

2.6 COVERAGE ANGLE AND SLANT RANGE

Communication with a satellite is possible if the earth station is in the footprint of the satellite. In other words, the earth-satellite link is established only when the earth station falls in the beamwidth of the satellite antenna. This would be a function of time and the satellite is to be tracked in case of a non-geostationary satellite. But for a geostationary satellite once the link is established, the link is available throughout the lifetime of the satellite without any tracking. To have the communication between the earth station-satellite-earth station, both the antennas of the transmitting and receiving earth station are to be pointed towards the antenna of the spacecraft. With the help of look angle determination, this can be established. To locate the earth

station in the footprint of the satellite, the information of slant range and coverage area/angle are required. To determine this information, Fig. 2.8 is considered.

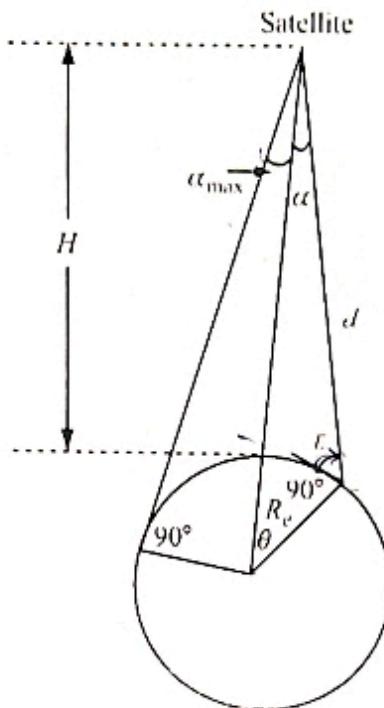


Figure 2.8 Coverage angle and slant range.

The earth coverage angle, $2\alpha_{\max}$, is the total angle subtended by the earth as seen from the satellite. This angle is important in design of a global coverage antenna and depends on satellite altitude. The communication coverage angle 2α is similarly defined, except that the minimum elevation angle E_{\min} of the earth station antenna must be taken into account. For an elevation angle E of the earth station antenna, the communication coverage angle 2α is given as

$$\frac{\sin \alpha}{R_e} = \frac{\sin(90^\circ + E)}{R_e + H} = \frac{\cos E}{R_e + H} \quad (2.22)$$

where R_e is the radius of the earth, assuming the earth to be spherical, H is the altitude of the satellite, which is a function of time for non-geostationary satellites. For geostationary satellite, $H = 35,786$ km. Thus,

$$2\alpha = 2 \sin^{-1} \left[\frac{R_e}{R_e + H} \cos E \right] \quad (2.23)$$

and the earth coverage angle can be determined when $E = 0^\circ$

$$2\alpha_{\max} = 2 \sin^{-1} \left[\frac{R_e}{R_e + H} \right] \quad (2.24)$$

For a geostationary satellite orbit where $R_e = 6378$ km, the earth coverage angle is $2\alpha_{max} = 17.4^\circ$, the central angle, θ , which is the angular radius of the satellite footprint,

$$\theta = 180^\circ - (90^\circ + E + \alpha) = 90^\circ - E - \alpha \quad (2.25)$$

for geostationary orbit. For global coverage, when $2\alpha_{max} = 17.4^\circ$, $\theta = 81.3^\circ$, if a minimum elevation angle of 5° is required for the above, these northern and southern latitudes of 76.3° will not be covered by the footprint of the satellite.

Besides the coverage angle, it is important to know the slant range from the earth station to the satellite, because this range determines the satellite roundtrip delay of the earth station. The slant range, d can be determined as

$$d^2 = (R_e + H)^2 + R_e^2 - 2 R_e (R_e + H) \sin \left[E + \sin^{-1} \left(\frac{R_e}{R_e + H} \cos E \right) \right] \quad (2.26)$$

for a geostationary orbit and a minimum elevation angle of 5° , the maximum slant range is $d = 41,127$ km, yielding a satellite roundtrip delay of $2d/c = 0.274$ s, where $c = 2.997925 \times 10^8$ km/s which is the speed of light.

2.7 ECLIPSE

A geostationary satellite utilizes solar energy to generate the required DC power to operate all the subsystems of the spacecraft. Solar energy is not available for a geostationary satellite when eclipse occurs. This occurs when the earth comes in between the sun and satellite in line and blocks the solar energy from reaching the solar panels of the satellite. This is a periodic feature and estimation of maximum duration of the eclipse period is very much required so as to determine the maximum capacity of the standby battery, which supplies the energy required for the subsystems during eclipse period.

