UNIT-I

Introduction and Orbital Mechanics and launchers: A brief history of satellite communications, Orbital mechanics, Look angle determination, Orbital perturbations, Orbit determination, launch and launch vehicles, Orbital effects in communication system performance

UNIT-II

Satellites:- Satellite sub systems, Attitude and Orbit Control system (AOCS), Telemetry, Tracking, Command & Monitoring, Power systems, Communication subsystems, satellite antennas.

Multiple access techniques:- Introduction, FDMA, TDMA, DAMA, Random Access.

UNIT-III

Satellite Link Design: Basic transmission theory, System noise temperature and G/T ratio. Design of down links, Satellite systems using small earth stations, Uplink design, Design for specified C/N: Combining C/N and C/I values in satellite links.

VSAT systems: Introduction, overview of VSAT systems, Network Architectures, Access control Protocols, Basic techniques, VSAT earth station engineering.

UNIT-IV

Satellite Navigation and Global positioning system: Introduction Radio and satellite Navigation, GPS position location Principles, GPS receivers and codes, Satellite signal Acquisition, GPS

Navigation message, GPS signal levels, Timing Accuracy, GPS receiver operation, GPS c/A code accuracy, Differential GPS.

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UNIT-I

-> 1945. Arthur C Clarke provide an article

Satellite communication began in October, 1957, with the launch by the U.S.S.R of a small satellite called spurnik 1. spurnik 1 carry only a Txer and didn't have communication capabilities.

The first satellite successfully launched by the USA was EXPLORER 1, SCORE (35 days) on January 31, 1958. -> First communication satellite ECHO I & II were launched as floating balloons. They are passive satellites.

The true communication satellites TELSTAR I & II were launched in 1962 and 1963. These satellites were built by BELL Telephone laboratories. -> In 1963 first geostationary satellite SYNCOM was launched by NASA and US Dept. of Defense.

The orbit choosen for The TELSTAR satellities caused early failure of the electronics on board.

The most critical step was in August 1962, when the U.S congress passed the communication satellite Act. This set the stage for commercial investment in an international satellite organisation, on July 19, 1964; first 12 countries, to investing Intelsat (International Telecommunication satellite Organisation).

The first 'INTELSAT' satellite was launched on April 16, 1965. The satellite weighed 36 kg and incorporated two 6/4 GHz transponder.

CANADA was the first country to build a national telecommunication system using 'GEO' SATELLITE.

ANIK 1A was launched in May, 1974.

The commercial success of 'INTELSAT' let many nations to invest in satellite systems, within the 25 years, INTELSAT has 145 countries as 1K members.

In 1970's and 1960's, there was rapid development of 'GEO' satellite systems for international region and domestic region, telephon traffic and video distribution.

Types of satellites:-

1) Natural satellites Eg:- Moon.

2) Artificial satellites.

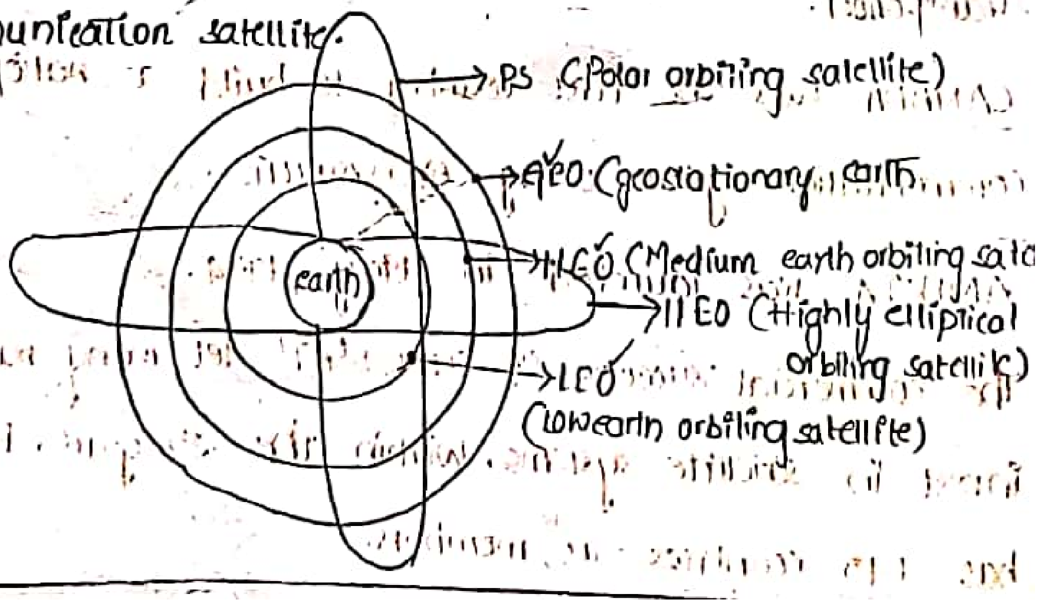
Based on orbit of earth, satellites are classified as:-

1. Geostationary satellite (GEO)
2. Medium earth orbiting satellite (MEO)
3. Low earth orbiting satellite (LEO)
4. Highly elliptical orbiting satellite (HEO)
5. Polar orbiting satellite.

Based on applications, they are classified as:-

1. Remote sensing satellite.
2. Meteorological satellite.
3. Navigation satellite.
4. Scientific & Military satellite.
5. Communication satellite.
(C-band)

Planes of orbit
1. Equitorial plane.
2. Inclined plane.
3. Polar plane.



The geostationary satellites are placed above equator at a distance of 35,000 kms.

The 'MEO' satellites operate at a distance of about 5000 to 12000 kms.

The 'LEO' satellites placed at an altitude of 500 to 1500 kms and it uses advanced compression schemes with a transfer rate of 2400 bits/second.

The 'HEO' comprises with a relatively low altitude perigy (nearest) and extremely high altitude apeogy (farest).

The polar satellites orbit from Northern hemisphere to southern hemisphere, inclined about 86° with an orbital period of 18hrs.

3 ORBITAL MECHANICS:-

The fundamental newton equation describe the motion of a satellite. We will give some co-ordinate axis within which the orbit of the satellite can be set and determine various forces on earth's satellite.

