B is negative

If earth station is east of sub satellite point, B is positive

When earth station is north, c is less than 90□,

if it is south, c□ greater than 90□.

By special rules, known as Napier's rules;

$$b = \arccos(\cos B \cos \lambda_E) \quad \rightarrow (8.5)$$

$$A = \arcsin\left(\frac{\sin|B|}{\sin b}\right) \quad \rightarrow (8.6)$$

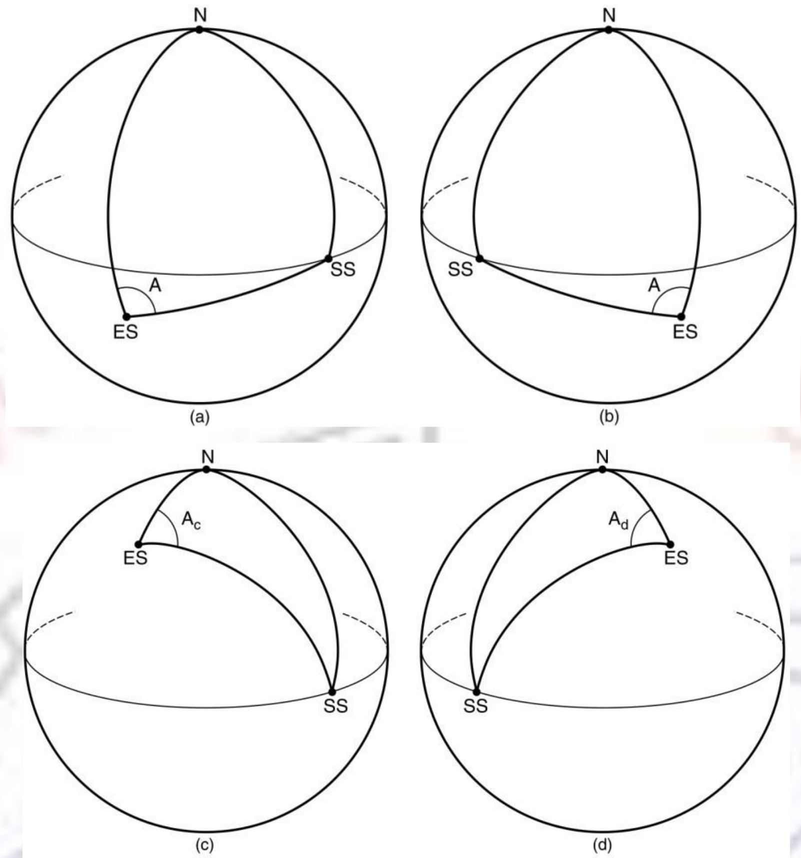


Figure 1.11 Azimuth angles related to angle *A* (see Table 8.1).

TABLE 8.1. Azimuth Angles A_z from Fig. 3.3

Figure	**□E**	**B**	**Az degrees**	
A	< 0	< 0	A	□ Angle ‘ A’ is acute
B	< 0	> 0	360□ - A	□ Angle ‘ A’ is acute
C	>0	>0	180□ - A	□ Angle ‘ n’ is obtuse
D	>0	>0	180□ + A	□ Angle ‘ A’ is obtuse

- ➢ Applying the cosine rule for plane triangles to the triangle of Fig.3.2*b* allows the range *d* to be found to a close approximation:

$$d = \sqrt{R^2 + a_{GSO}{}^2 - 2Ra_{GSO}\cos b} \qquad \rightarrow (8.7)$$

- ➢ Applying the sine rule for plane triangles to the triangle of Fig. 3.2*b*allows the angle of elevation to be found:

$$El = \arccos\left(\frac{a_{GSO}}{d}\sin b\right) \qquad \rightarrow (8.8)$$

- ➢ *Figure 1.12 shows the look angles for Ku-band satellites as seen from Thunder Bay, Ontario, Canada.*

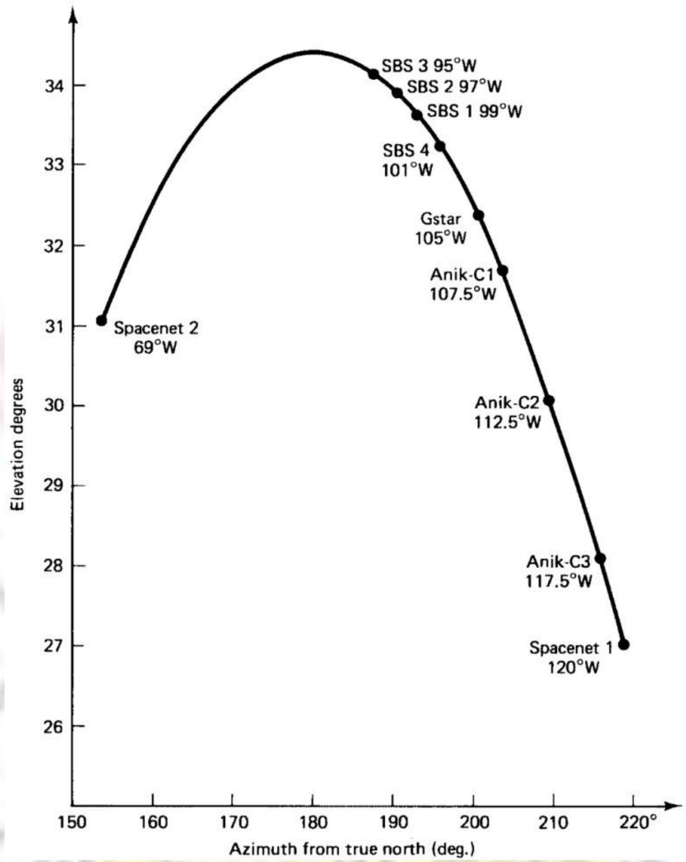


Figure 1.12 Azmuth-elevation angles for an earth station location of 48.42°N, 89.26°W (Thunder Bay, Ontario). Ku-band satellites are shown.

- The preceding results do not take into account the case when the earth station is on the equator.
- Obviously, when the earth station is directly under the satellite, the elevation is 90°, and the azimuth is irrelevant.
- When the subsatellite point is east of the equatorial earth station ($B < 0$), the azimuth is 90°, and when west ($B > 0$), the azimuth is 270°.
- Also, the range as determined by Eq. (8.7) is approximate, and where more accurate values are required, as, for example, where propagation times need to be known accurately, the range is determined by measurement.

- For a typical home installation, practical adjustments will be made to align the antenna to a known satellite for maximum signal. Thus the look angles need not be determined with great precision but are calculated to give the expected values for a satellite whose longitude is close to the earth station longitude.
- In some cases, especially with direct broadcast satellites (DBS), the home antenna is aligned to one particular cluster of satellites.

1.8 : Limits of Visibility

14. Write a note on
(i) Limits of visibility
(ii) Eclipse
(iii) Sun transit outage.

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.

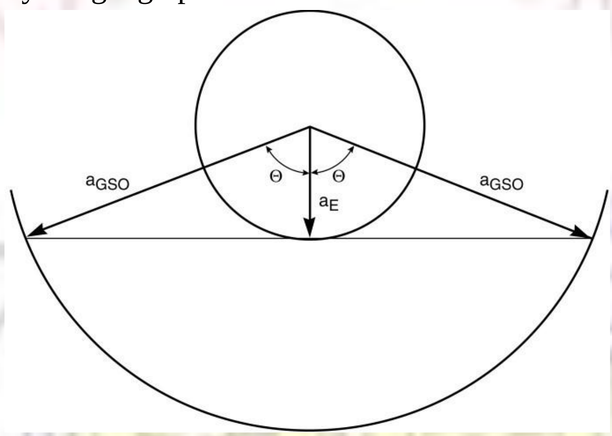


Figure 1.13. Illustrating the limits of visibility.

Estimate of the longitudinal limits

- Consider an earth station at equator, with the antenna pointing either east (or) west along the horizontal, as shown in Fig. 1.13.
- Then the limiting angle is given by

$$\theta = \arccos \frac{a_E}{a_{GSO}} = \arccos \frac{6378}{42{,}164} = 81.3° \quad \rightarrow (9.1)$$

- Thus, for this situation, an earth station could see satellites over a geostationary arc bounded by ±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 Fig. 1.10 (b), let *S* represent the angle subtended at the satellite when the angle

$$\sigma_{min} = 90° + El_{min}.$$

- Applying the sine rule gives

$$S = \arcsin\left(\frac{R}{a_{GSO}} \sin \sigma_{min}\right) \quad \rightarrow (9.2)$$

- 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

$$b = 180 - \sigma_{min} - S \quad \rightarrow (9.3)$$

- From Eq. (8.5),

$$B = \arccos\left(\frac{\cos b}{\cos \lambda_E}\right) \quad \rightarrow (9.4)$$

- If B is known, the satellite longitude will be determined,

$$B = \phi_E - \phi_{SS}$$

$$\phi_{SS} = \phi_E + B, \text{ East limit}$$

$$\phi_{SS} = \phi_E - B, \text{ West limit}$$

1.9 : Eclipse and Sun transit outage

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 Fig. 1.14.

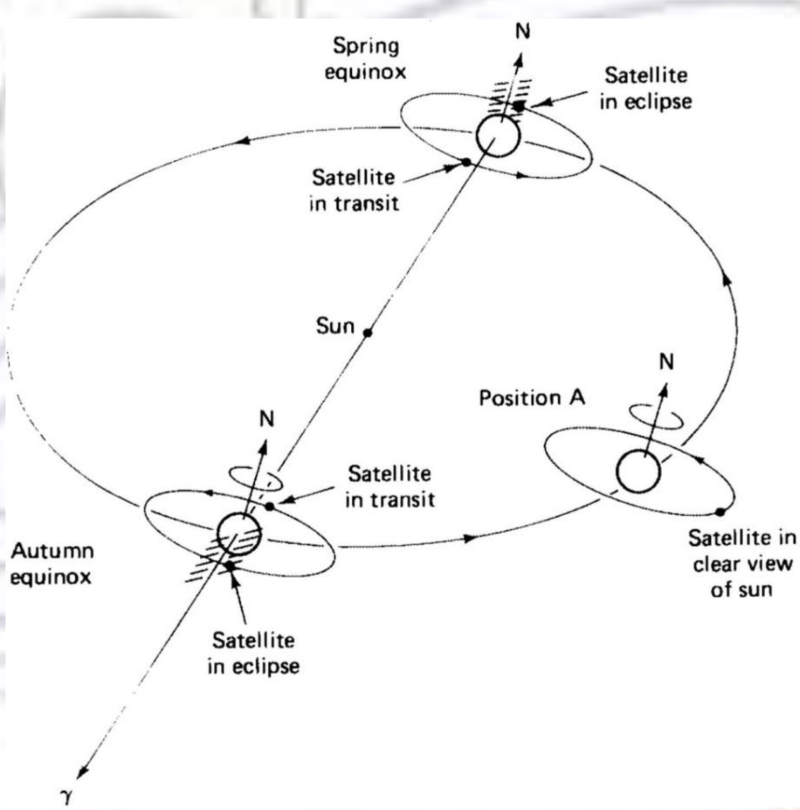


Figure 1.14 Showing satellite eclipse and satellite sun transit around spring and autumn equinoxes.