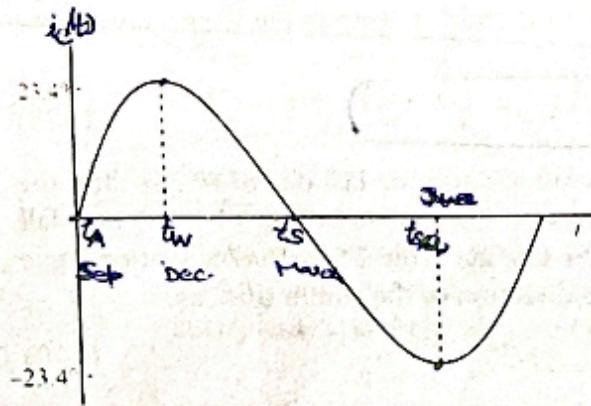


Figure 2.9(a) Sinusoidal variation of the earth's inclination angle.

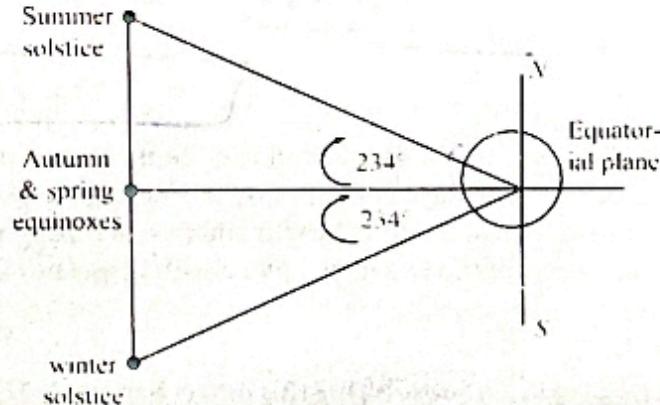


Figure 2.9(b) Apparent movement of the sun.

The earth's equatorial plane is inclined at an angle $i_e(t)$ with respect to the sun. The annual sinusoidal variation is given in degrees by

$$i_e(t) = 23.4 \sin(2\pi t/T) \quad (2.27)$$

where the annual period $T = 365$ days and maximum inclination is $i_{e,\max} = 23.4^\circ$. The time t_A and t_S when the inclination angle is zero are called the *autumn equinox* and *spring equinox* that occur on September 21 and March 21 respectively. The times t_W and t_S , when the inclination angle is at its maximum are called *winter solstice* and the *summer solstice* that occur on December 21 and June 21 respectively.

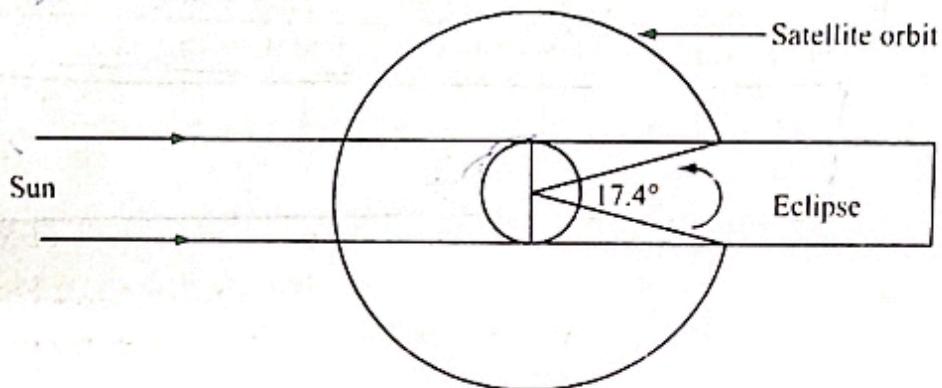


Figure 2.10 Eclipse when sun is at equinox.