Newton's law of equations are

$$s = ut + \frac{1}{2}at^2 \quad \text{——} \quad (1)$$

$$v^2 = u^2 + 2as \quad \text{——} \quad (2)$$

$$v = u + at \quad \text{——} \quad (3)$$

$$F = ma \quad \text{——} \quad (4)$$

where, s: distance travelled from $t=0$

u: initial velocity at time $t=0$.

v: final velocity at time $t=T$.

a: acceleration

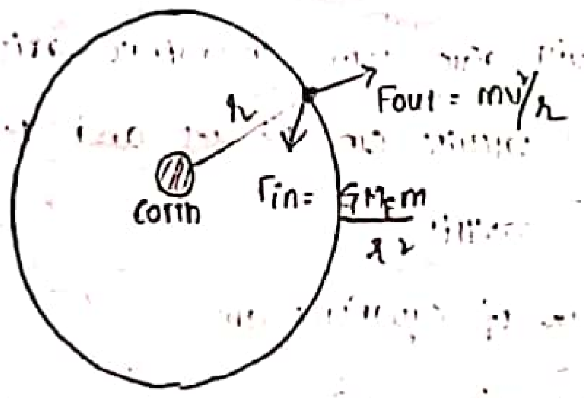
F: force acting on the object.

m: mass of the object.

The last equation ④ describes the motion of the satellite in a stable orbit. It states that force acting on a body is equal to mass of the body multiplied by the resulting acceleration of the body.

In a stable orbit, there are two main forces acting on a satellite. A centrifugal force due to kinetic energy of the satellite and a centripetal force due to gravitational attraction which attempts to pull the satellite down towards the planet. If these two forces are equal, the satellite will remain in a stable orbit.

The satellite has a mass of 'm' and is travelling with the velocity 'v' in the plane of the orbit is shown below.



The acceleration by the kinetic energy of a satellite is

$$a = v^2/r \quad —⑤$$

∴ the centrifugal force $(F_{out}) = ma = \frac{mv^2}{r}$ —⑥

The acceleration due to gravity is $a = \mu/r^2$, where

μ is the product of universal gravitational constant (G) and mass of the earth.

∴ The centripetal force $(F_{in}) = ma = \frac{mGM_E}{r^2}$ —⑦

If the forces of the satellite are balanced,

$$F_{in} = F_{out}$$

$$\Rightarrow \frac{m G M_E}{r^2} = \frac{m v^2}{r}$$

$$\boxed{v = \sqrt{\frac{G M_E}{r}}}$$

If the orbit is circular, the distance travelled by the satellite in one orbit is '$2\pi r$'.

$\therefore$ The orbital time period of the satellite $(T) = \frac{2\pi r}{v}$

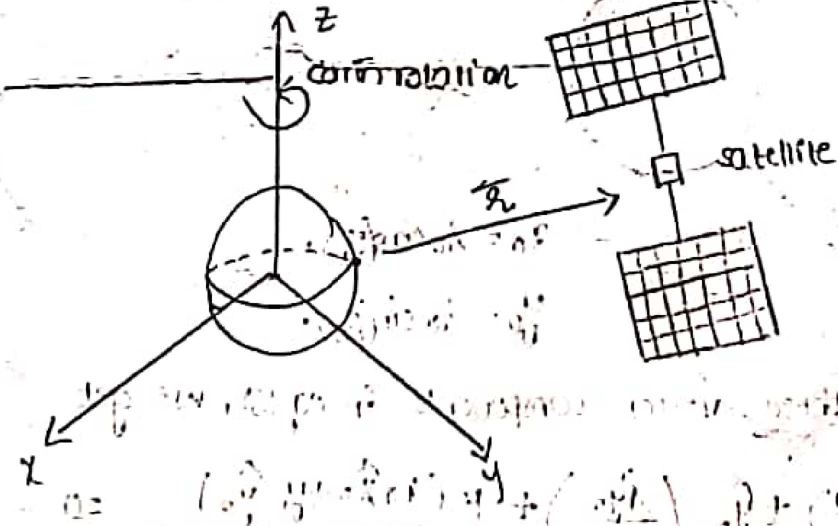
$$T = \frac{2\pi r}{\sqrt{\mu/r}} = \frac{2\pi (r)^{3/2}}{\mu^{1/2}}$$

$$\boxed{T = \frac{2\pi (r)^{3/2}}{(\mu)^{1/2}}}$$

20-6-13

* **Equation for the Radius of satellite:-**

The initial co-ordinate system that could be used to describe the relationship b/w the earth and the satellite is the cartesian co-ordinate system. A cartesian coordinate system with geographical axis of of the earth as principle axis is shown below.



If the satellite of mass 'm' located at a vector distance r from the centre of the earth.