- 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 Fig. 1.14.
- 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. 1.15.
- 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, when usage is likely to be low.
- Thus satellite longitudes which are west, rather than east, of the earth station are more desirable.

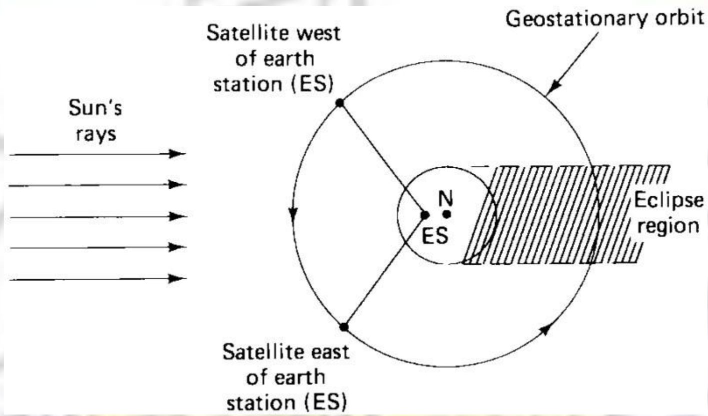


Figure 1.15 A satellite east of the earth station enters eclipse during daylight (busy) hours at the earth station. A satellite west of the earth station enters eclipse during night and early morning (nonbusy) hours.

Sun Transit Outage:-

- Another event which must be allowed for during the equinoxes is the transit of the satellite between earth and sun (see Fig. 1.14), such that the sun comes within the beamwidth of the earth station antenna.
- When this happens, the sun appears as an extremely noisy source which completely blanks out the signal from the satellite.
- This effect is termed *sun transit outage*, and it lasts for short periods each day for about 6 days around the equinoxes.
- The occurrence and duration of the sun transit outage depends on the latitude of the earth station, a maximum outage time of 10 min being typical.

1.10 : Subsatellite point

15. How will you determine the sub satellite point? Or Explain the determination of sub satellite point. (May/June 2013,April/May 2011,April/May 2010, April/May 2008)*

- 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.**

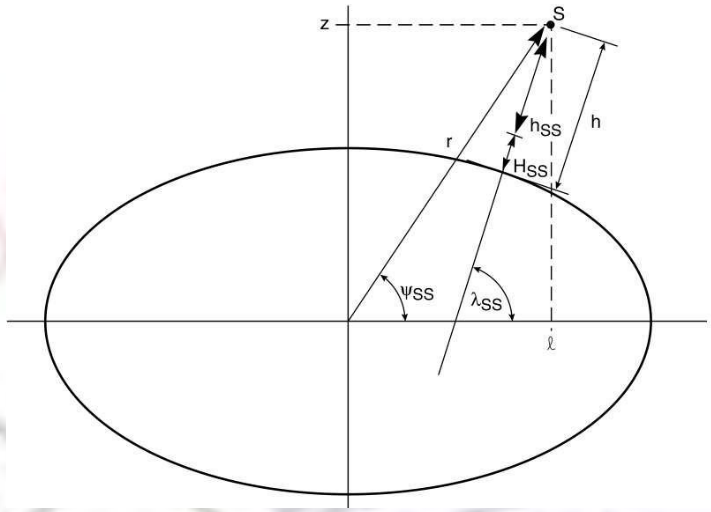


Figure 1.16 Geometry for determining the subsatellite point.

- Figure 1.16 shows the meridian plane which cuts the subsatellite point.
- The height of the terrain (ground) 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

$$h = H_{SS} + h_{SS} \quad \rightarrow (7.1)$$

$$N = \frac{a_E}{\sqrt{1 - e_E^2 \sin^2 \lambda_{SS}}} \quad \rightarrow (7.2)$$

By, IJK frame,

$$r_I = (N + h) \cos \lambda_{SS} \cos LST \quad \rightarrow (7.3)$$

$$r_J = (N + h) \cos \lambda_{SS} \sin LST \quad \rightarrow (7.4)$$

$$r_K = [N(1 - e_E^2) + h] \sin \lambda_{SS} \quad \rightarrow (7.5)$$

East longitude,

$$\mathrm{EL} = \mathrm{LST} - \mathrm{GST} \quad \rightarrow (7.6)$$

Where, GST is the Greenwich sidereal time.

Predicting satellite position

The basic factors affecting satellite position are outlined in the previous sections. The NASA two-line elements are generated by prediction models contained in Spacetrack report no. 3 (ADC USAF, 1980), which also contains Fortran IV programs for the models. Readers desiring highly accurate prediction methods are referred to this report. Spacetrack report No. 4 (ADC USAF, 1983) gives details of the models used for atmospheric density analysis.

1.11 : Launching Procedures

Launching Orbits

9. Discuss the launching orbits in detail.*(April/May 2011)

- Satellite can be directly injected for low altitude orbits (upto 200km)
- Launch vehicle may be
  - ✓ Expendable such as US Atlas-centaur, Delta rockets, European Space Agency Ariane Rocket
  - ✓ Reusable launch vehicle as space shuttle used by US (also called ‘Space Transportation System-STS’)

- When altitude is >200km, not economical to inject directly into orbit hence transfer orbit it used.
- For high altitude orbit, the satellite is first placed in transfer orbit.
- ‘Hohmann transfer orbit’ is used always because of minimum energy requirement but takes long time for placement then other.

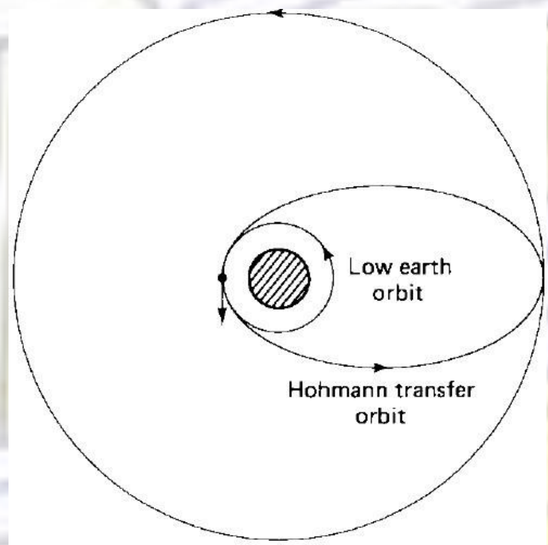


Figure 1.16 Hohmann transfer orbit.

- Transfer orbit – transformation of satellite from low circular orbit to high circular orbit.
- In low orbit, at Perigee, a Perigee Kick Motor (PKM) is used, which imparts the thrust at Perigee. (elliptical orbit)
- At Apogee, a Apogee Kick Motor (AKM) is used, which changes the velocity of satellite to place it into circular orbit.
- It take nearly 1-2 months for full operation.
- A network of ground station spread across the earth is required to perform T T& C functions.
- Prograde rotation is always used for energy efficiency.
- For prograde orbit

$$0 \leq i \leq 90°$$

By Nepier rule $\cos i = \cos \lambda \sin Az$

λ=latitude, Az=Azimuth angle

by Cape Kennedy, the smallest initial inclination which can be achieved for easterly launch is approximately 28°

Launch vehicles

11. Write a brief note on Launch vehicles and propulsion.*(April 2014,May/June 2013, May/June 2012)

Expendable Launch Vehicle (ELV):

- These launch vehicles have multiple stages and as each stage are completed, that portion of the launcher is expended until the final stage places the satellite into the desired trajectory.

Reusable Launch Vehicle (RLV):

- ➢ The space shuttle, called the Space Transportation System (STS) by NASA, is partially reusable.
- ➢ The solid rocket boosters are recovered and reused for future missions and the shuttle vehicle itself is flown back to earth for reuse.

- ➢ All Expendable Launch Vehicles use the same basic technology to get into space – two or more rocket-powered stages, which fall away when their engine burns are completed.
- ➢ Whatever a launch vehicle carries above the final discarded stage is considered the payload.

A payload's weight, orbital destination and purpose determine what size launch vehicle is required.

ELV Services Fleet

Atlas/Centaur

- ➢ The Atlas/Centaur vehicles first became operational in 1966.
- ➢ Lockheed Martin used the Atlas II and III vehicles to launch military, commercial and scientific payloads into space from Space Launch Complex 36 at CCAFS and Space Launch Complex 3E at VAFB.
- ➢ More than 580 Atlas flights have taken place, including 170 flights with the Centaur stage added to create the Atlas/Centaur vehicle.
- ➢ The Centaur was the first high-energy, liquid-hydrogen/liquid-oxygen launch vehicle stage, and it provided the most power for its weight of any proven stage then in use.
- ➢ The Atlas/Centaur was the launch vehicle for Surveyor I, the first U.S. spacecraft to soft-land on the Moon.
- ➢ Other spacecraft launched by Atlas/Centaurs include the Orbiting Astronomical Observatories; Applications Technology Satellites; the Intelsat IV, IV-A and V series of communications satellites; Mariner Mars orbiters; a Mariner spacecraft that made a flyby of Venus and three flybys of Mercury; Pioneers, which accomplished flybys of Jupiter and Saturn; and Pioneers that orbited Venus and sent probes plunging through its atmosphere to the surface.
- ➢ Most recently, NASA launched the Tracking Data and Relay Satellite-J communication satellite Dec. 4, 2002, on an Atlas IIA from CCAFS.