Consider Fig. 2.10 to estimate the duration of eclipse. The diameter of the sun is neglected in these calculations (the sun is considered to be at infinity with respect to the earth and the rays of the sun are parallel) and hence the earth shadow is considered to be a cylinder of constant diameter. The maximum shadow angle occurs at the equinoxes and is given by

$$\varphi_{\max} = 180^\circ - 2 \cos^{-1}(R_e/a) = 17.4^\circ \quad (2.28)$$

Because the geostationary satellite period is 24 h, this maximum daily eclipse duration is

$$T_{\max} = 17.4^\circ \times 24 / 360^\circ \approx 1.16 \text{ h} \quad (2.29)$$

The first day of the eclipse before an equinox and the last day of eclipse after the equinox corresponds to the relative position of the sun such that the sun's rays fall tangent to the earth and passes through the satellite orbit. Thus, the inclination angle of the equatorial plane with respect to the direction of the sun in this case is

$$i_e = \frac{1}{2} \varphi_{\max} = 8.7^\circ \quad (2.30)$$

Substituting this into equation (2.27), it yields the time from the first day of eclipse to the equinox and also the time from the equinox to the last day of the eclipse.

$$t = (365/2\pi) \sin^{-1}(8.7/23.4) = 22.13 \text{ days} \quad (2.31)$$

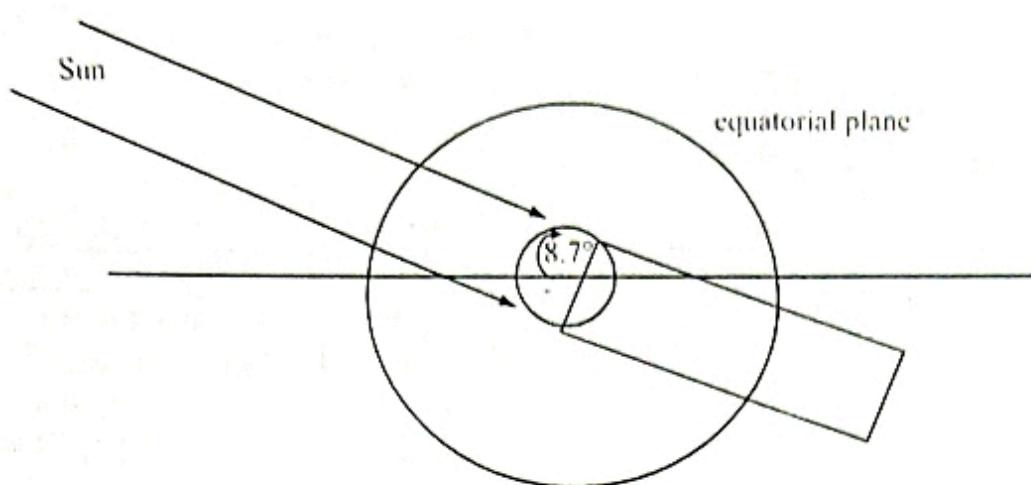


Figure 2.11 Earth inclination at first day of eclipse before equinox.

So, in total the satellite will be in eclipse for a period of 44.26 days around each equinox of the year. The period of equinox starts from zero per day from the starting day of the eclipse period i.e., 22.13 days before the equinox, gradually increases to 1.16 h per day on the day of equinox and then gradually reduces and becomes zero at the end of the eclipse period after 22.13 days after equinox. This process of eclipse occurs twice in a year.

2.8 ORBITAL PERTURBATIONS

Using the Kepler's laws, the equations for various parameters of orbit have been derived in the previous sections, i.e. it is assumed that the satellite orbit is ellipse. But this is not true ellipse due to various forces on the satellite tending to deviate from the perfect locus of ellipse. The factors, which influence this, are asymmetry of earth's gravitational field, the gravitational fields of sun and moon, solar radiation pressure, atmospheric drag, etc. If these interfering forces were left unchecked, the subsatellite point of the geostationary satellite would tend to move with time.

The approach to check and correct these effects of perturbations is first to derive an osculating orbit for some instant of time with orbital elements. The perturbations are assumed to cause the orbital elements to vary with time and the orbit and satellite location at any instant are taken from the osculating orbit calculated with orbital elements corresponding to that time. To visualize the process, assume the osculating orbital elements at time t_0 or a_0, e_0 , etc. Then, assume that the orbital elements vary linearly with time at a constant rates given by $da/dt, de/dt$, etc. The satellite's position at any time t_1 is then calculated from a Keplerian orbit with elements as

$$a_0 + (t_1 - t_0) da/dt, \quad e_0 + (t_1 - t_0) de/dt$$

As the perturbed orbit is not an ellipse, some care must be taken in defining an orbital period. Since the satellite does not return to the same point in space once per

revolution, the quantity most frequently specified is the *anomalistic period* defined as the time lapse between two successive perigee passages.

