2.2 LOOK ANGLE DETERMINATION

Navigation around the earth's oceans became more precise when the surface of the globe was divided up into a gridlike structure of orthogonal lines: latitude and longitude. Latitude is the angular distance, measured in degrees, north or south of the equator and longitude is the angular distance, measured in degrees, from a given reference longitudinal line. At the time that this grid reference became popular, there were two major seafaring nations vying for dominance: England and France. England drew its reference zero longitude through Greenwich, a town close to London, England, and France,

SIDEBAR

Frequencies and orbital slots for new satellites are registered with the International Frequency Registration Board (IFRB), part of the ITU located in Geneva. The initial application by an organization or company that wants to orbit a new satellite is made to the national body that controls the allocation and use of radio frequencies—the FCC in the United States, for example—which must first approve the application and then forward it to the IFRB. The first organization to file with the IFRB for a particular service is deemed to have protection from newcomers. Any other organization filing to carry the same service at, or close to, that orbital location (within 2°) must coordinate their use of the frequency bands with the first organization. The first user may cause interference into subsequent filer's satellite systems, since they were the first to be awarded the orbital slot and frequencies, but the later filers' satellites must not cause interference with the first user's system.

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not surprisingly, drew its reference longitude through Paris, France. Since the British Admiralty chose to give away their maps and the French decided to charge a fee for theirs, it was not surprising that the use of Greenwich as the zero reference longitude became dominant within a few years. [It was the start of .com market dominance through giveaways three centuries before E-commerce!] Geometry was a much older science than navigation and so 90° per quadrant on the map was an obvious selection to make. Thus, there are 360° of longitude (measured from 0° at the *Greenwich Meridian*, the line drawn from the North Pole to the South Pole through Greenwich, England) and ±90° of latitude, plus being measured north of the equator and minus south of the equator. Latitude 90° N (or +90°) is the North Pole and latitude 90° S (or −90°) is the South Pole. When GEO satellite systems are registered in Geneva, their (subsatellite) location over the equator is given in degrees east to avoid confusion. Thus, the INTELSAT primary location in the Indian Ocean is registered at 60° E and the primary location in the Atlantic Ocean is at 335.5° E (not 24.5° W). Earth stations that communicate with satellites are described in terms of their geographic latitude and longitude when developing the pointing coordinates that the earth station must use to track the apparent motion of the satellite.

The coordinates to which an earth station antenna must be pointed to communicate with a satellite are called the *look angles*. These are most commonly expressed as *azimuth* (Az) and *elevation* (El), although other pairs exist. For example, right ascension and declination are standard for radio astronomy antennas. Azimuth is measured eastward (clockwise) from geographic north to the projection of the satellite path on a (locally) horizontal plane at the earth station. Elevation is the angle measured upward from the local horizontal plane at the earth station to the satellite path. Figure 2.10 illustrates these look angles. In all look angle determinations, the precise location of the satellite is critical. A key location in many instances is the subsatellite point.

The Subsatellite Point

The subsatellite point is the location on the surface of the earth that lies directly between the satellite and the center of the earth. It is the *nadir* pointing direction from the satellite and, for a satellite in an equatorial orbit, it will always be located on the equator. Since geostationary satellites are in equatorial orbits and are designed to stay "stationary" over