Delta

- Delta rockets have been built and launched since 1960. Delta's origins go back to the Thor intermediate-range ballistic missile, which was developed in the mid-1950s for the U.S. Air Force.
- The Thor, a single-stage, liquid-fueled rocket, was modified to become the Delta launch vehicle, which later evolved into the Delta II.
- Known as the "workhorse" of the launch industry, the Delta II comprises a group of expendable rockets that can be configured as two- or three-stage vehicles and with three, four or nine strap-on graphite epoxy motors (GEMs) depending on mission needs.
- Delta IV was developed in partnership with the U.S. Air Force EELV program and is the most advanced family of Delta rockets. Delta IV blends advanced and proven technology to launch virtually any size medium-to-heavy class payload to space.

Pegasus

- On April 5, 1990, Orbital began a new era in commercial space flight when our Pegasus rocket was launched for the first time from beneath a NASA B-52 carrier aircraft in a mission that originated from Dryden Flight Research Center in California.

- In the decade since its maiden flight, Pegasus has become the world's standard for affordable and reliable small launch vehicles. It has conducted 38 missions, launching 78 satellites.
- The three-stage Pegasus is used by commercial, government and international customers to deploy small satellites weighing up to 1,000 pounds into low-Earth orbit.
- Pegasus is carried aloft by our "Stargazer" L-1011 aircraft to approximately 40,000 feet over open ocean, where it is released and then free-falls in a horizontal position for five seconds before igniting its first stage rocket motor.
- With the aerodynamic lift generated by its unique delta-shaped wing, Pegasus typically delivers satellites into orbit in a little over 10 minutes.
- This patented air-launch system reduces cost and provides customers with unparalleled flexibility to operate from virtually anywhere on Earth with minimal ground support requirements.
- Pegasus launches have been conducted from six separate sites in the U.S., Europe and the Marshall Islands, the first time a space launch vehicle has demonstrated such operational flexibility.

Taurus

- The Taurus rocket offers an affordable, reliable means of launching small satellites into low-Earth orbit.
- Developed under the sponsorship of the Defense Advanced Research Projects Agency (DARPA), Taurus was designed for easy transportability and rapid set-up and launch. Since its debut flight in 1994, Taurus has conducted six of seven successful missions launching 12 satellites for commercial, civil, military, and international customers.
- Taurus is a ground-based variant of our air-launched Pegasus rocket. The four stage, inertially guided, all solid propellant vehicle can deploy 1,350-kilogram (3,000 pound) satellites into low-Earth orbit.
- Two fairing sizes offer flexibility in designing a particular mission, and with the addition of a structural adapter, either can accommodate multiple payloads, resulting in lower launch costs for smaller satellites "sharing" a mission.
- A cornerstone of the Taurus program is a simplified integration and test capability that includes horizontal integration of the rocket's upper stages and offline encapsulation of the payload within the fairing.
- The upper stages and the encapsulated cargo are delivered to the launch site, where they are mated. The whole assembly is then stacked on the first stage using a mobile crane

- The Taurus launch system includes a complete set of ground support equipment to ensure the ability to operate from austere sites.
- Thus far, Taurus has launched from the U.S. Government's Western range at Vandenberg Air Force Base (VAFB) in California. Taurus is also approved for launch from Cape Canaveral Air Station (CCAS) in Florida, Wallops Flight Facility (WFF) in Virginia, and Kodiak Launch Complex, Alaska.

Titan

- The Titan was used by NASA to launch interplanetary missions from CCAFS. An earlier version of the Titan vehicle, the Titan III-E/Centaur, built by Martin Marietta and General Dynamics, was used to launch two Helios missions to the Sun, two Viking missions to Mars, and two Voyager missions to Jupiter and Saturn beginning in the 1970s.
- One of the Voyagers also continued on to Uranus and Neptune. All of the missions provided remarkable new scientific data about our Solar System and spectacular color photographs of the planets they explored, as well as some of their moons.
- The Titan IV launched NASA's Cassini spacecraft to Saturn in 1997. The Titan III sent NASA's Mars Observer on its journey in 1992.
- The Titan II was used This Titan IVB/Centaur rocket launches Oct. 15, 1997, carrying the Cassini spacecraft. to launch many National Oceanic and Atmospheric Administration (NOAA) weather satellites.
- Most recently, a Titan II launched NASA's NOAA-M satellite June 24, 2002, from VAFB.

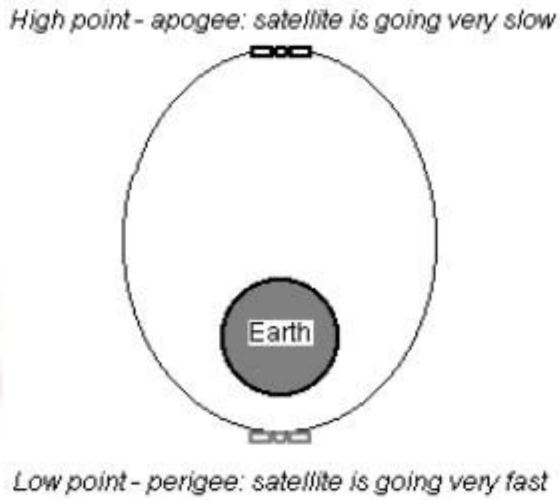
Ariane

- The Ariane family of launch vehicles has been developed by the European Space Agency under the management of the Centre National d'Etudes Spatiales (CNES).
- In July 1973, during the European Space Conference, it was decided to combine the European
- Launcher Development Organisation (ELDO) and the European Space Research Organisation (ESRO) into a single body called the European Space Agency (ESA).
- Among the objectives of the Agency was that of developing a range of three-stage launchers to permit direct injection of the satellite–apogee motor combination from a transfer orbit into the geostationary satellite orbit without a ballistic phase, starting from the Kourou launch site in French Guyana.
- This orbit can be obtained accurately using guidance of the launch vehicle by an on-board computer which uses information provided by an inertial unit.
- Steering is provided by orientation of the jet of the main motors of the various stages.

Launching of geostationary satellite:

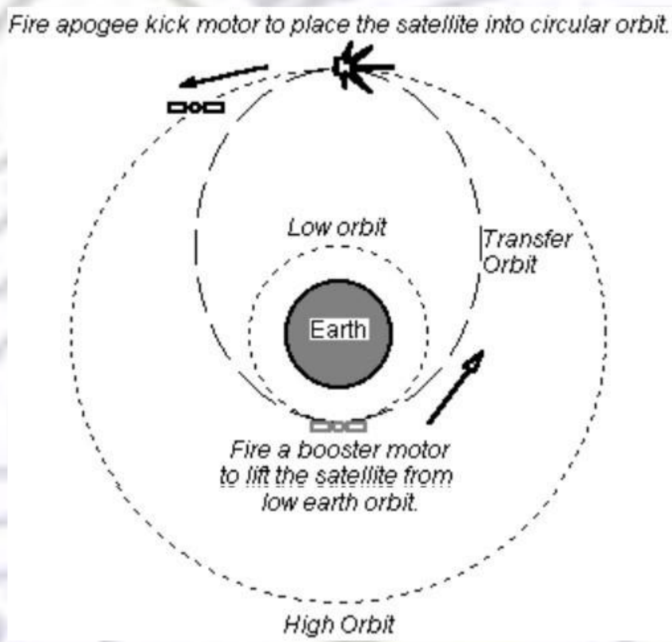
12. Give a detailed note on launching vehicles and the procedures employed for launching spacecraft in GEO orbits. (May/June 2012)

- Initially place spacecraft with the final rocket stage into LEO.
- After a couple of orbits, during which the orbital parameters are measured, the final stage is reignited and the spacecraft is launched into a geostationary transfer orbit(**GTO**).
- After a few orbits in GTO, while the orbital parameters are measured, a rocket motor (**AKM**) is ignited at apogee and GTO is raised until it is circular geostationary orbit.
- AKM (Apogee Kick Motor) is used to circularize the orbit at GEO.



Geostationary Transfer Orbit:-

- Firing a rocket motor at apogee is called "apogee kick", and the motor is called the "**apogee kick motor".**



- Installing a geostationary satellite into orbit using a launcher with several stages requires three phases.

Launch phase

- From launcher take-off to injection at the perigee of the transfer orbit, the actions are as follows:
  - Increase the altitude in order to achieve the altitude of the perigee.
  - Drop the fairing after passing through the dense layers of the atmosphere.
- Bring the last stage–satellite assembly onto a trajectory which intersects the equatorial plane
- parallel to the surface of the earth with the required velocity on passing through the equatorial plane (the perigee of the transfer orbit).

Transfer phase

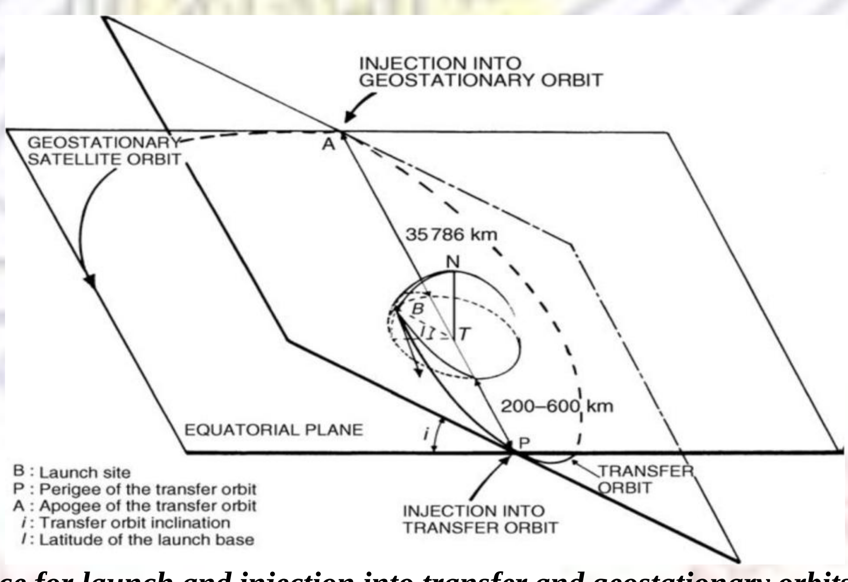
- The transfer phase starts with injection of the composite satellite–launcher final stage and terminates with injection into the quasi-geostationary satellite orbit at the apogee of the transfer orbit. In this phase, the actions are as follows:
  - Separate of the satellite and the final stage.
  - Determine orbit parameters.
  - Measure the satellite attitude.
  - Correct the satellite orientation in view of the apogee maneuver.
- The orientation may be maintained either by causing the satellite to rotate (spin stabilization) or by active attitude control using sensors and actuators (three-axis stabilization).