The gravitational force "F" on the satellite is given by

$$\bar{F} = \frac{-GM_E m \bar{r}}{r^3} \quad \text{———(1)}$$

But the force due to kinetic energy of the satellite is

$$\bar{F} = \frac{m d^2(\bar{r})}{dt^2} \quad \text{———(2)}$$

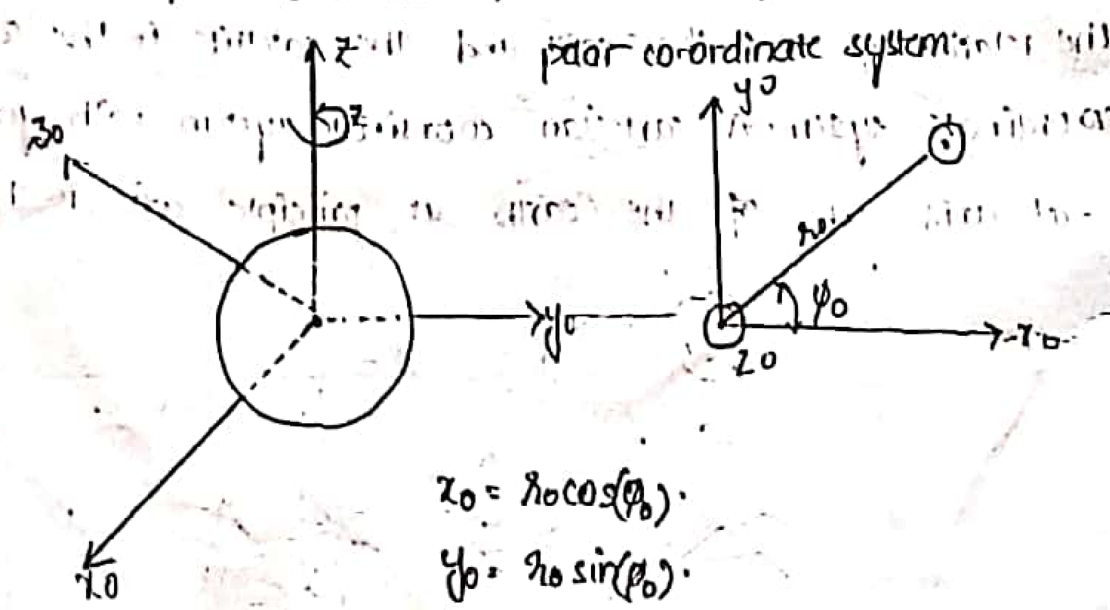
∴ From the eq ① & ②, we have,

$$\frac{-GM_E m\bar{r}}{r^3} = \frac{m d^2\bar{r}}{dt^2}$$

$$-\frac{\bar{r}}{r^3}\mu = \frac{d^2\bar{r}}{dt^2} \qquad [\text{where: } GM_E = \mu]$$

$$\frac{d^2\bar{r}}{dt^2} + \frac{\bar{r}}{r^3}\mu = 0 \quad \text{———(3)}$$

This is the 2nd order linear differential eq and its solution will involve six undeterminant constants called orbital elements. The solution fo eq 3 is difficult. In order to solve this eq, a different set of co-ordinates can be chosen, which is known as paar co-ordinate system.



$$x_0 = r_0\cos(\phi_0).$$

$$y_0 = r_0\sin(\phi_0).$$

Substituting these vector components in eq(3), we get

$$\hat{x}_0\left(\frac{d^2x_0}{dt^2}\right) + \hat{y}_0\left(\frac{d^2y_0}{dt^2}\right) + \mu\frac{(x_0\hat{x}_0 + y_0\hat{y}_0)}{(x_0^2+y_0^2)^{3/2}} = 0 \quad \text{———(4)}$$

The solution for the above equation will give the equation for the radius of the satellite 'r_0', as

$$\boxed{r_0 = \frac{p}{1+e\cos(\phi_0-\theta_0)}}$$

where θ_0 is a constant

• e, is the eccentricity of ellipse, whose semilatus rectum p is given by

$$\boxed{p = h^2/\mu}$$

where h is the magnitude of the orbital angular momentum of the satellite.

Kepler's laws of planetary motion:-

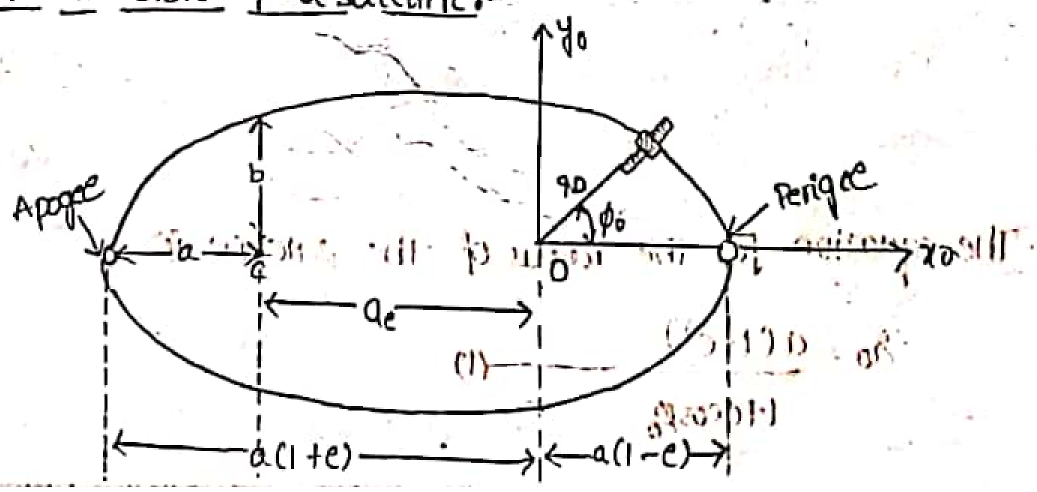
Law 1: The orbit of any smaller body about a larger body is always an ellipse, with the centre of the larger body as one the two foci.

Law 2: The orbit of the smaller body sweeps out equal area in equal time.

Law 3: The square of the period of revolution of the smaller body about the larger body equals a constant multiplied by the third power of the semi-major axis of the orbital ellipse i.e,

$$T^2 = \frac{4\pi^2 a^3}{\mu}$$ a: radius

Describe the orbit of a satellite:-



The eq. for the radius of the satellite is

$$\boxed{r_0 = \frac{p}{1+e\cos(\phi_0-\theta_0)}}$$

The quantity θ_0 serves to orient the ellipse with respect to the orbital plane axis x_0 and y_0. We always choose x_0 & y_0 plane so that $\theta_0=0$. Then the equation of the radius of the orbit becomes

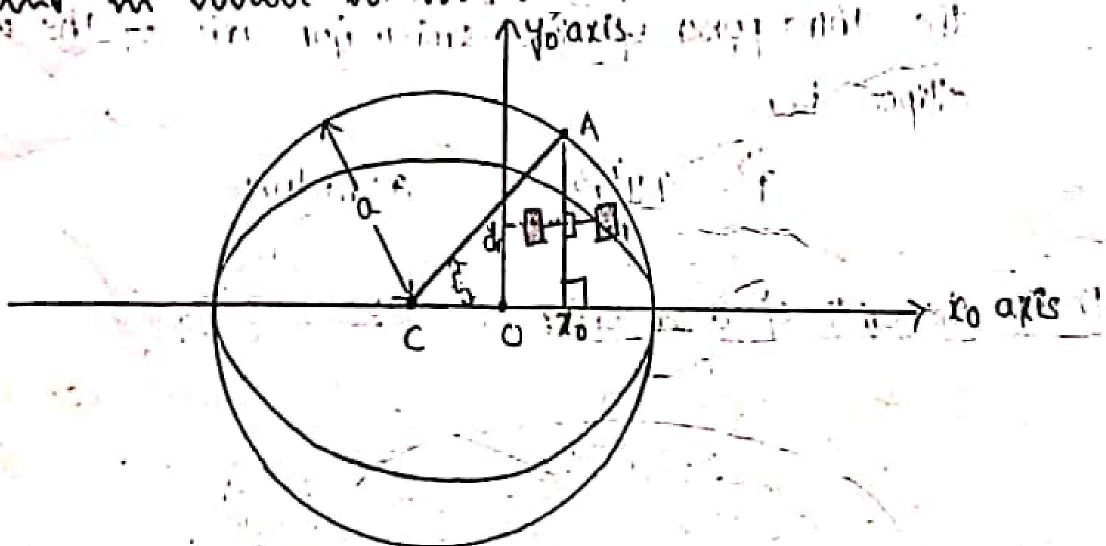
$$\boxed{r_0 = \frac{p}{1+e\cos(\phi_0)}}.$$