Effects of the earth's oblateness

Since the earth is not a perfect sphere with a symmetric distribution of mass, its gravitational potential does not have a simple $(1/r)$ dependence assumed by the equations of previous sections. The earth's gravitational potential is represented more accurately by an expression in Legendre polynomials J_n in ascending powers of earth's radius/orbital radius. The effect of the dominant J_1 coefficient term is to cause an unconstrained geosynchronous satellite to drift towards and circulate around the nearer of two stable points. These correspond to subsatellite longitudes of 105°W and 75°E, locations called *graveyards* because they collect old satellites whose station-keeping fuel is exhausted.

Effects of sun and moon

Gravitational attractions by the sun and moon cause the orbital inclination of a geostationary satellite to change with time. If not countered by north-south station-keeping, these forces would increase the orbital inclination from an initial 0° to 14.67° 26.6 years later. Since, no satellite has such a long lifetime, the problem is not acute.

2.9 ORBITAL EFFECTS IN COMMUNICATIONS SYSTEM PERFORMANCE

Doppler shift

Doppler shift in frequency due to relative motion between the satellite and a point on the earth surface is the prime concern in the low orbit satellites and is to be taken care for establishment of perfect communication link. Doppler can be given as

$$\frac{f_R - f_I}{f_I} = \frac{\Delta f}{f_I} = \frac{V_T}{v_p} \quad (2.32)$$

f_R , f_I are transmitted and received frequencies respectively. V_T is the component of transmitter velocity directed towards the receiver and v_p is the phase velocity of light.

The Doppler is not available in the downlink, but it affects the uplink if unchecked. The Doppler is not available with geostationary satellite since there is no relative motion between the earth station and the satellite.

Range variations

With the best station-keeping systems available, the position of a geostationary satellite with respect to the earth exhibits cyclic variations daily. The resulting range variations has negligible effect on the power equations, an effect on the roundtrip delay. This adds unacceptably large guard times in the TDMA systems. So, the TDMA system continuously monitors the range and adjusts the burst timing accordingly.

Eclipse

A satellite is said to be in *eclipse* when the earth blocks the solar energy to solar panels of the satellite when the three come in line. As discussed in the previous sections, eclipse occurs twice in a year around equinox. Figure 2.12 shows the eclipse time per day during the period of eclipse.

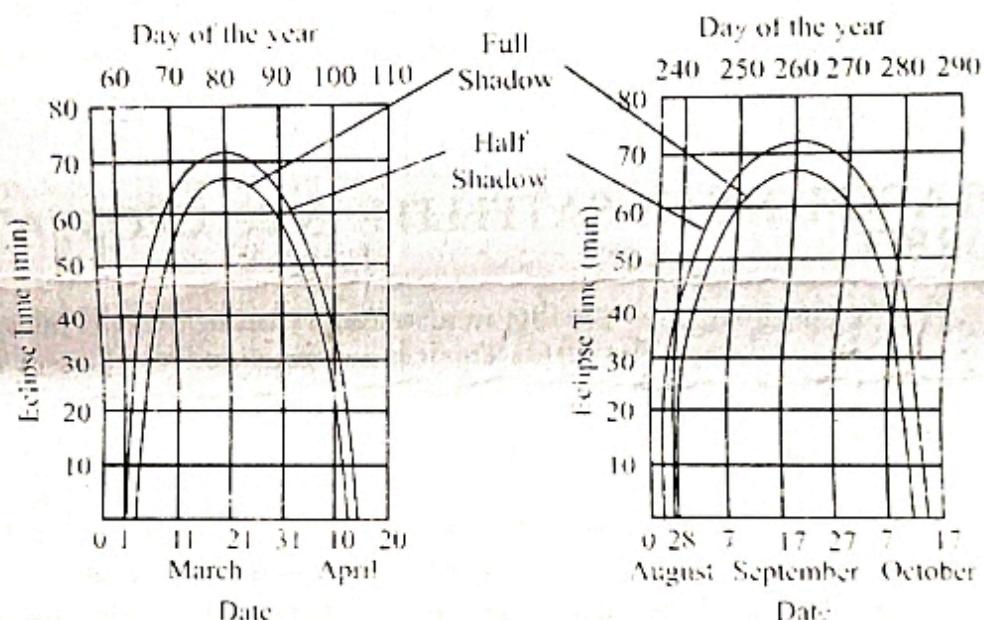


Figure 2.12 Dates and durations of eclipses.