Positioning phase

- This phase starts with injection into the quasi-geostationary satellite orbit at the apogee of the transfer orbit and terminates with positioning of the satellite at the chosen station in the geostationary satellite orbit.

Other procedures

- Other procedures for installation into orbit which permit the launch vehicle to move out of the plane defined by the launch azimuth and the latitude of the launch base ('dog leg' maneuvers) can be envisaged.
- In this way, by reigniting the last stage in flight, the Proton launcher permits direct injection of the satellite into the geostationary satellite orbit.
- This approach avoids the need for an apogee motor and thus permits launching of satellites with larger useful mass.



Sequence for launch and injection into transfer and geostationary orbits with an expendable launch vehicle.

1. 11: Launching Procedures

13. Derive from basic principles, the orbital velocity of a satellite and calculate the same if it is a circular orbit.(Nov/Dec 2012)

(or)

Discuss the launching orbits in detail.*(April/May 2011)

- Satellites may be *directly injected* into low-altitude orbits, up to about200 km altitude, from a launch vehicle.
- Launch vehicles may be classifiedas *expendable* or *reusable.* 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).

- 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 low earth orbit and the final high-altitude orbit.
- Mostly, 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.

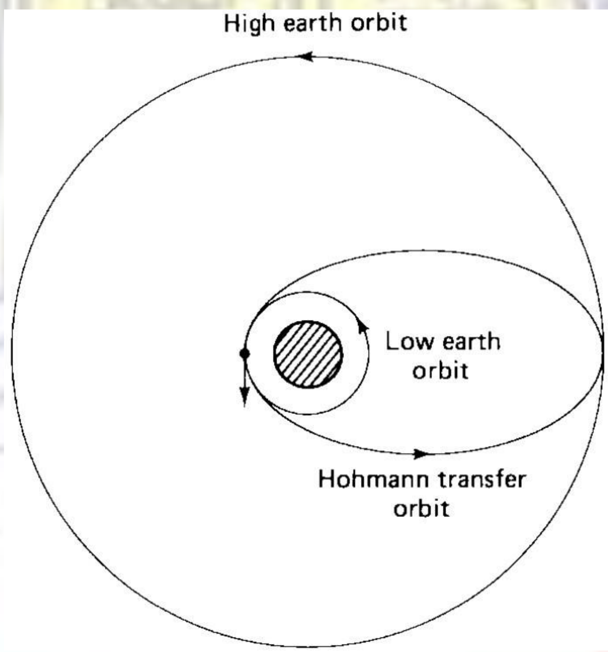


Figure 3.10 Hohmann transfer orbit.

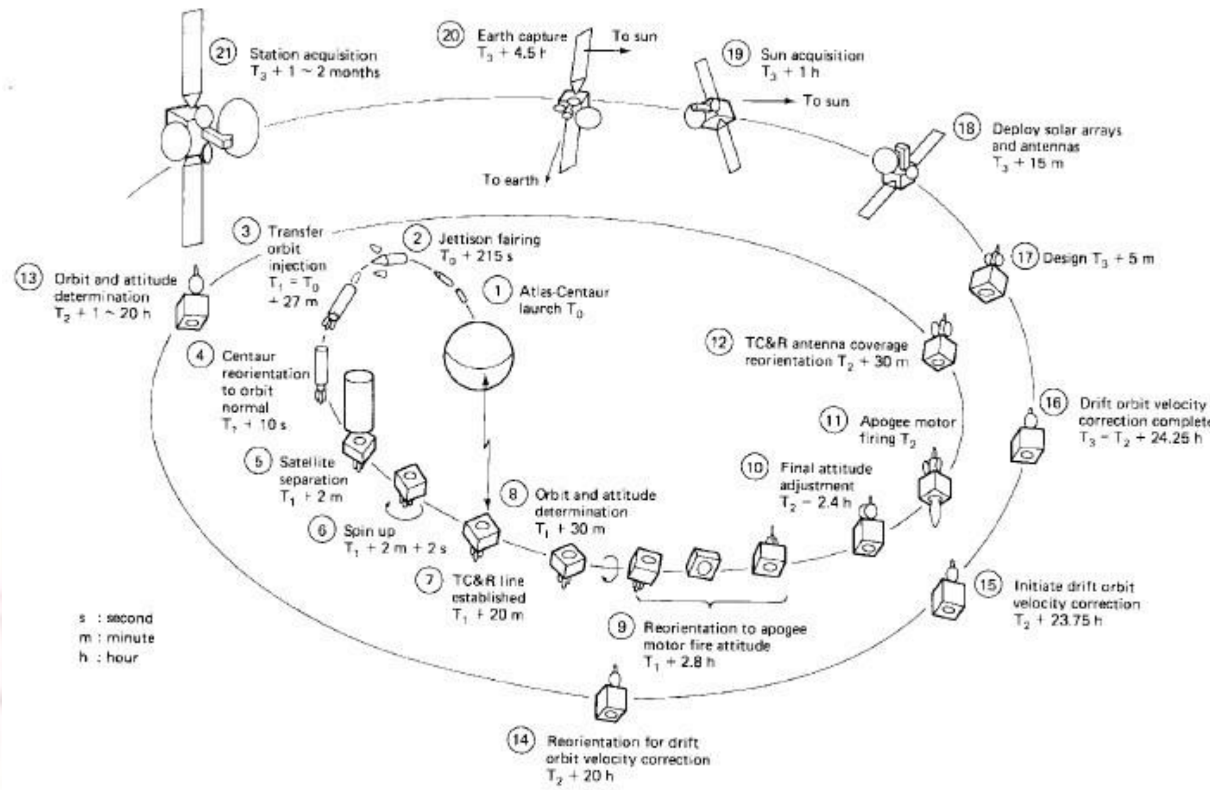


Figure 3.11 From launch to station of INTELSAT V (by Atlas-Centaur). (© *KDD Engineering & Consulting, Inc., Tokyo. From Satellite Communications Technology, edited by K. Miya, 1981*.0029

- ➢ Assume for the moment that all orbits are in the same plane and that transfer is required between two circular orbits, as illustrated in Fig. 3.10.
- ➢ 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 STS, the satellite must carry a perigee kick motor which imparts the required thrust at perigee.
- ➢ Details of the expendable vehicle launch are shown in Fig. 3.11 and of the STS launch, in Fig. 3.12.
- ➢ At apogee, the apogee kickmotor (AKM) changes the velocity of the satellite to place it into a circular orbit in the same plane.
- ➢ As shown in Fig. 3.11, it takes 1 to 2 months for the satellite to be fully operational (although not shown in Fig. 3.12, the same conditions apply).
- ➢ 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.

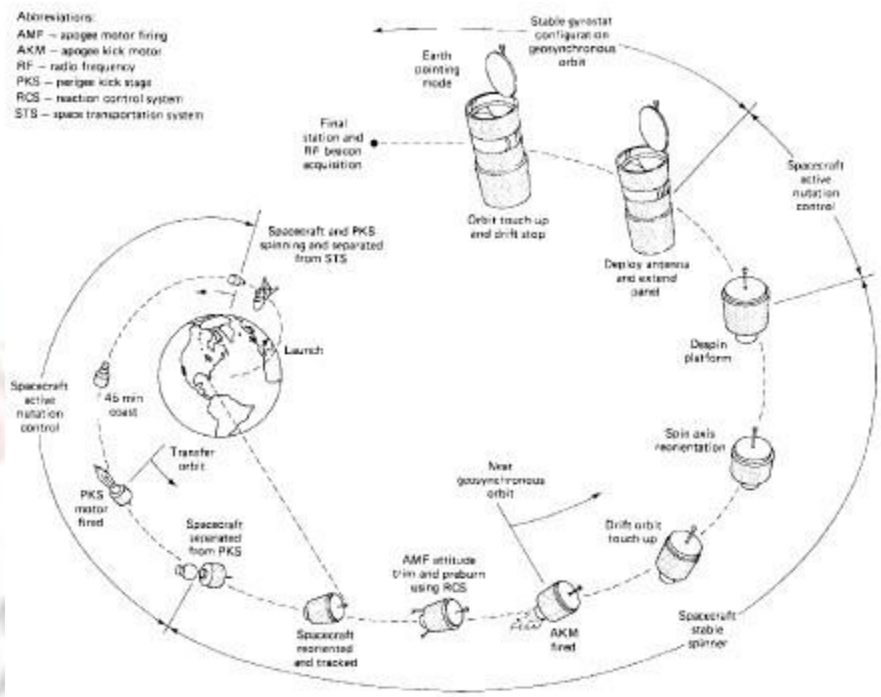


Figure 3.12 STS-7/Anik C2 mission scenario.
(*From Anik C2 Launch Handbook, courtesy of Telesat Canada.*)

- ➢ 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, and these 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.

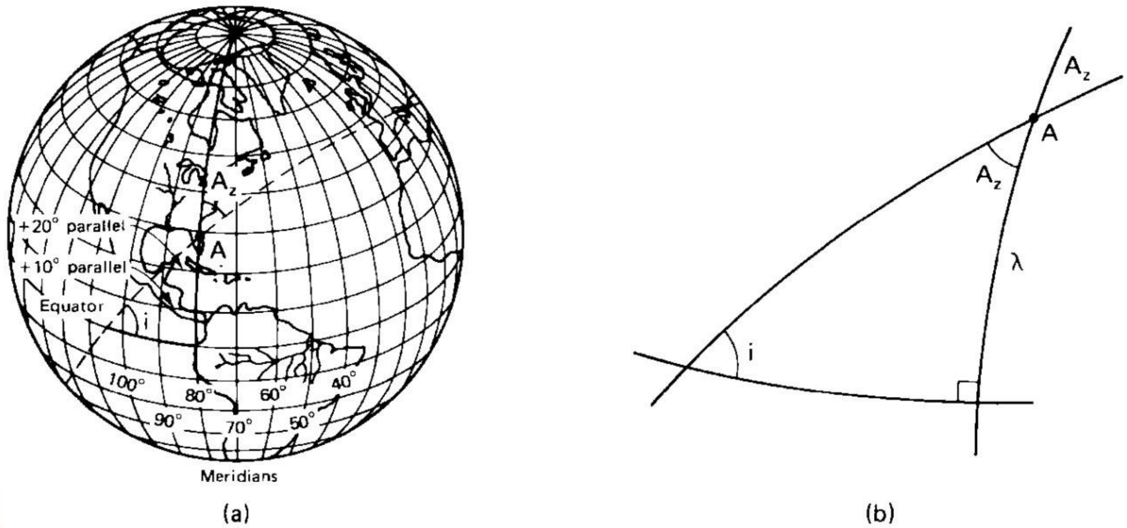


Figure 3.13 (*a*) Launch site *A*, showing launch azimuth A_z; (*b*) enlarged version of the spherical triangle shown in (*a*). λ is the latitude of the launch site.