The lengths a & b of the semi major and semi minor axis are given by.

$$\boxed{a = \frac{p}{1-e^2}} \qquad \boxed{b = \frac{a}{p}(1-e^2)^{1/2}}$$

The point in the orbit where the satellite is closest to the earth is called 'perigee' and the point where the satellite is farthest from the earth is called 'apogee'. The pt 'o' and 'c' is the centre of the earth.

3. Locating the satellite in the orbit:



The equation for the radius of the satellite is

$$\boxed{r_0 = \frac{a(1-e^2)}{1+e\cos\phi_0}} \text{——(1)}$$

The rectangular co-ordinates are given by

$$\boxed{x_0 = r_0 \cos\phi_0 \; ; \; y_0 = r_0 \sin\phi_0}$$

To be perfectly geostationary orbit, the orbit of a satellite need to have 3 features.

1) It must be exactly circular.

2) It must be in the plane of equator.

3) It must be at a certain altitude.

So, we projected the satellite onto the circumscribed.

A line from the centre of the ellipse to the pt of projection 'A' makes an angle E with x_0 axis, and "E" is called <u>eccentric anamoly</u>. In order to calculate the eccentric anamoly, 1st we

$$\boxed{M = \eta(t - t_p) = E - e\sin E}$$

need to calculate mean anamoly, which depends on the angular velocity & time of perigee.

The average angular velocity $(\eta) = \frac{2\pi}{T}$ ③

where $T = \frac{2\pi\, a^{3/2}}{\mu^{1/2}}$

$$\eta = \frac{2\pi}{2\pi\, a^{3/2}} \mu^{1/2}$$

$$\boxed{\eta = \frac{\mu^{1/2}}{a^{3/2}}}$$

If we know the time of perigee (t_p), eccentricity (e) and length of the major axis (a), we determine the co-ordinates (r_0, ϕ_0) and (x_0, y_0).

The process is as follows.

- First of all, calculate angular velocity.

• Calculate Mean anamoly.

• Calculate eccentric anamoly.

The relationship b/w the eccentric anamoly & radius r_0 is

$$\boxed{ae\cos(E) = a - r_0}$$

• Find r_0, using the above relation.

• Solve eq (1) to calculate ϕ_0.

• Calculate x_0 & y_0 using the relations

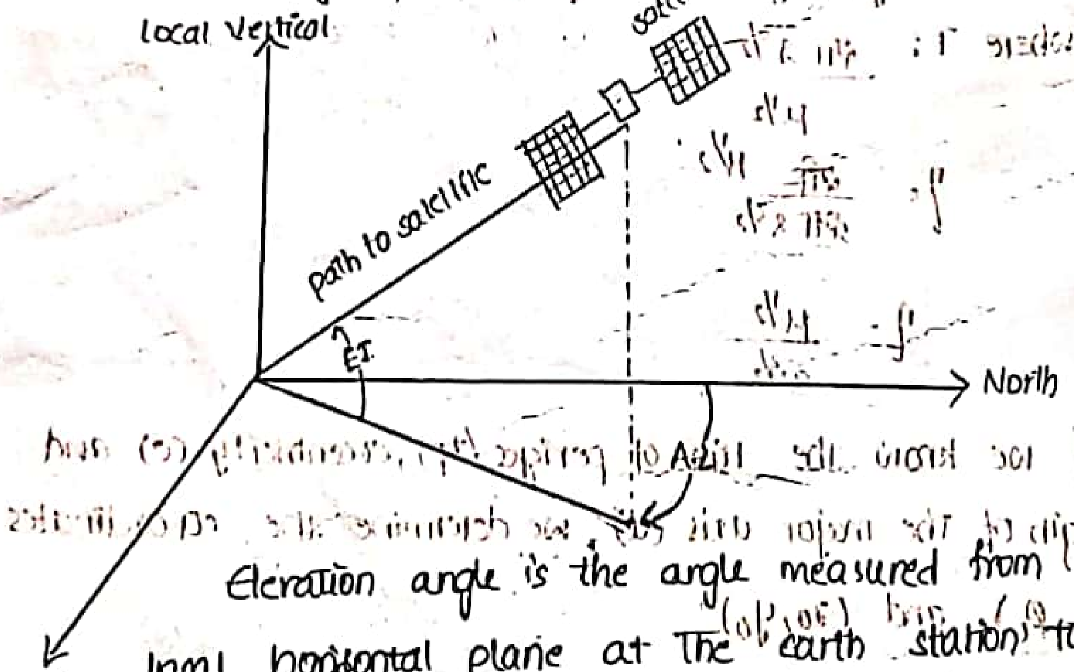
$$\boxed{x_0 = r_0\cos\phi_0; \quad y_0 = r_0\sin\phi_0}$$

LOOK-ANGLE DETERMINATION:

Look-angles: The co-ordinates at which an earth station antenna must be pointed to communicate with the satellite are called Look angles. They are most commonly expressed as

① Azimuth angle (Az).