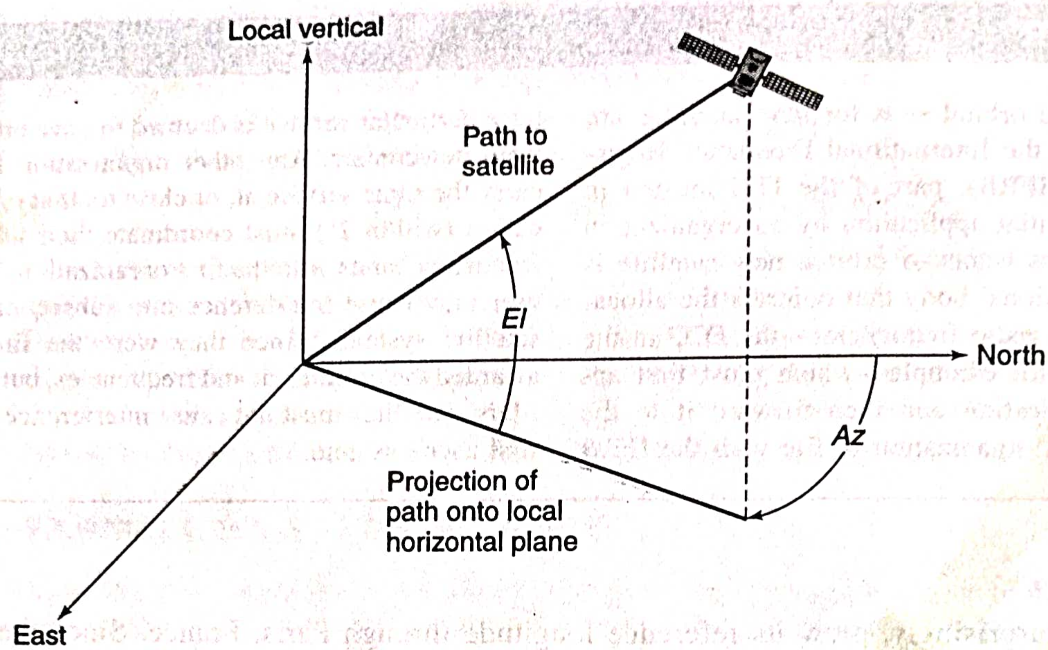

FIGURE 2.10 The definition of elevation (El) and azimuth (Az). The elevation angle is measured upward from the local horizontal at the earth station and the azimuth angle is measured from true north in an eastward direction to the projection of the satellite path onto the local horizontal plane.

the earth, it is usual to give their orbital location in terms of their subsatellite point. As noted in the example given earlier, the Intelsat primary satellite in the Atlantic Ocean Region (AOR) is at 335.5° E longitude. Operators of international geostationary satellite systems that have satellites in all three ocean regions (Atlantic, Indian, and Pacific) tend to use longitude east to describe the subsatellite points to avoid confusion between using both east and west longitude descriptors. For U.S. geostationary satellite operators, all of the satellites are located west of the Greenwich meridian and so it has become accepted practice for regional systems over the United States to describe their geostationary satellite locations in terms of degrees W.

To an observer of a satellite standing at the subsatellite point, the satellite will appear to be directly overhead, in the *zenith* direction from the observing location. The zenith and nadir paths are therefore in opposite directions along the same path (see Figure 2.11). Designers of satellite antennas reference the pointing direction of the satellite's antenna beams to the nadir direction. The communications coverage region on the earth from a satellite is defined by angles measured from nadir at the satellite to the edges of the coverage. Earth station antenna designers, however, do not reference their pointing direction to zenith. As noted earlier, they use the local horizontal plane at the earth station to define elevation angle and geographical compass points to define azimuth angle, thus giving the two look angles for the earth station antenna toward the satellite (Az, El).

Elevation Angle Calculation

Figure 2.12 shows the geometry of the elevation angle calculation. In Figure 2.12, $\boldsymbol{r}_s$ is the vector from the center of the earth to the satellite; $\boldsymbol{r}_e$ is the vector from the center of the earth to the earth station; and $\boldsymbol{d}$ is the vector from the earth station to the satellite. These three vectors lie in the same plane and form a triangle. The central angle γ measured between $\boldsymbol{r}_e$ and $\boldsymbol{r}_s$ is the angle between the earth station and the