- The relationship between inclination, latitude, and azimuth may be seen as follows [this analysis is based on that given in Bate et al.(1971)].
- Figure 3.13*a* shows the geometry at the launch site *A* at latitude λ (the slight difference between geodetic and geocentric latitudes may be ignored here). The dotted line shows the satellite earth track, the satellite having been launched at some azimuth angle A_z. Angle *I* is the resulting inclination.

- The spherical triangle of interest is shown in more detail in Fig.3.13*b*.
- This is a right spherical triangle, and Napier's rule for this gives

$$\cos i = \cos \lambda \sin Az \quad (3.23)$$

- For a prograde orbit (see Fig. 2.4 and Sec. 2.5), $0 \leq i < 90°$, and hence cos *i* is positive.
- Also, $-90° \leq \lambda \leq 90°$, and hence cos λ is also positive.
- It follows therefore from Eq. (3.23) that $0 \leq Az \leq 180°$, or the launch azimuth must be easterly in order to obtain a prograde orbit, confirming what was already known.
- For a fixed _, Eq. (3.23) also shows that to minimize the inclination *i*, cos *i* should be a maximum, which requires sin *Az* to be maximum, or $Az = 90°$. Equation (3.23) shows that under these conditions

$$\cos i_{min} = \cos \lambda \quad (3.24)$$

or

$$i_{min} = \lambda \quad (3.25)$$

- Thus the *lowest* inclination possible on initial launch is equal to the latitude of the launch site.
- This result confirms the converse statement made in Sec. 2.5 under *inclination* that the greatest latitude north or south is equal to the inclination.
- From Cape Kennedy the smallest initial inclination which can be achieved for easterly launches is approximately 28°.

1.12 : Launch Vehicles And Propulsion

14. Write a brief note on Launch vehicles and propulsion.*(April 2014,May/June 2013, May/June 2012)

Launch Vehicle and Propulsion:-

- Launching of satellite into an orbit is extra ordinary complex and costly operation.
- The launch Vehicle costs as much as the satellite itself.
- Launch Vehicle as a system includes structure, engines, Propellant storage and pumps guidance and control.
- A rocket is a launch vehicle when it is used to launch a satellite (or) other pay load into space.

Principle of Rocket Propulsion

The Rocket Equation

Rocket engine develops its thrust ‘F’ by expelling gas at a high exhaust velocity (Ve) relative to the vehicle.

The gas may be produced by the combustion of a propellant.

Rocket is a M/C that develops thrust by the rapid expulsion of makes.

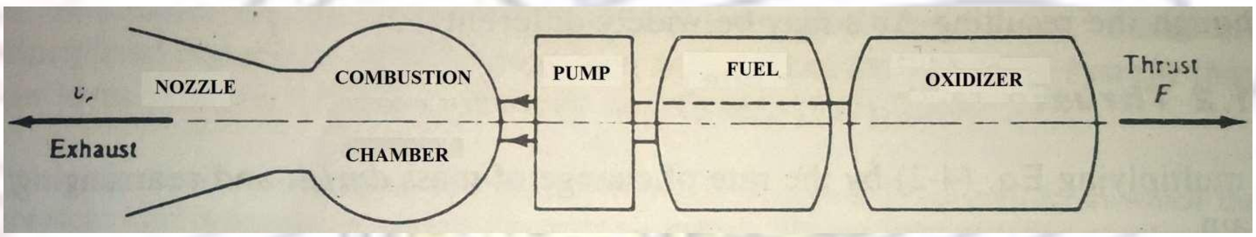


Figure: Basic Liquid Rocket Engine

Rocket propulsion is based on the principle of conservation of Momentum,

m = instantaneous mass of rocket and propellant

v = velocity of rocket in an inertial Co-ordinate system

dm' = mass of propellant ex pulled during the period, the rocket velocity changes from v to $v + dv$

v' = Velocity of the exhaust in the retrial system.

The total momentum must remain constant.

$$(m - dm')(v + dv) + v'\,dm' = mv \qquad \text{□(1)}$$

Momentum of rocket remaining propellant

But, $dm' = -dm$ and $v' = v - v_e$ { v_e = exhaust velocity relative to rocket.}

Equation becomes,

$$\frac{dv}{dm} = \text{□}\frac{v_e}{m} \quad \text{□ Equation of motion} \qquad \text{□(2)}$$

If v_e is constant, eqn. (2) is directly integrable, subject to the initial conditions

m_0 = intial mass and v_0 = intial velocity

Therefore, the increase in velocity, □$v = v - v_e$,

$$\text{□}v = v_e \ln\frac{m_0}{m} \qquad \text{□(3)}$$

It is called the rocket equation.

It determines the increase in the rocket's velocity Δv in the absence of external forces for given initial mass m_0 and final mass m and is independent of variation of mass with time.

$$\Delta v = -v_e \ln\left(1 - \frac{\Delta m}{m_0}\right) \quad \text{Rocket Equation} \qquad (4)$$

It determines the increase in rocket velocity, Δv in the absence of external forces with m_0 and m.

$$\text{Mass ratio } \Gamma = \frac{m_0}{m} = \frac{\text{Initial total mean of rocket including propellant}}{\text{final mass of rocket after propellant has been consumed}}$$

$$\text{Propellant mass, } \Delta m = m_0 - m$$

$$\text{Propellant mass fraction, } \zeta = \frac{\Delta m}{m_0} = 1 - \frac{1}{\Gamma},$$

- In an efficient rocket, the value of ζ usually exceeds 85%.
- Solving for Δm, we obtain

$$\Delta m = m_0\left[1 - \exp\left(-\frac{\Delta v}{v_e}\right)\right] \qquad (5)$$

- It gives propellant mass required to attain a given increase in velocity for a given exhaust velocity.
- The exhaust velocity is a measure of the efficacy of the propellant and, to a lesser extent, that of the rocket nozzle.
- The required value of the final velocity ***v*** depends on the orbit to be attained and the final mass ***m*** is the sum of the rocket structure and payload; hence the magnitude of $\boldsymbol{v_e}$ is the characteristic of the propellant that largely determines the payload capacity of the rocket.

- It is important to note eqns. (3) and (5) are applicable to any rocket propulsion system, regardless of the means to impart velocity to the exhaust.
- They apply equally well to chemical rockets, ion propulsion, and photon engines, although the resulting Δv's may be widely different.

Thrust

By equation of motion, $\frac{dv}{dm} = -\frac{v_e}{m}$

g1dm/dt and rearranging,

$$m\frac{dv}{dt} = -\frac{dm}{dt}v_e \qquad (A)$$

This gives equation of motion of rocket which specifies its acceleration called "Momentum thrust".

From the principle of momentum for system particles, time rate of change of the total linear momentum 'P' of the system is equal to net external force,

$$\frac{d\mathbf{P}}{dt} = \sum_{i=1}^{n} F_i$$

Where, n = no. of external forces

$$\frac{d\mathbf{P}}{dt} = \frac{d}{dt}(m\mathbf{v}) + \frac{d}{dt}(m'\mathbf{v}')$$

where, m' – total mass
m' – total man of the particles expelled in the exhaust
$\mathbf{v}'$ – velocity if exhaust
$\mathbf{v}_e$ – velocity of exhaust relative to rocket.

At the given instant $m'=0$ but $\frac{dm'}{dt} = -\frac{dm}{dt}$

Also, $\mathbf{v}' = \mathbf{v} - \mathbf{v}_e$, where $\mathbf{v}_e$ - velocity of the exhaust relative to the rocket

Therefore, at $t =$

$$\frac{d\mathbf{P}}{dt} = \frac{d}{dt}(m\mathbf{v}) + \frac{dm}{dt}(\mathbf{v} - \mathbf{v}_e)$$

Substituting this expression into eqn. () and simplifying, we obtain

$$m\frac{d\mathbf{v}}{dt} = -\frac{dm}{dt}\mathbf{v}_e + \sum_{i=1}^{n} F_i \quad \text{(D)}$$

where, is arbitrary.
This equation is valid for all times.
This is the equation of motion of the rocket alone.

Equation (D) has the superficial appearance of Newton's second law, $F = m\frac{dv}{dt}$, if the momentum thrust is regarded as additional force on rocket and man is considered as time–dependent.

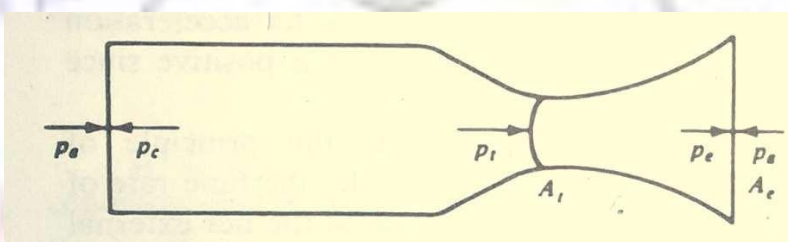


Figure: Nozzle Expansion

In general, pressure p_e of the exhaust at exit plane of the nozzle will not be exactly equal to ambient pressure p_a of the atmosphere.

The net external force on the rocket along the rocket axis thus includes the constraint $-F$ of the test stand and a force $(p_e - p_a)A_e$ due to the difference in pressures of the exhaust and the atmosphere over the area A_e of the exit plane of the rocket nozzle as shown in Figure.

Therefore, since $\frac{dv}{dt} = 0$, we obtain

$$F = \frac{dm}{dt} v_e + (p_e - p_a) A_e$$

Total thrust on rocket = momentum thrust and pressure thrust

Pressure of the exhaust at the exit plane of nozzle is determined by, "expansion ratio"

$$\epsilon \equiv \frac{A_e}{A_t} \quad \{\equiv \text{ means 'identical to'}\}$$

where, A_e - Exit area

A_t - Throat area

- If ϵ is too small – the nozzle will be underexpanding because $(p_e > p_a)$
- If ϵ is too large – the nozzle will be overexpanding because $(p_e < p_a)$
  Either situation will result in a reduction of exhaust velocity.
- Thermodynamics considerations imply that for any given value of p_a, the thrust is maximum when $(p_e = p_a)$. This condition is known as optimum expansion.