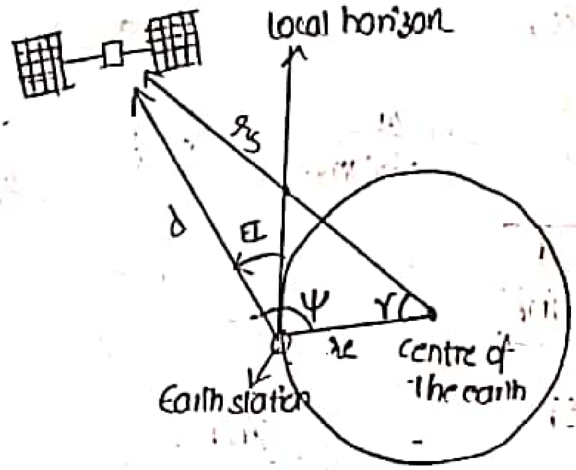
② Elevation angle (El).



Elevation angle is the angle measured from the local horizontal plane at the earth station to the satellite path.

Azimuth angle is the angle measured eastwards (clockwise) from the geographical north to the projection of the satellite on the horizontal plane at earth station.

Elevation angle calculation :-



The above diagram shows the geometry of the elevation angle calculation. Here, r_s is the vector from the center of the earth to the satellite. 'r_e' is the vector from the center of the earth to the earth station. d is the vector from the earth station to satellite. Here 'El' is the elevation angle & 'ψ' is the angle made b/w the vectors r_e and d. These three vectors lie in the same plane and form a Δle. Thus the central angle 'γ' is measured b/w r_e and r_s is related to the earth station north latitude (L_e) and west longitude (l_e) and subsatellite point north latitude (L_s), and west longitude (l_s) and the relation is given by

$$\boxed{\cos\gamma = \cos(L_e)\cos(L_s)\cos(l_s - l_e) + \sin(L_e)\sin(L_s)}$$

The law of cosine allows us to the magnitude of the vector joining the satellite and the earth station 'd' is given by

$$\boxed{d = r_s\sqrt{1+\left(\frac{r_e}{r_s}\right)^2 - 2\left(\frac{r_e}{r_s}\right)\cos\gamma}}$$

From the diagram,

$$\boxed{\psi = 90^\circ + EL}$$

By the law of sines,

$$\boxed{\frac{r_s}{\sin\psi} = \frac{d}{\sin\gamma}} \Rightarrow \frac{r_s}{\sin(90+EL)} = \frac{d}{\sin\gamma}$$

$$\boxed{\frac{r_s}{\cos(EL)} = \frac{d}{\sin\gamma}}$$

$$\cos(EL) = \frac{r_s(\sin\gamma)}{d} = \frac{\sin\gamma}{\left[1+\left(\frac{r_e}{r_s}\right)^2 - 2\left(\frac{r_e}{r_s}\right)\cos\gamma\right]^{1/2}}$$

$$\boxed{\cos(EL) = \frac{\sin\gamma}{\left[1+\left(\frac{r_e}{r_s}\right)^2 - 2\left(\frac{r_e}{r_s}\right)\cos\gamma\right]^{1/2}}}$$

Azimuth Angle Presentation:-

In order to find the azimuthal angle, an intermediate angle α must be first found. The angle α permits the correct 90° quadrant to be found for azimuth calculation. Since the azimuth angle lie anywhere b/w 0° and clockwise through 360°. Therefore the intermediate angle

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

After calculating 'α', the azimuth look angle.

Case 1: Earth's station in the northern hemisphere

a) Satellite to the SE of earthstation $A_z = 180^\circ - \alpha$

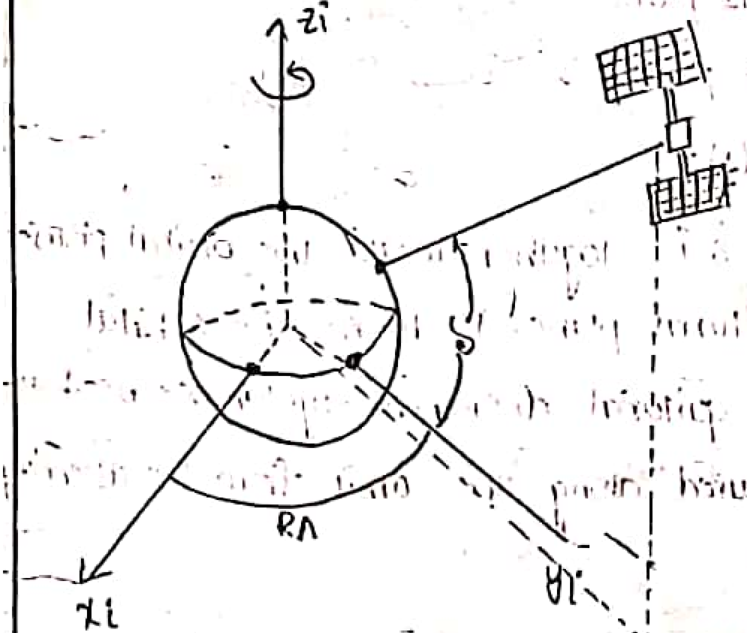
b) satellite to the SW of earth station ; $A_z = 180° + \alpha$

Case 2: Earth station in the southern hemisphere.

a) Satellite is located to the NE of earth station ; $A_z = \alpha$

b) satellite is located to the NW of earth station , $A_z = 360° - \alpha$

LOCATION OF SATELLITE W.R.T EARTH:-

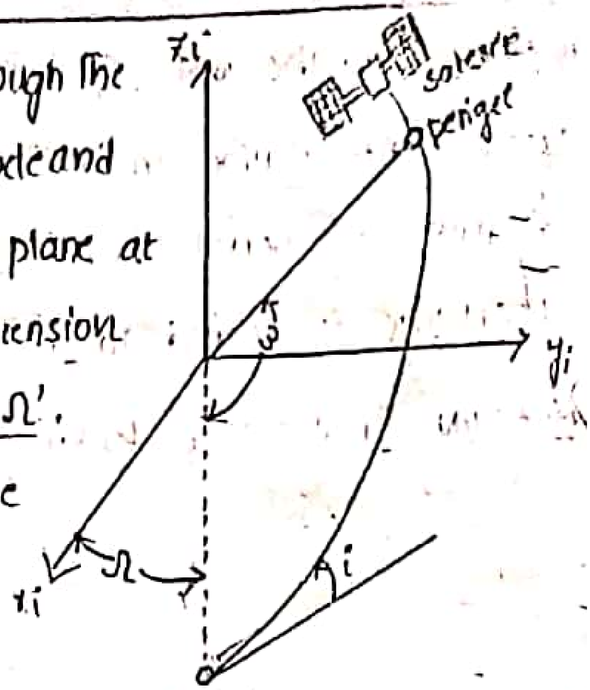


In order to locate the satellite on the rotating surface of the earth, we will begin with the geocentric equitorial co-ordinate system as shown in the fig beside.