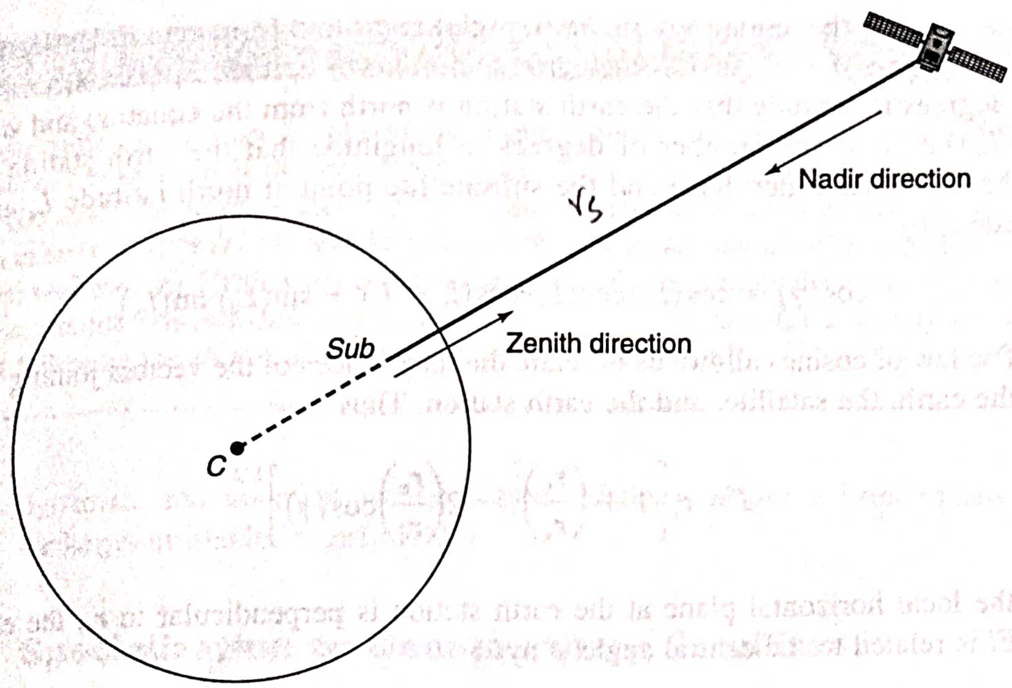

FIGURE 2.11 Zenith and nadir pointing directions. The line joining the satellite and the center of the earth, C, passes through the surface of the earth at point *Sub*, the subsatellite point. The satellite is directly overhead at this point and so an observer at the subsatellite point would see the satellite at zenith (i.e., at an elevation angle of 90°). The pointing direction from the satellite to the subsatellite point is the nadir direction from the satellite. If the beam from the satellite antenna is to be pointed at a location on the earth that is not at the subsatellite point, the pointing direction is defined by the angle away from nadir. In general, two off-nadir angles are given: the number of degrees north (or south) from nadir; and the number of degrees east (or west) from nadir. East, west, north, and south directions are those defined by the geography of the earth.