- It was assumed that the momentum flux was axial but Rn practice; three is always some radial component of momentum due to divergence of the nozzle.
- Therefore axial momentum thrust must be corrected by a factor '>'.
- For conical nozzle, having apex angle 2α, base rading R_e and side r = R_e / sinα, the total man rate is,

$$\dot{m} = \int_A \rho v_e dA = \rho v_e A = 2\pi r^2 (1 - \cos\alpha) \rho v_e$$

where, $dA = 2\pi r^2 \sin\phi \, d\phi$ is an angular element of area and A is the total area as a spherical cap of radius r over the nozzle exit.

The momentum flux in the axial direction across this spherical cap is thus

$$P = \int_A v_e \cos\phi \, d\dot{m} = \int_A \rho v_e^2 \cos\phi \, dA = 2\pi r^2 \rho v_e^2 \int_A \sin\phi \cos\phi \, dA = \pi r^2 (1 - \cos^2\alpha) \rho v_e^2 = \lambda \dot{m} v_e$$

where, the correction factor is $\lambda = \frac{1}{2}(1 + \cos\alpha)$

Taking the account of both exist pressure mismatch and non-axial flow, the equation may be expressed as
F = Am·Ve + (Pe - Pa) Ae = m·C
where,
C → effective exhaust velocity,
V_e → actual exhaust velocity,

Thrust co-efficient, **$C_F = F / A_t P_C$**

where, P_C → chamber pressure

Characteristic exhaust velocity, → **$C_K = C / C_F$**

Assuming effective exhaust velocity as constant, **$\Delta V = C\, ln(m_o/m)$**

- **Specific Impulse:-**

Used to describe the propellant performance, and it is defined as

$$I_{SP} = \frac{\text{thrust (unit of force)}}{\text{Rate of propellant flow (unit of width/time}}$$

$= F/\dot{w} = F/\dot{m}g$

where, $\dot{m}$ → mass flow rate of propellant

g → acceleration due to gravity, ≈ 9.80665m/s2

in terms of effective exhaust velocity

$I_{SP} = C/g = C_K C_F/g$

Vacuum specific impulse,

$I_{SP} = V_{AC} = 1/g\ (AV_e + PA_e / \dot{m})$

Rocket equation,

$\Delta V = C\, ln\, (m_o/m)$

$= I_{SP}\, g.\, ln(m_o/m)$ {∵ $I_{SP} = C/g$ }

$\Delta V = -I_{SP}\, .\, g.\, ln\, (1 - \Delta m/ m_o)$

where,

m_o → initial mass,

m → final mass,

$\Delta m = m_o – m$ → propellant mass,

∴ $\Delta m = mo \left[1 – CMP\, (-\Delta V/ I_{SP}.g) \right]$

In solid propellant, which have fixed change of propellant,

∴ total impulse, $I_t = \int_0^{t_p} F(t)dt = Ft_p$

where, F → average thrust, tp → burn time

∴ **$I_t = -I_{SP}\, .\, m_p\, g.$**

- **Chemical population:-**

(a) Nozzle thermodynamic:-

If a chemical propellant is used the exhaust velocity (v_e) of the molecular expelled is related to the absolute combustion temperature (T_C) and the molecular weight (M) of the gas.

By equation partition of energy, $½ m_o V_e^2 = 3/2 KT_C$

where, K Boltzman’s constant

mo mass of one molecule

$V_e = \sqrt{3R/M T_C}$

Where,

$K = R/N_0$, $M = N_0/m_0$; R universal gas constant

$R = N_0 – K$, $N_0 = A_0 . M$ N_0 Averageardo’s number

Realistic value of exhaust velocity is calculated in terms of

- Combustion temperature
- Chamber pressure
- Modular weight
- Specific heat ratio with first law of thermodynamics and estimation of state.

For ideal rocket, certain simplifying assumptions are mode such as,

- Ideal gas
- Perfect fluid (no viscosity)
- Steady, one – dimensional flow
- In compressible flow (constant density and no shock waves)
- Adiabatic flow (no heat transfer)
- Frozen flow (composition has no time to vary)

(b) Propellant chemistry:-

Analytic thermodynamic formulas require knowledge of temperature (T_C) of reaction in combustion chamber molecular weight (M) and specific next ratio (γ) of exhaust.

These quantities can be calculated if the chemical reaction is known and frozen composition is assumed.

Temperature of reaction is determined by total enthalpy of products and reactants in combustion chamber.

$$H_C = \sum n_j H_j(T_0) = \sum n_j H_j(T_C)$$

Reactants product

where, H enthalpy

$H = U + P_V$

$U = C_V T$; $PV = nR_T$

Molecular weight,

$C_P = 1/m \sum n_j (C_P) j$
products

Specific heat ratio, $\gamma = C_P/C_V$

$C_V = C_P – n’R$

$n_1 = 1/m \sum n_j$
gases

➢ **Solid Propellant:-**

In solid propellant rockets, addition constraints are summarized a part from chemical propellant.

At equilibrium,

Rate of flow of mass through the nozzle exist (m) = rate of consumption of propellant mass (mP)

$$m = mP = A_b\, rP_b$$

A_b area of burning surface

r burning rate

P_b propellant man density

Thrust, $$F = m\, C = A_b r P_b I_{SP}\, .g$$

$$A_b r P_b = AtPc/Ck \qquad PC = krPbck$$

where. PC combustion pressure

K = Ab /At

Burning rate, r = aPcn [a and n are constants

- **Solid propellants are temperature dependents, hence temperature sensitivity**

  = 1/F dF/dT = d/dt*ln*F

  Correction on thrust,

  $$F = F_O (e^{T} - 1)$$

  F_0 uncorrected average thrust

  Ion engine:-

  High exhaust velocity can be achieved by ion engine

  Metallic ions such as mercury and cesium are ionized a accelerated by an electrical field.

  $\frac{1}{2} mV_e^2 = qv$

  where,

  q charge of ion

  m mass of ion

  q accelerating voltage

- Chemical rockets because of high thrusts used in powered flight and orbital maneuvers, but ion engine Fn restricted to low thrust maneuvers.
- Design life:-

  Mission life 'V' is defined at the end of which the service is truncated.

  Average life 'τ'

  $$T = \int_0^U \tau\, tedt + e\text{-}v/T \int_0 t\,(t - V)dt = T(1 - 0\text{-}U/T)$$

- Second term has delta function normalized by factor **– 0-U/T**

  T Mean time to failure

  If time to replace a satellite is Te, the average time of unavailability during L (system life time) is LTR/PT

  system availability; **$A = 1 - T_R/T_T$**

  If there is an "in – orbit spare", then

  System availability, $A = 1- 2\,(T_R/T_T)$

- Systems using more than one satellite to cover the operating territory, then each satellite is consider as "Bernoulli trial".

➢ If the satellites are interchange, a system of ‘N’ operational satellite and ‘S’ spares, probability of at least ‘n’ operating satellites, for exactly
‘K’ sorceries is,

$$P (K) = (K_n)\ P_k(1 – P)_{n – k}$$

⍰Binomial coefficient, ⍰ $n1 / k_i(n – k)!$

⍰for S =1, (only one spare), then $R = P_N[N(1 – P) + 1]$

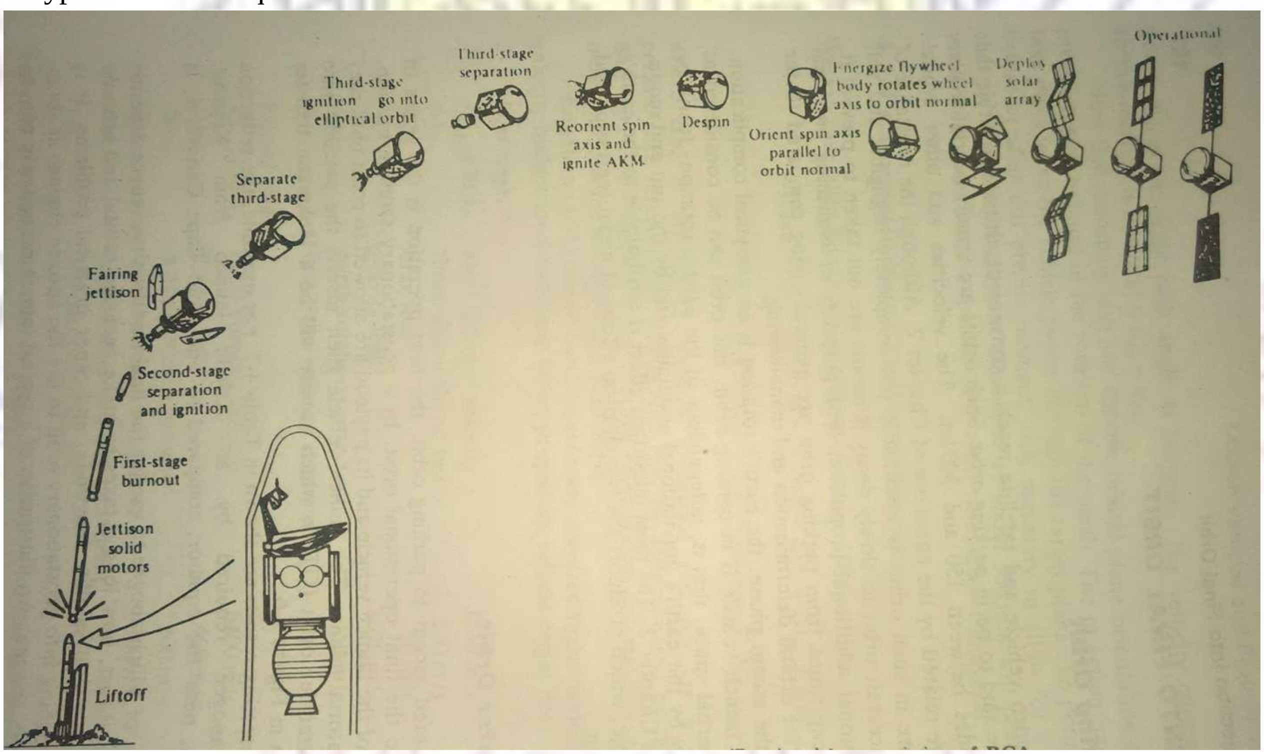
Launch from an expandable Launcher

15. With a neat sketch show the various stages involved in satellite launch.*(Nov/Dec 2012, May/June 2012)

- A Satellite cannot be placed into a stable orbit unless two parameters that are uniquely coupled together- the velocity vector and the orbital height.
- To make the most efficient use of fuel, it is common to shed excess mass from the launcher as it moves upward, this is called “staging”.