The rotational axis of the earth is zi axis which is through the geographical north pole. The xi axis from the center of the earth towards a fixed location in space is called the first point of Aries. The xi direction is always the same whatever the earth's position around the sun. The xi, yi plane contains the earth's equator and is called equitorial plane. The angular distance measured eastward in the equitorial plane from the x-y axis is called Right Ascension and is given the symbol RA.

The two points at which the orbit penetrates the equitorial plane are called nodes. The fig is shown below indicating the nodes.

The satellite moves upward through the equitoral plane at the ascending node and downward through the equitorial plane at the descending node. The right ascension of the ascending node is called 'Ω'. The angle that the orbital plane makes with the equitorial plane is called the inclination (i).



The variables Ω & i together locate the orbital plane with respect to the equitoral plane. To locate the orbital co-ordinate system w.r.t equitorial co-ordinate system, we need ω. The ω is the angle measured along the orbit from the ascending node to the perigee.

ORBITAL ELEMENTS:-

To specify the absolute co-ordinates of a satellite at time T we need to know six quantities. This quantities are called orbital elements. They are

a) eccentricity (e);

b) semi major axis (a),

c) Time of perigee (t_p).

d) right ascension of ascending node (Ω).

e) Inclination (i).

f) argument of perigee (ω).

These are the six orbital elements.

ORBITAL PERTURBATIONS:-

~~The~~ Under ideal conditions, the orbital equations developed b/w the earth and the satellite are influenced only by gravitational attraction. In practical, the satellite & the earth responds to many other influences including assymmetry of earth's gravitational field, the gravitational fields of the sun and the moon, solar radiation pressure and non-symmetry of equitorial radius. For LEO satellites, atmospheric drag can also be important.

The approach normally adopted for communication satellite is first to derive an osculating orbit for some instant time i.e., the keplerian orbit spacecraft could follow if all the perturbing forces were removed: ~~i.e.~~ at that time, with orbital elements $(a, e, t_p, \Omega, i, \omega)$. The perturbations are assumed to cause the orbital elements vary with time and the orbit and satellite location at any instant are taken from the osculating orbit.

To visualize this process, the osculating orbital elements at time 't_0' are $(a_0, e_0, t_{p_0}, \Omega_0, i_0, \omega_0)$, then assume that orbital elements vary linearly with time at constant rates are given by.

$$\delta t\,\frac{da_0}{dt},\ e_0\,\delta t\,\frac{de_0}{dt},\ t_{p_0}\,\delta t\,\frac{dt_{p_0}}{dt},\ \Omega\,\delta t\,\frac{d\Omega_0}{dt},\ i_0\,\delta t\,\frac{di_0}{dt},\ \omega_0\,\delta t\,\frac{d\omega_0}{dt}$$

13 * Position of satellite changes due to i) longitudinal changes &
ii) inclination changes

LONGITUDINAL CHANGES:-

There is longitudinal changes in the satellite occur due to earths oblanterels.

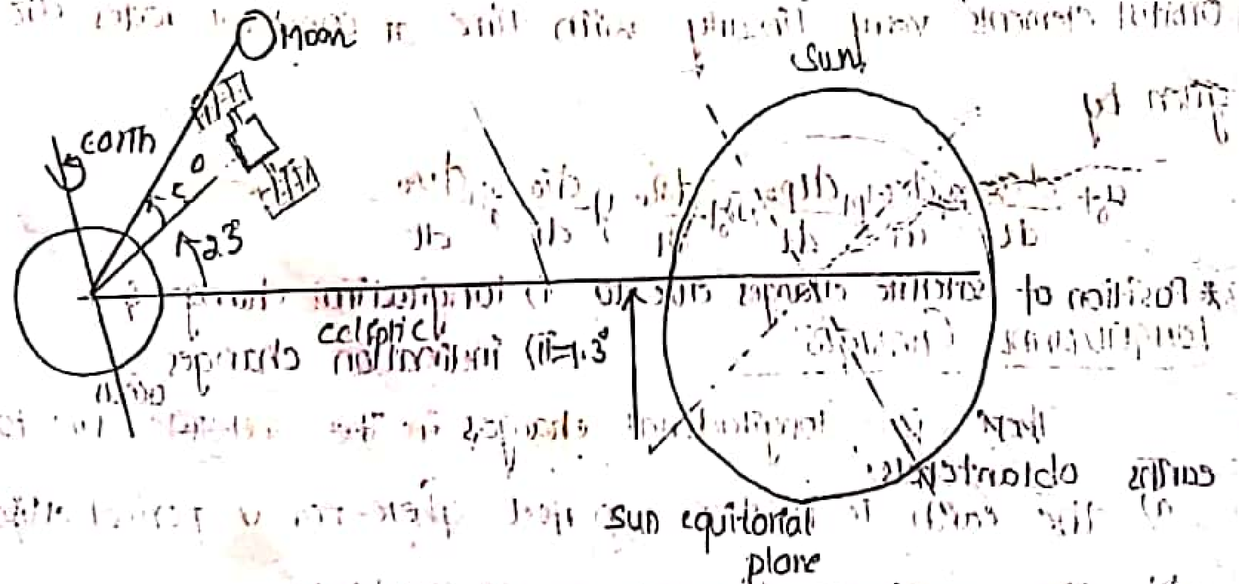
a) the earth is neither a perfect sphere nor a perfect ellipse.

b) the equitorial diameter is not constant.

4) In addition to these non regular features of the earth, there are regions where avg density of the earth appear to be high. These are referred as regions of mass concentration or Masons.