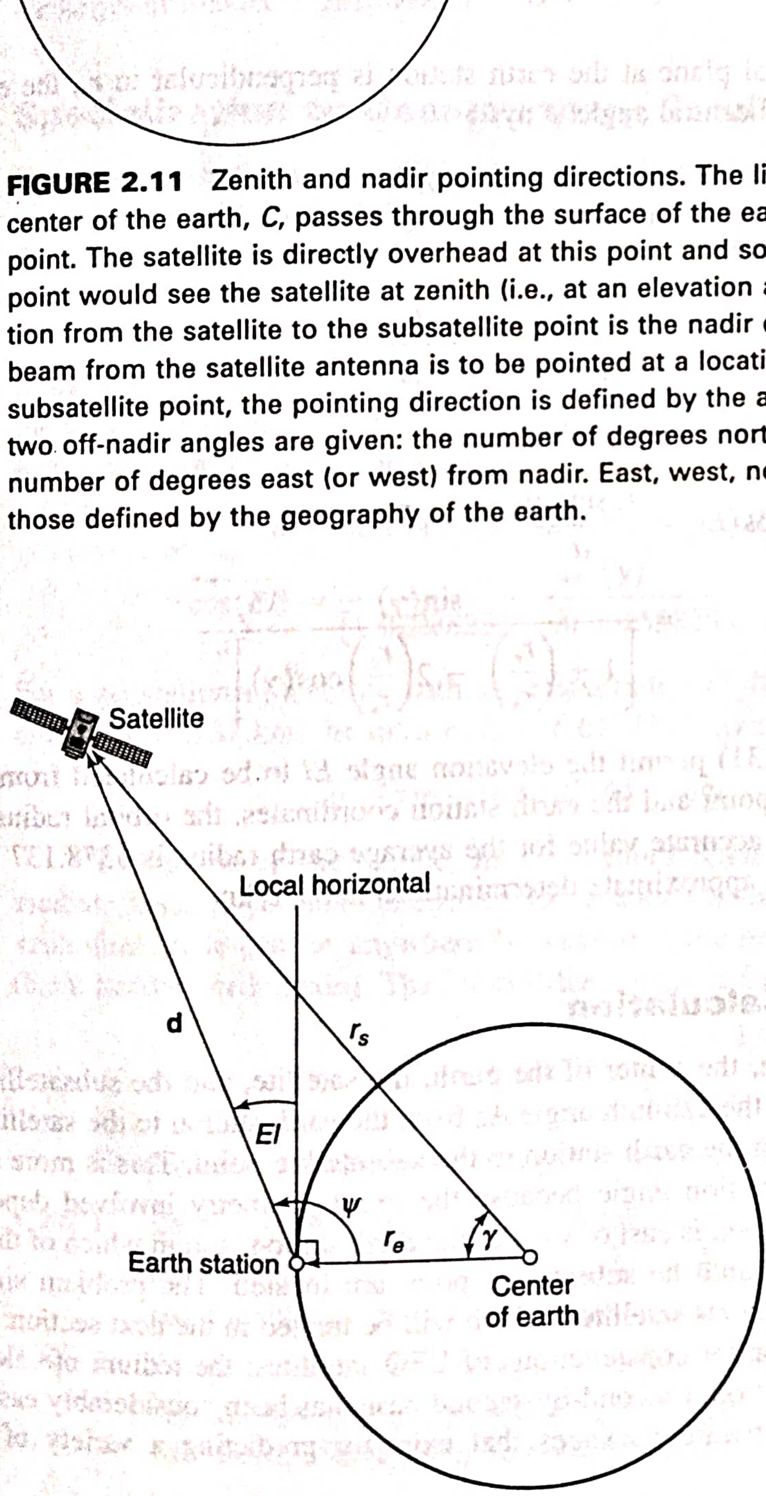

FIGURE 2.12 The geometry of elevation angle calculation. The plane of the paper is the plane defined by the center of the earth, the satellite, and the earth station. The central angle is γ. The elevation angle El is measured upward from the local horizontal at the earth station.

satellite, and ψ is the angle (within the triangle) measured from $\boldsymbol{r_e}$ to $\boldsymbol{d}$. Defined so that it is nonnegative, γ is related to the earth station north latitude L_e (i.e., L_e is the number of degrees in latitude that the earth station is north from the equator) and west longitude l_e (i.e., l_e is the number of degrees in longitude that the earth station is west from the Greenwich meridian) and the subsatellite point at north latitude L_s and west longitude l_s by

$$\cos(\gamma) = \cos(L_e)\cos(L_s)\cos(l_s - l_e) + \sin(L_e)\sin(L_s) \tag{2.31}$$

The law of cosines allows us to relate the magnitudes of the vectors joining the center of the earth, the satellite, and the earth station. Thus

$$d = r_s\left[1 + \left(\frac{r_e}{r_s}\right)^2 - 2\left(\frac{r_e}{r_s}\right)\cos(\gamma)\right]^{1/2} \tag{2.32}$$

Since the local horizontal plane at the earth station is perpendicular to $\boldsymbol{r_e}$, the elevation angle El is related to the central angle ψ by

$$El = \psi - 90° \tag{2.33}$$

By the law of sines we have

$$\frac{r_s}{\sin(\psi)} = \frac{d}{\sin(\gamma)} \tag{2.34}$$

Combining the last three equations yields

$$\begin{aligned}\cos(El) &= \frac{r_s \sin(\gamma)}{d} \\ &= \frac{\sin(\gamma)}{\left[1 + \left(\frac{r_e}{r_s}\right)^2 - 2\left(\frac{r_e}{r_s}\right)\cos(\gamma)\right]^{1/2}}\end{aligned} \tag{2.35}$$

Equations (2.35) and (2.31) permit the elevation angle El to be calculated from knowledge of the subsatellite point and the earth station coordinates, the orbital radius r_s, and the earth's radius r_e. An accurate value for the average earth radius is 6378.137 km[1] but a common value used in approximate determinations is 6370 km.