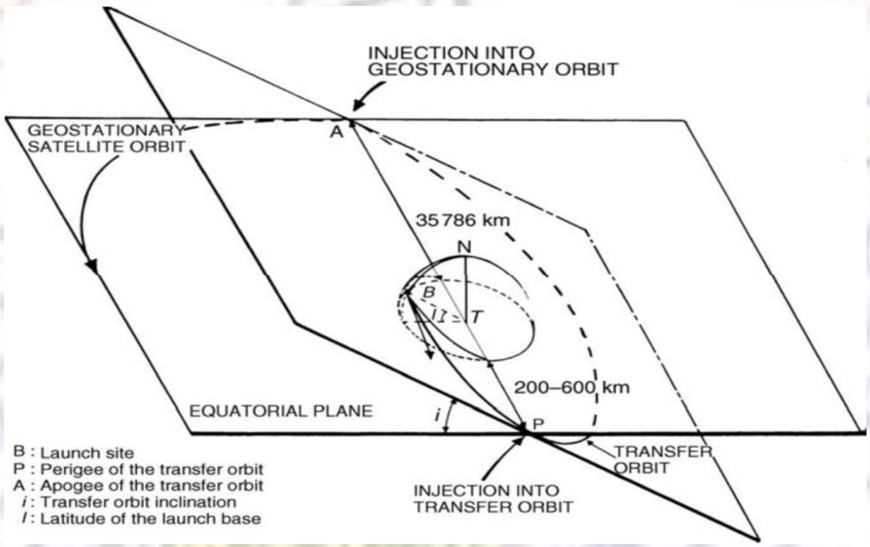
Launch from an expandable Launcher

- A typical launch sequence is shown below:



- A satellite is launched in an easterly direction as close to the equator and minimizes the fuel required for reducing the inclination to zero. To minimize drag from the atmosphere a satellite is launched vertically.
- The vehicle is gradually tilted by its guidance system during the flight until, at the point of injection, it is tilted by 90^0 in an easterly direction. After a few minutes the first stage rockets are burnt and jettisoned.
- The second stage is ignited soon after. When the initial parking orbit-typically between 185 and 250Km- is reached the ignition of the second stage is cut off. The satellite together with the remaining second-stage rocket drifts in the parking orbit.

- Shortly before reaching the equator the second stage rocket is re-ignited. This stage is burnt to depletion, followed by a burn to depletion of the third stage. This maneuver injects a satellite into the elliptical transfer orbit and the payload is separated from the launch vehicle.
- The satellite trajectory is closely monitored by a network of tracking stations.
- An apogee kick motor is then fired at the apogee of the transfer orbit.
- The firing of the apogee kick motor modifies the transfer orbit into nearly circular geosynchronous orbit.
- The satellite begins to drift slowly with respect to the earth and hence this phase is referred as “drift phase”. During the operational phase, drifts caused by various perturbations are corrected periodically to maintain the satellite position within the specified limits.



Sequence for launch and injection into transfer and geostationary orbits with an expendable launch vehicle.

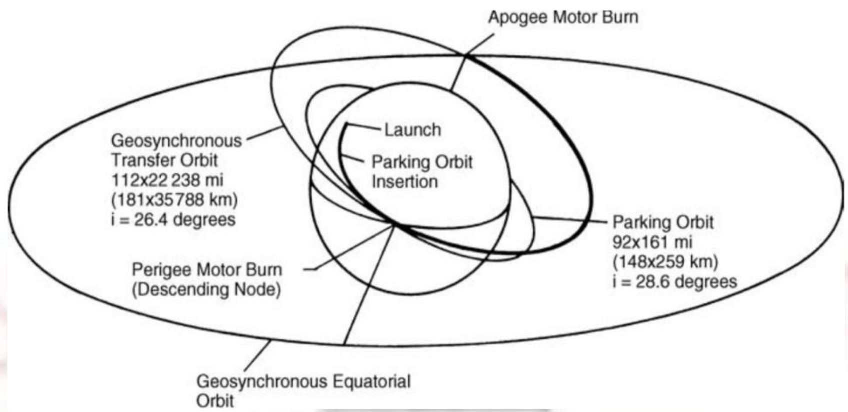
Launch from a Reusable Launcher

- ➢ The expandable launchers lose most of the expensive hardware during launch.
- ➢ The shuttle can only launch satellites in low earth orbits and therefore additional propulsion is necessary to inject a satellite into the geostationary orbit.
- ➢ The heavy lift capability of the shuttle is effectively used for carrying shuttle upper stages for the extra propulsion.
- ➢ Various stages have been developed.

- **Perigee stages**
  - ➢ These stages provide the perigee thrust to inject a satellite into a geostationary transfer orbit.
  - ➢ The geostationary orbit is attained by firing an apogee-kick motor.

- **Integrated stages**
  - ➢ These types of upper stages combine the perigee and apogee motors into a single package.
  - ➢ The space craft is placed by the shuttle into a parking orbit of ~290km at an inclination of $28^{0.}$
  - ➢ At this stage the satellite is injected out of the shuttle.
  - ➢ The thrust to attain the geostationary transfer orbit is applied at the equator using a suitable upper stage.

- The geosynchronous orbit is achieved by applying the required thrust at the apogee.

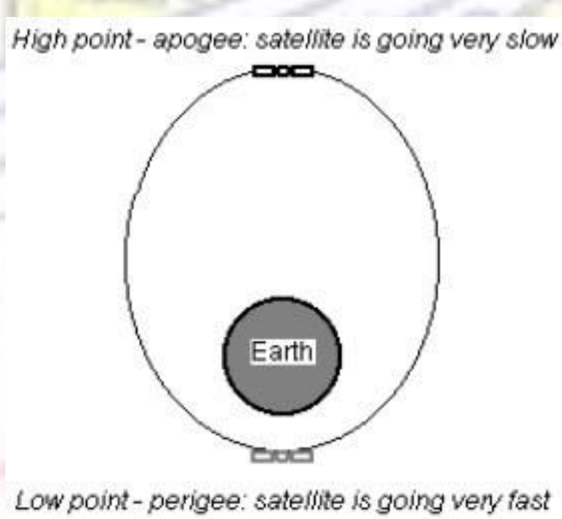


Geostationary orbit injection sequence from a low altitude parking orbit.

Launching of geostationary satellite:

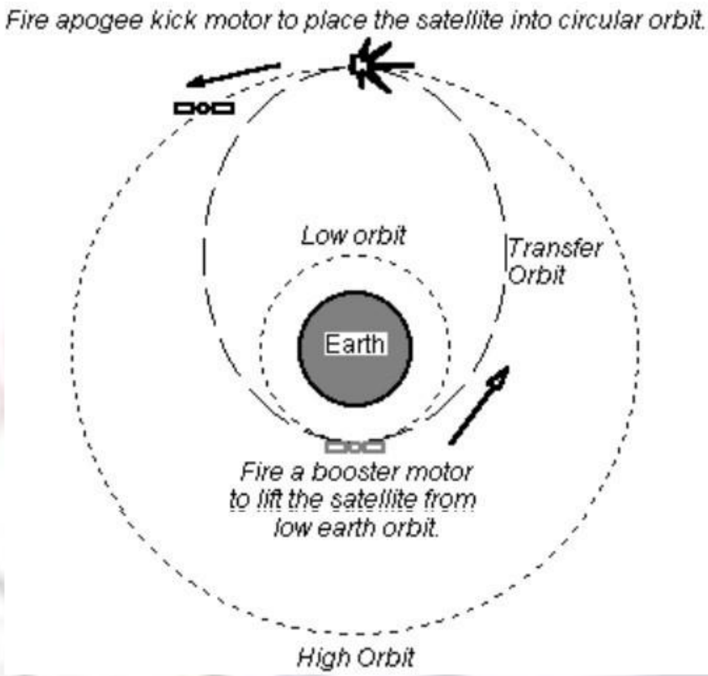
16. Give a detailed note on launching vehicles and the procedures employed for launching spacecraft in GEO orbits. (May/June 2012)
17. Describe the steps involved in locating the satellite in the orbit. [Nov/Dec 2022]

- Initially place spacecraft with the final rocket stage into LEO.
- After a couple of orbits, during which the orbital parameters are measured, the final stage is reignited and the spacecraft is launched into a geostationary transfer orbit(**GTO**).
- After a few orbits in GTO, while the orbital parameters are measured, a rocket motor (**AKM**) is ignited at apogee and GTO is raised until it is circular geostationary orbit.
- AKM (Apogee Kick Motor) is used to circularize the orbit at GEO.



Geostationary Transfer Orbit:-

Firing a rocket motor at apogee iscalled "apogee kick", and the motor is called the "**apogee kick motor".**



- Installing a geostationary satellite into orbit using a launcher with several stages requires three Phases

Launch phase

- From launcher take-off to injection at the perigee of the transfer orbit, the actions are as follows:
  - Increase the altitude in order to achieve the altitude of the perigee.
  - Drop the fairing after passing through the dense layers of the atmosphere.
- Bring the last stage–satellite assembly onto a trajectory which intersects the equatorial planeparallel to the surface of the earth with the required velocity on passing through the equatorial plane (the perigee of the transfer orbit).