The non sphericity of the earth and non-circularity of the equitorial radius & the mascons lead to non-gravi uniform gravitational field around the earth. Therefore the force on a orbiting satellite vary with the position. This lead to longitudinal changes in the satellite. Due to the position of the mascons and equitoral bulges, there are 4 equilibrium points in the stationary orbit. 2 of them are stable & 2 are unstable points. The stable points are at about 75° east and 252° east, and unstable point are at 162° E and 348° E. If a satellite is perturbated slightly from one of the stable points, it will tend to drift back to the stable point without any cluster required. If a satellite is perturbated slightly from one of the unstable pts, it will oscillate in longitudinal position about this point.

INCLINATION CHANGES:-



The plane of the earth's orbit around the sun is called as ecliptic. Now the equitorial plane of the sun, makes an inclination of 7.3° with ecliptic. The earth makes an inclination of 23° from the ecliptic.

The moon circles the earth with an inclination of around 5° to the equitorial plane of the earth. Due to this facts of various planes such as the sun's equator, the ecliptic, the earth's equator and the moon's orbital plane around the earth are all different. Due to this difference, a satellite in an orbit around the earth will subject to variety of out of plane forces i.e., there is an inclination changes in the satellite.

The mass of the sun is slightly larger than that of moon, but the moon is considerably closer to the earth than the sun. For this reason, the acceleration force induced by the moon on the satellite is about twice as large as that of the sun.

Some maneuvers are designed to correct both inclination and longitudinal changes simultaneously.

ORBIT DETERMINATION:

Orbit determination requires

6.3 **ORBITAL EFFECTS ON COMMUNICATION SYSTEM PERFORMANCE:**

1) Doppler shift: To a stationary observer, the frequency of a moving radio transmitter varies with the transmitter's velocity relative to the observer. Take the frequency when the transmitter is at rest is (f_T). Then the received frequency f_R is higher than f_T when the transmitter moves towards the receiver and lower than f_T

when Txer moves away from the Rxer this change in frequency is called Doppler shift. Mathematically the relationship b/w Txed and Rxed frequencies is

$$\boxed{\frac{f_R - f_T}{f_T} = \frac{\Delta f}{f_T} = \frac{V_T}{c}}$$

$$\therefore \boxed{\Delta f = \frac{V_T f_T}{c} \text{ ; Change in freq } \Rightarrow \text{Doppler shift}}$$

where V_T : Velocity component / component of the Txer velocity directed towards the receiver

c : Phase velocity of light.

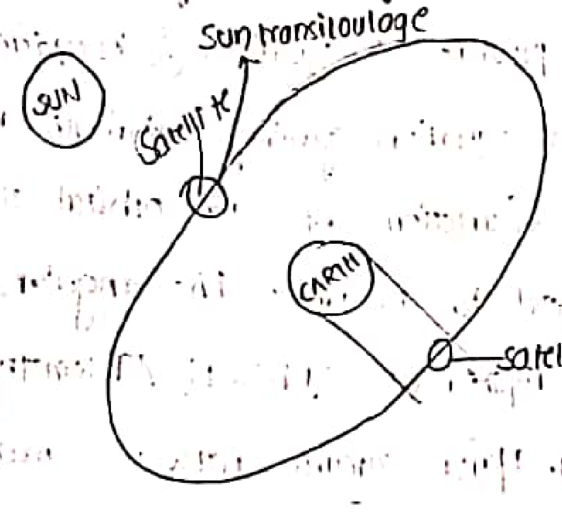
f_T : frequency of the Txer at rest.

ii) Range Variations:-

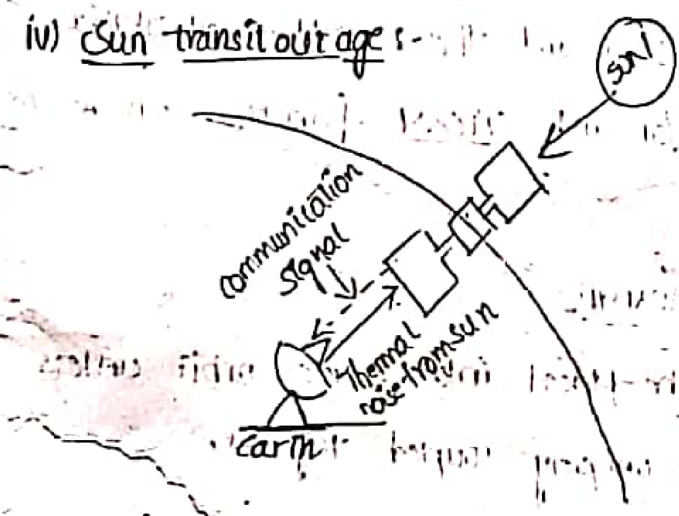
The position of the satellite w.r.t the earth exhibits a cyclic daily variations. The variation in position will lead to variation in range b/w the satellite and user terminals. If, we choose TDMA, careful attention must be paid to the timing of the frames within the TDMA, so that the individual user frames arrive at the satellite in the correct sequence at the correct time.

iii) Solar eclipse:-

A satellite is said to be in eclipse when the earth prevents sunlight from reaching it i.e., when the satellite is in the shadow of the earth. For geostationary satellites, eclipse occurs during two periods that begin with 23 days before the equinox (from about March 21 & about Sep 23). During full eclipse, a satellite receives no power from the sun and it must operate entirely from its batteries. Batteries are designed to operate with a maximum depth of discharge. The depth of discharge therefore sets the power drain limit during eclipse operation.



iv) Sun transit outage:-



The sun is a hot microwave source with an equivalent temp of about 6000-10,000K depending on the time within the 11 years, sunspot cycle. At this time, the earth station receive not only the signal from the satellite, but also the noise temperature transmitted by the sun. This added noise temperature will cause the noise margin of the receiver to be exceeded, and an out-age occurs which is sun-transit out age.

* **ORBIT DETERMINATION:-**

① Orbit determination requires that sufficient measurements to be made, to determine the six orbital elements ② three angular position measurements are needed because they are 6 unknown and each measurement will provide 2 equations. ③ Conceptually, this can be thought as one equation giving azimuth angle & other elevation angle as a function of six orbital elements.
④ The control earth station used to measure the angular position of satellites are generally refered as TTC&M (Telemetry tracking command and Monitoring). Major satellite networks maintain their own TTC&M stations around the world. Small satellite systems generally contract for such TTC&M functions from space craft manufacturer.

* **LAUNCH VEHICLES AND LAUNCH:-**

A satellite cannot be placed into stable orbit unless two parameters that are uniquely coupled together.