Azimuth Angle Calculation

Because the earth station, the center of the earth, the satellite, and the subsatellite point all lie in the same plane, the azimuth angle Az from the earth station to the satellite is the same as the azimuth from the earth station to the subsatellite point. This is more difficult to compute than the elevation angle because the exact geometry involved depends on whether the subsatellite point is east or west of the earth station, and in which of the hemispheres the earth station and the subsatellite point are located. The problem simplifies somewhat for geosynchronous satellites, which will be treated in the next section. For the general case, in particular for constellations of LEO satellites, the tedium of calculating the individual look angles on a second-by-second basis has been considerably eased by a range of commercial software packages that exist for predicting a variety of orbital

SIDEBAR

A popular suite of software employed by many launch service contractors is that developed by Analytical Graphics: the *Satellite Tool Kit*[3]. The core program in early 2001, STK 4.0, and the subsequent subseries, was used by Hughes to rescue *AsiaSat3* when that satellite was stranded in a highly elliptical orbit following the failure of an upper stage in the launch vehicle. Hughes used two *lunar* flybys to provide the necessary additional velocity to circularize the orbit at geostationary altitude. A number of organizations offer web sites that provide orbital plots in a three-dimensional graphical format with rapid updates for a variety of satellites (e.g., the NASA site[4]).

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dynamics and intercept solutions (see reference 13 for a brief review of 10 software packages available in early 2001).

Specialization to Geostationary Satellites

For most geostationary satellites, the subsatellite point is on the equator at longitude l_s, and the latitude L_s is 0. The geosynchronous radius r_s is 42,164.17 km[1]. Since L_s is zero, Eq. (2.31) simplifies to

$$\cos(\gamma) = \cos(L_e)\cos(l_s - l_e) \tag{2.36}$$

Substituting r_s = 42,164.17 km and r_e = 6,378.137 km in Eqs. (2.32) and (2.35) gives the following expressions for the distance d from the earth station to the satellite and the elevation angle El at the earth station

$$d = 42{,}164.17[1.02288235 - 0.30253825\cos(\gamma)]^{1/2}\ \text{km} \tag{2.37}$$

$$\cos(El) = \frac{\sin(\gamma)}{[1.02288235 - 0.30253825\cos(\gamma)]^{1/2}} \tag{2.38}$$

For a geostationary satellite with an orbital radius of 42,164.17 km and a mean earth radius of 6378.137 km, the ratio r_s/r_e = 6.6107345 giving

$$El = \tan^{-1}[(6.6107345 - \cos\gamma)/\sin\gamma] - \gamma \tag{2.39}$$

To find the azimuth angle, an intermediate angle α must first be found. The intermediate angle α permits the correct 90° quadrant to be found for the azimuth since the azimuthal angle can lie anywhere between 0° (true north) and clockwise through 360° (back to true north again). The intermediate angle is found from

$$\alpha = \tan^{-1}\left[\frac{\tan|(l_s - l_e)|}{\sin(L_e)}\right] \tag{2.40}$$

Having found the intermediate angle α, the azimuth look angle Az can be found from:

Case 1: Earth station in the Northern Hemisphere with

(a) Satellite to the SE of the earth station: $Az = 180° - \alpha$ (2.41a)

(b) Satellite to the SW of the earth station: $Az = 180° + \alpha$ (2.41b)

Case 2: Earth station in the Southern Hemisphere with

(c) Satellite to the NE of the earth station: $Az = \alpha$ (2.41c)

(d) Satellite to the NW of the earth station: $Az = 360° - \alpha$ (2.41d)

2.3 ORBITAL PERTURBATIONS

The orbital equations developed in Section 2.1 modeled the earth and the satellite as point masses influenced only by gravitational attraction. Under these ideal conditions, a "Keplerian" orbit results, which is an ellipse whose properties are constant with time. In practice, the satellite and the earth respond to many other influences including asymmetry of the earth's gravitational field, the gravitational fields of the sun and the moon, and solar radiation pressure. For low earth orbit satellites, atmospheric drag can also be important. All of these interfering forces cause the true orbit to be different from a simple Keplerian ellipse; if unchecked, they would cause the subsatellite point of a nominally geosynchronous satellite to move with time.