Transfer phase

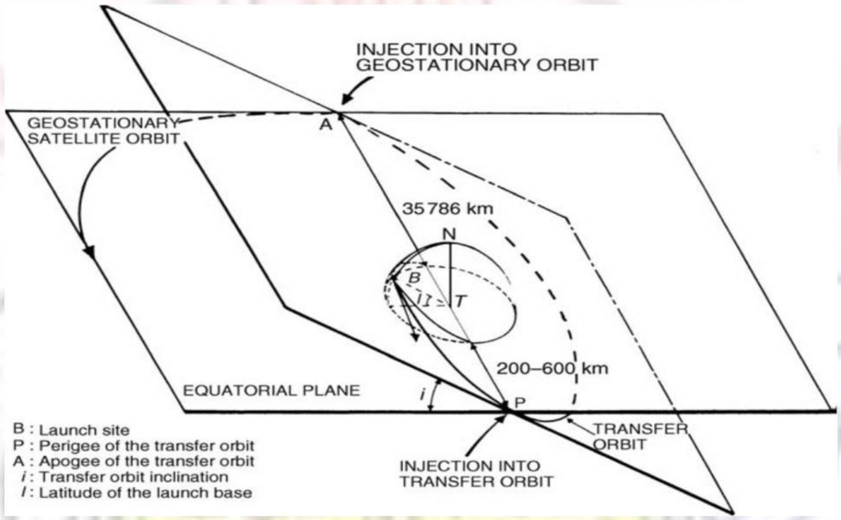
- The transfer phase starts with injection of the composite satellite–launcher final stage and terminates with injection into the quasi-geostationary satellite orbit at the apogee of the transfer orbit. In this phase, the actions are as follows:
  - Separate of the satellite and the final stage.
  - Determine orbit parameters.
  - Measure the satellite attitude.
  - Correct the satellite orientation in view of the apogee maneuver.
- The orientation may be maintained either by causing the satellite to rotate (spin stabilization) or by active attitude control using sensors and actuators (three-axis stabilization).

Positioning phase

- This phase starts with injection into the quasi-geostationary satellite orbit at the apogee of the transfer orbit and terminates with positioning of the satellite at the chosen station in the geostationary satellite orbit.

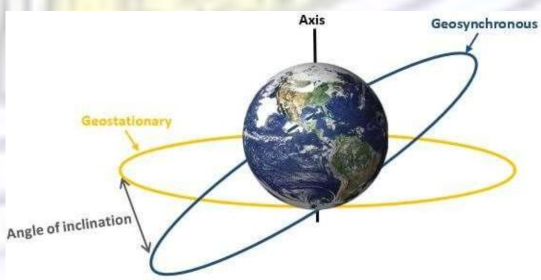
Other procedures

- Other procedures for installation into orbit which permit the launch vehicle to move out of the plane defined by the launch azimuth and the latitude of the launch base ('dog leg' maneuvers) can be envisaged.
- In this way, by reigniting the last stage in flight, the Proton launcher permits direct injection of the satellite into the geostationary satellite orbit.
- This approach avoids the need for an apogee motor and thus permits launching of satellites with larger useful mass.



Sequence for launch and injection into transfer and geostationary orbits with an expendable launch vehicle.

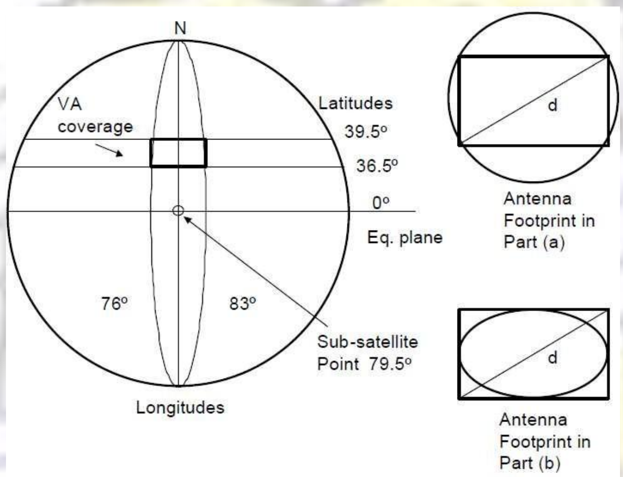
Differentiate between Geosynchronous and Geostationary orbits. [Dec 2020, May 2021]



Sl. No.	Geosynchronous orbits	Geostationary orbits
1	The orbit around the Earth with orbit period equal to one sidereal day (i.e. 23 Hrs, 56 minutes, 4 seconds) is known as Geosynchronous orbit. The word "synchronous" means object in this orbit returns to the same position after period of 1 sidereal day to the observer on the Earth surface.	Object in this orbit has period equal to rotation period of the earth. Hence it appears motionless from earth or at fixed position to observers on the ground w.r.t. his/her position. This is similar to how star looks to observers on the earth. Hence the word "stationary".
2	There many such orbits around the Earth.	There is only one such orbit around the Earth.
3	It may be circular or non-circular types.	It is circular orbit.
4	Geosynchronous satellite has inclination with respect to equator.	Geostationary satellite has zero inclination with respect to equator.

The state of Virginia may be represented roughly as a rectangle bounded by 39.5° N latitude on the north, 36.5° N latitude on the south, 76.0° W longitude on the east and 86.3° W longitude on the west. If a geostationary satellite must be visible throughout virginia at an elevation angle no lower than 20°, what is the range of longitudes within which the sub-satellite point of the satellite must lie? [Dec 2020, May 2021]

The State of Virginia can be represented approximately on a map as an area bounded by 39.5° N latitude, 36.5° N latitude, 76.0° W longitude, and 83.0° W longitude. A geostationary satellite located at 79.5° W longitude has an antenna with a spot beam that covers all of Virginia at a downlink center frequency of 11.155 MHz. In this problem you will estimate the antenna dimensions subject to two different assumptions. In both cases use an aperture efficiency of 65 percent.



a. The antenna is a circular parabolic reflector generating a circular beam with a 3 dB beamwidth equal to the diagonal of the area bounding the State of Virginia. Estimate the length of the diagonal by measuring the distance on a map of the US, and calculate the beamwidth of the antenna from simple geometry. Hence determine the diameter of the antenna on the satellite in meters and its approximate gain in decibels.

Answer: The diagram above shows the boundaries of the rectangle enclosing the State of Virginia. Measurement on a map shows that the diagonal pf the rectangle is approximately 800 km. The diagonal can be calculated from the lengths of the sides of the rectangle using the given altitude and longitude boundaries.

$D_{N-S} = 6378 \times 3.0/57.3 = 334$ km

$D_{E-W} = 6378 \times 7.0/57.3 = 779$ km

Diagonal $d = [D_{E-W}^2 + D_{N-S}^2]^{1/2} = 848$ km

Working in the N-S longitude plane at 79.5° W, the distance from the satellite to the center of the rectangle at Lat 37.5° N, Long 79.5° W is s where a = GEO radius, r_e = earth radius

$s^2 = a^2 + r_e^2 - 2\, a\, r_e \cos 37.5° = 1.3918 \times 10^9$ hence $s = 37{,}307$ km

The beamwidth of the antenna beam at the satellite can be found as a first approximation from the length of the diagonal of the rectangle. Using the map value of d = 800 km:

$\theta_{3\,dB} = d / s$ radians $= 57.3 \times 800 / 37{,}307 = 1.23°$.

For an antenna operating at 11.155 GHz, $\lambda = 0.02689$ m, the antenna diameter is $75\ \lambda/D$ giving

$D = 75 \times 0.02689 / 1.23 = 1.640$ m.

The gain of the antenna, with aperture efficiency of 65%, can be found from

$G = \eta_A \times (\pi D / \lambda)^2 = 0.65 \times (\pi \times 1.640 / 0.02689)^2 = 23{,}862$ or 43.8 dB

b. The antenna is an elliptical parabolic reflector with 3 dB beamwidths in the N-S and E-W directions are equal to the height and the width of the area bounding the State of Virginia. Find the N-S and E-W dimensions from a map of the US, and use geometry to calculate the required 3 dB beamwidths of the satellite antenna. Calculate the approximate gain of the antenna.

Answer: Dimensions of the rectangle from a map are approximately 330 km in the N-S direction and 750 km in the E-W direction. Similar dimensions based on the latitudes and longitudes are given in part (a) above. In this case, the dimensions of the antenna must create an elliptical footprint that fits inside the rectangle. Ignoring earth curvature and using the distance from the center of the rectangle to the satellite, s = 37,345 km, the N-S and E-W beamwidths are approximately

$\theta_{N-S} = y_{N-S} / s$ radians $= 57.3 \times 330 / 37{,}307 = 0.51°$

$\theta_{E-W} = x_{E-W} / s$ radians $= 57.3 \times 750 / 37{,}307 = 1.15°$

The antenna dimensions are

$D_{N-S} = 75 \times 0.02689 / 0.51 = 3.95$ m

$D_{E-W} = 75 \times 0.02689 / 1.15 = 1.75$ m

The gain of the antenna can be found from

$$G = 33{,}000 / (0.51 \times 1.15) = 56{,}273 \text{ or } 47.5 \text{ dB}$$

A ground station lies at latitude = 39.2906 degrees N and longitude = 280.2629 degrees E. A Geostationary satellite at radius r = 42164 km has a longitude of 280.2629 degrees E. Calculate the range and look angles (azimuth and elevation angles) to the satellite. [Dec 2020, May 2021]

Curvature of the earth in the N-S direction makes the N-S angle at the satellite smaller than the result in the calculation above where earth curvature is ignored. A more accurate result can be obtained by recalculating the distance from the satellite to the upper and lower edges of the rectangle and then using the rule of sines to find the angle at the satellite.

$$s_1^2 = a^2 + r_e^2 - 2\, a\, r_e \cos 39.5^\circ = 1.4034 \times 10^9 \quad \text{hence} \quad s = 37{,}423 \text{ km}$$

$$s_2^2 = a^2 + r_e^2 - 2\, a\, r_e \cos 36.5^\circ = 1.3861 \times 10^9 \quad \text{hence} \quad s = 37{,}230 \text{ km}$$

The angle between the line from the satellite to the earth's surface and from the satellite to the center of the earth is α where

$$\sin\alpha / r_e = \sin \text{Lat} / s \quad \text{or} \quad \sin\alpha = r_e \times \sin \text{Lat} / s$$

For the two latitudes 39.5° and 36.5° we have

$$\alpha_1 = 6378 / 37{,}423 \times \sin 39.5^\circ = 0.10841 \quad \text{and} \quad \alpha_1 = 6.223^\circ$$

$$\alpha_2 = 6378 / 37{,}230 \times \sin 36.5^\circ = 0.10191 \quad \text{and} \quad \alpha_2 = 5.849^\circ$$

The satellite antenna beamwidth in the N-S direction is $\theta_{3\,dB} = \alpha_1 - \alpha_2 = 0.374^\circ$

The difference from the approximate result is significant because it makes the antenna even larger in the N-S direction. The approximation of a flat earth is reasonable for the E-W direction. Using the new N-S beamwidth, the N-S dimension of the antenna is

$$D_{N\text{-}S} = 75 \times 0.02689 / 0.374 = 5.392 \text{ m}$$

The gain of the antenna increases to $33{,}000 / (0.374 \times 1.15) = 76{,}726$ or 48.8 dB

****