The solar eclipse caused by moon to the geostationary satellite occurs when the moon moves to front of the sun. The eclipse occurs irregularly in time of duration and depth. In general, the eclipse may occur twice within a 24 hr period. Eclipse may range from a few minutes to over two hours within an average duration of about 40 minutes. Compared to earth-solar eclipse, the number of moon-solar eclipse range from zero to four with an average of two per year. It is worthwhile to note that if the moon-solar eclipse of long duration occurs just before or just after the earth-solar eclipse, the satellite has to face special problems in connection with battery recharging and spacecraft thermal reliability. In order to cope with the solar battery problems during eclipses an energy reserve is provided with the satellite.

During full eclipse a satellite receives no power from sun and it must operate entirely from batteries. This can reduce the available power significantly as the spacecraft nears the end of its life, and it may necessitate shutting down some of the transponders during the eclipse period. Spacecraft designers must guard harmful transients as solar power fluctuates sharply at the beginning and end of an eclipse. There is a possibility of having the primary power failure and so, the probability that a primary power supply failing is much more during eclipse rather than any other operations like deployment.

Sun-transit outage

The overall receiver noise will rise significantly to effect the communications when the sun passes through the beam of an earth station antenna. This effect is predictable and can cause outage for as much as 10 min a day for several days and for about 0.02% an average year. The receiving earth station has to wait until the sun moves out of the main lobe of the antenna. This occurs during the daytime, where the traffic is at its peak and forces the operator to hire some other alternative channels for uninterrupted communication link.

2.10 PLACEMENT OF SATELLITE INTO GEOSTATIONARY ORBIT

The placement of a satellite in a stationary orbit involves many complex sequences and is shown in Fig. 2.13. This type of satellite launching is known as *Hohmann Transfer*. First, the launch vehicle places the satellite in an elliptical transfer orbit whose apogee distance is equal to the radius of the geostationary orbit (42,164.2 km). The perigee distance of the elliptical transfer orbit is approximately 6678.2 km, about 300 km above the earth's surface. The satellite spin is stabilized in the transfer orbit so that ground control can communicate with the telemetry system. When the orbit and attitude of the satellite have been determined exactly and when the satellite is at the apogee of the transfer orbit, the apogee kick motor is fired to circularize the orbit. This circular orbit, with a radius of 42,164.2 km is a geostationary orbit if the launch is carried at 0° latitude, the equator. If the satellite is launched from any other latitude, the orbit would be geosynchronous with inclination i greater than or equal to the latitude θ , when the injection at the perigee is horizontal.

The velocity at the perigee and apogee can be determined as follows:

At perigee, $r = 6678.2$ km, $a = 24,421.2$ km and the velocity $V_p = 10.15$ km/s is given by equation (2.10) and

At apogee, $r = 42,164.2$ km, $V_a = 1.61$ km/s.

Velocity in the synchronous orbit is $V_c = 3.07$ km/s with $r = a = 42,164.2$ km.

The incremental velocity required to circularize the orbit at the apogee of transfer orbit must be

$$\Delta V = V_c - V_a = 1.46 \text{ km/s} \quad (2.33)$$

- (A) GS Transfer orbit w/ apogee kick motor
- (B) GS " " " w/ slow orbit increment
- (C) Direct Insert to GSO (Geo-stationary Earth Orbit)

Launch Vehicle is responsible for launching the sat till LEO. When sat is ~~not~~ in LEO after 1 or 2 rotations, it will make its path elliptical from LEO to GTO (Geostationary Transfer orbit) Above GEO.

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① Initially from some point on earth the sat is launched to LEO. with the help of some launching vehicle.

④ After reaching LEO sat itself has built motor that will make this sat to move further.