Historically, much attention has been given to techniques for incorporating additional perturbing forces into orbit descriptions. The approach normally adopted for communications satellites is first to derive an *osculating orbit* for some instant in time (the Keplerian orbit the spacecraft would follow if all perturbing forces were removed at that time) with orbital elements (a, e, t_p, Ω, i, ω). 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 that the osculating orbital elements at time t_0 are (a_0, e_0, t_p, Ω_0, i_0, ω_0). Then assume that the orbital elements vary linearly with time at 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

$$a_0 + \frac{da}{dt}(t_1 - t_0), e_0 + \frac{de}{dt}(t_1 - t_0), \text{ etc.}$$

This approach is particularly useful in practice because it permits the use of either theoretically calculated derivatives or empirical values based on satellite observations.

As the perturbed orbit is not an ellipse, some care must be taken in defining the orbital period. Since the satellite does not return to the same point in space once per revolution, the quantity most frequently specified is the so-called *anomalistic period*: the elapsed time between successive perigee passages. In addition to the orbit not being a perfect Keplerian ellipse, there will be other influences that will cause the apparent position of a geostationary satellite to change with time. These can be viewed as those causing mainly longitudinal changes and those that principally affect the orbital inclination.

Longitudinal Changes: Effects of the Earth's Oblateness

The earth is neither a perfect sphere nor a perfect ellipse; it can be better described as a triaxial ellipsoid[1]. The earth is flattened at the poles; the equatorial diameter is about 20 km more than the average polar diameter. The equatorial radius is not constant, although the noncircularity is small: the radius does not vary by more than about 100 m around the equator[1]. In addition to these nonregular features of the earth, there are regions where the average density of the earth appears to be higher. These are referred to as regions of mass concentration or *Mascons*. The nonsphericity of the earth, the noncircularity of the equatorial radius, and the Mascons lead to a nonuniform gravitational field around the earth. The force on an orbiting satellite will therefore vary with position.

For a low earth orbit satellite, the rapid change in position of the satellite with respect to the earth's surface will lead to an averaging out of the perturbing forces in line with the orbital velocity vector. The same is not true for a geostationary (or geosynchronous) satellite. A geostationary satellite is weightless when in orbit. The smallest force on the satellite will cause it to accelerate and then drift away from its nominal location. The satellite is required to maintain a constant longitudinal position over the equator, but there will generally be an additional force toward the nearest equatorial bulge in either an eastward or a westward direction along the orbit plane. Since this will rarely be in line with the main gravitational force toward the earth's center, there will be a resultant component of force acting in the same direction as the satellite's velocity vector or against it, depending on the precise position of the satellite in the GEO orbit. This will lead to a resultant acceleration or deceleration component that varies with longitudinal location of the satellite.

Due to the position of the Mascons and equatorial bulges, there are four equilibrium points in the geostationary orbit: two of them stable and two unstable. The stable points are analogous to the bottom of a valley, and the unstable points to the top of a hill. If a ball is perched on top of a hill, a small push will cause it to roll down the slope into a valley, where it will roll backwards and forwards until it gradually comes to a final stop at the lowest point. The satellite at an unstable orbital location is at the top of a gravity hill. Given a small force, it will drift down the gravity slope into the gravity well (valley) and finally stay there, at the stable position. The stable points are at about 75° E and 252° E and the unstable points are at around 162° E and 348° E[1]. If a satellite is perturbed slightly from one of the stable points, it will tend to drift back to the stable point without any thruster firings required. A satellite that is perturbed slightly from one of the unstable points will immediately begin to accelerate its drift toward the nearer stable point and, once it reaches this point, it will oscillate in longitudinal position about this point until (centuries later) it stabilizes at that point. These stable points are sometimes called the graveyard geosynchronous orbit locations (not to be confused with the graveyard orbit for a geosynchronous satellite, which is the orbit to which the satellite is raised once the satellite ceases to be useful). Note that, due to the nonsphericity of the earth, etc., the stable points are neither exactly 180° apart, nor are the stable and unstable points precisely 90° apart.