① orbital height
② Velocity vector.

Eg:- A GEO must be in orbit of height 36,000 kms above the surface of earth with an inclination of $0°$ ellipticity of zero and a velocity of 3074.7 m/s tangential to earth in plane of orbit.

In any earth satellite launch, the largest fraction of energy expended by rocket is to accelerate the vehicle from rest, until it is about 32 kms (20 miles).

To make efficient use of fuel system, it is common to shed excess mass from launcher as it moves upward. It is called staging:-

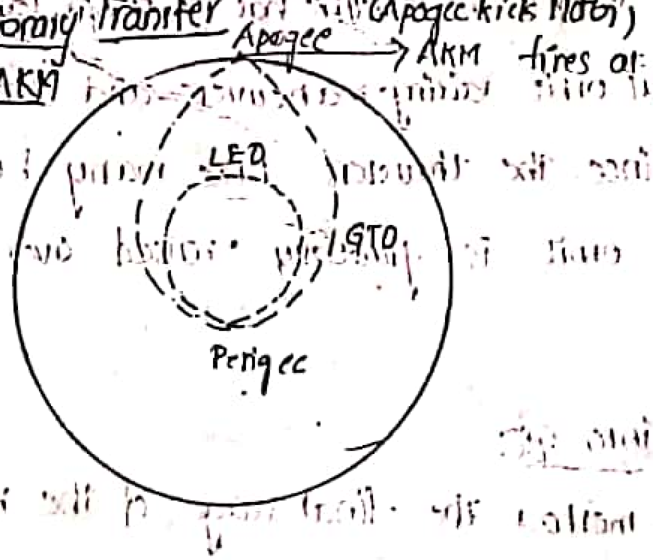
Most launch vehicles have multiple stages and as each stage is completed, that portion of launcher is expended until the final stage places the satellite into desired trajectory. Hence the term ELV (Expendable launch Vehicle). The solid rocket booster (SRB) are recovered and refurbished, for future machine mission and space settled itself, will fall back to earth. Hence the term RLV (Reusable Launch Vehicle).

Launch Vehicle Selection Features:-

1) Price /cost.
2) Reliability.
3) Performance.
4) Safety issues.
5) Launch site location.
6) Market conditions i.e., what market will bear.

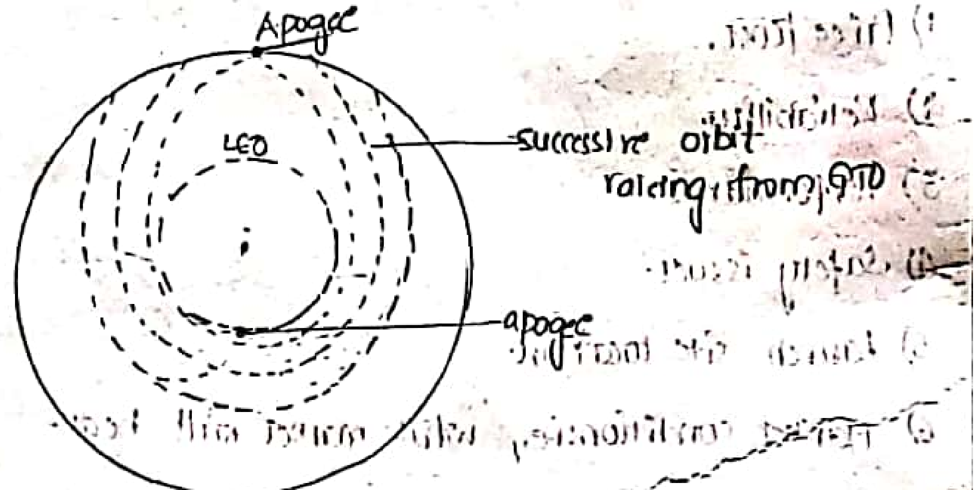
Placing satellites into GEO stationary orbit:-

1) Geostationary Transfer orbit & AKM (Apogee kick Motor)



The initial approach for launching geostationary satellites was to place the space craft into low earth orbit. Now after a couple of orbits, orbital elements are measured, then the final stage is reignited and the spacecraft is then launched into geostationary transfer orbit called as GTO. Again, after a few orbits in the GTO, orbital elements are measured; a rocket motor is ignited at apogee, and the GTO is raised until it is circular, geostationary earth orbit. Since the rocket motor fires at apogee, it is commonly referred as apogee kick motor (AKM)

Geostationary Transfer orbit with slow orbit Raising:-



In this procedure, space craft thrusters are used to raise the orbit from GTO to GEO. The satellite has 2 power levels of thrusters. One for powerful orbit raising maneuvers and other for low thrust maneuvers. Since the thrusters take many hours of operation and the orbit is gradually raised over successive thruster firing.

Direct insertion into GEO:-

In this method the final stage of the rocket places the satellite directly into GEO (geostationary earth orbit)

The spaceshuttle is an example of LEO satellite. Sometimes it orbits at an altitude of 250kms above the earth's surface. The mean earth radius is approximately 6378.14 kms. Calculate the period of the shuttle orbit when the altitude is 250kms and the orbit is circular. Find also the linear velocity of the shuttle along its orbit.

T, v

$T = 2\pi \sqrt{\frac{GM_E}{R}}$ $\quad T^2 = \frac{4\pi^2 a^3}{GM_E} = \frac{4\pi^2 a^3}{\mu}$

$a = 6378.14 + 250 = 6628.14$ kms, $\mu = 3.986004418 \times 10^5 \, km^3/s^2$

$T^2 = 28840119.49$

$T = 5370.29$ sec

Velocity $= \frac{2\pi a}{T} = 7.7548$ km/sec

one sidereal rotation $\rightarrow$ 23hr 56min 4 sec
for one day

Clarcke's orbit $\rightarrow$ Geostationary orbit

Subsatellite point $\rightarrow$ points on the surface of the earth, by which we can point the satellite.

(satellite to earth)
up to down $\rightarrow$ Nadir

down to up $\rightarrow$ Zenith
(earth to satellite)

VSAT $\rightarrow$ Very Small Aperture terminal antennas.

Input back of a transponder:

To avoid intermodulation distortions, we use input back of transponder.