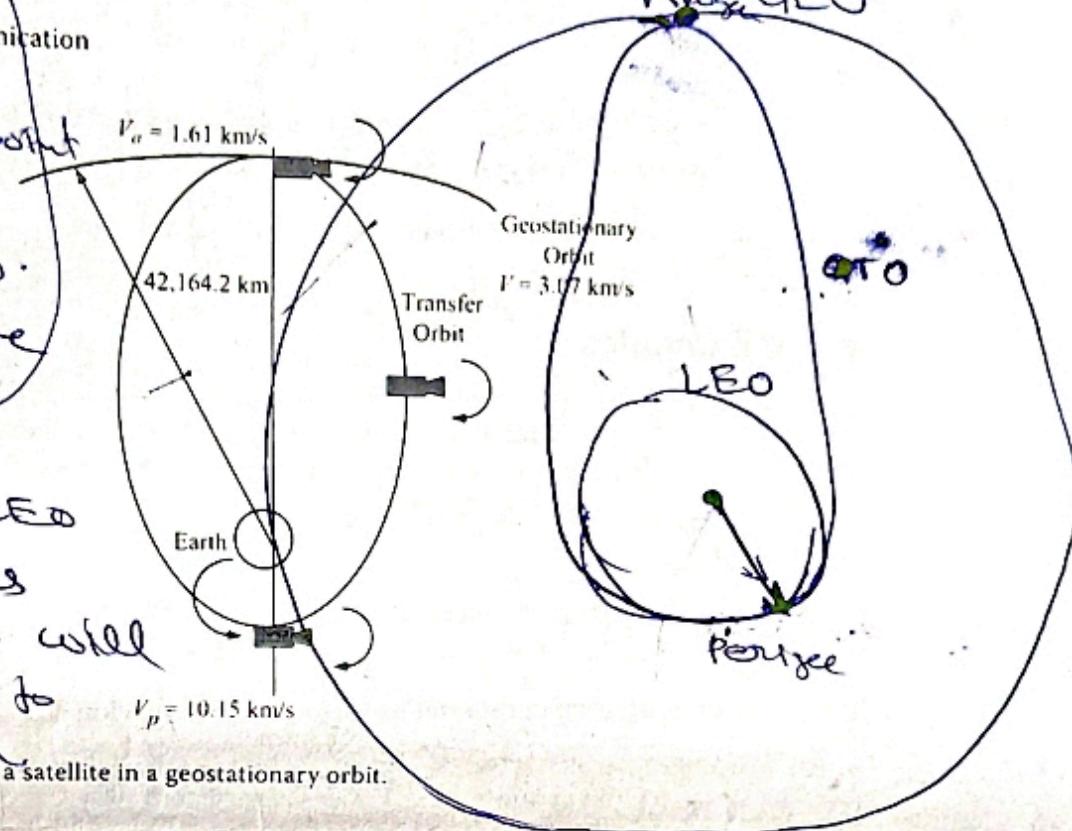


Figure 2.13 Placement of a satellite in a geostationary orbit.

⑤ There are 2 different motors. Since the plane of transfer orbit is formed by the position vector \mathbf{r} and the velocity vector \mathbf{V} of the satellite at a given instant in time, the inclination correction can be made at the ascending or descending node where the orbit intersects the equatorial plane at an incremental velocity vector $\Delta\mathbf{V}$ in such a way that the sum of the node velocity vector \mathbf{V}_n and the incremental velocity vector $\Delta\mathbf{V}$ is a vector \mathbf{V} in the equatorial plane. The inclination correction is as shown in Fig. 2.14 and is given by

$$\Delta\mathbf{V} = \sqrt{(V_n^2 + V_r^2 - 2 V_n V_r \cos i)} \quad (2.34)$$

1st one for circularization & the 2nd one for elliptical path

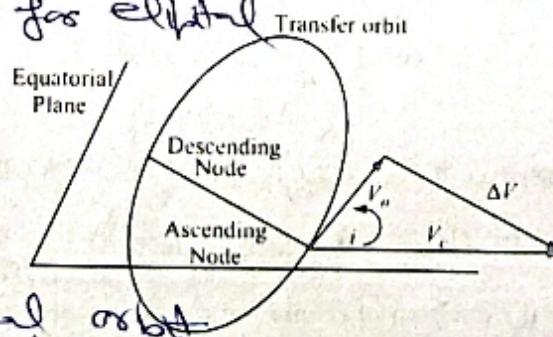


Figure 2.14 Simultaneous orbit circularization and inclination correction.

⑥ And at the apogee point of this elliptical orbit.

Due to an apogee kick motor it will do some boost and sat directly follow the GEO orbit.

- ⑦ Now incremental after GTO till it reaches GEO
- ⑧ It is responsible of launcher vehicle to put in GEO