Inclination Changes: Effects of the Sun and the Moon

The plane of the earth's orbit around the sun—the *ecliptic*—is at an inclination of 7.3° to the equatorial plane of the sun (Figure 2.14). The earth is titled about 23° away from the normal to the ecliptic. The moon circles the earth with an inclination of around 5° to the equatorial plane of the earth. Due to the fact that the various planes—the sun's equator, the ecliptic, the earth's equator (a plane normal to the earth's rotational axis), and the moon's orbital plane around the earth—are all different, a satellite in orbit around the earth will be subjected to a variety of out-of-plane forces. That is, there will generally be a net acceleration force that is not in the plane of the satellite's orbit, and this will tend to try to change the inclination of the satellite's orbit from its initial inclination. Under these conditions, the orbit will precess and its inclination will change.

The mass of the sun is significantly larger than that of the moon but the moon is considerably closer to the earth than the sun (see Table 2.2). For this reason, the acceleration force induced by the moon on a geostationary satellite is about twice as large as that of the sun. The net effect of the acceleration forces induced by the moon and the sun on a

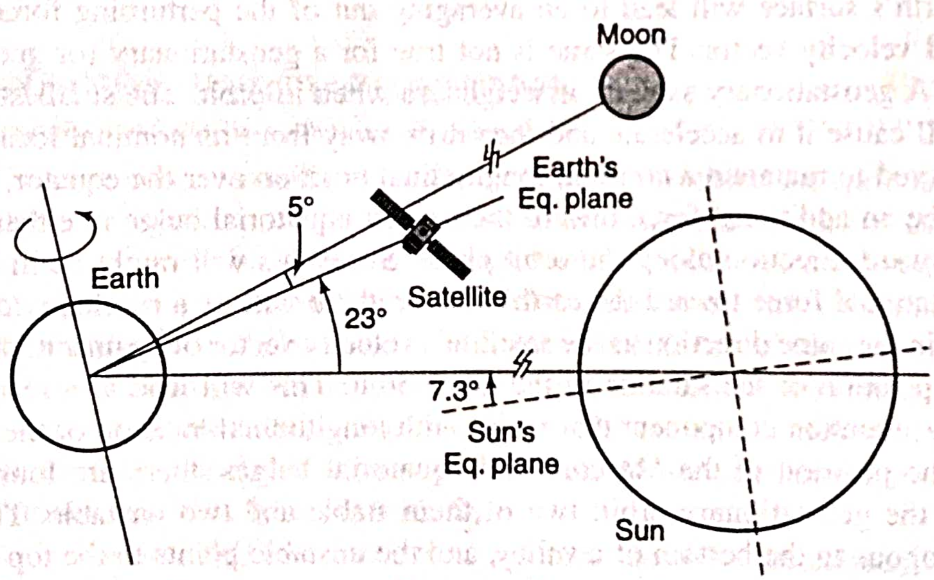

FIGURE 2.14 Relationship between the orbital planes of the sun, moon, and earth. The plane of the earth's orbit around the sun is the *ecliptic.* The geostationary orbit plane (the earth's equatorial plane) is about 23° out of the ecliptic, and leads to maximum out-of-geostationary-orbit-plane forces at the solstice periods (approximately June 21 and December 21). The orbit of the moon is inclined about 5° to the earth's equatorial plane. The moon revolves around the earth in 27.3 days, the earth (and the geostationary satellite) rotates once about 24 h, and the earth revolves around the sun every 365.25 days. In addition, the sun—which has a greater girth at the equator than at the poles—has its equator inclined about 7.3° to the ecliptic. All of these various angular differences and orbital periods lead to conditions where all of the out-of-plane gravitational forces are in one direction with respect to the equatorial (geostationary orbital) plane at a given time as well as to conditions where the various gravitational out-of-plane forces partially cancel each other out. The precessional forces that cause the inclination of the geostationary satellite's orbit to move away from the equatorial plane therefore vary with time.

TABLE 2.2 Comparative Data for the Sun, Moon, and Earth

	Mean radius	Mass	Mean orbit radius	Spin period
Sun	696,000 km	333,432 units	30,000 light years	25.04 earth days
Moon	3,476 km	0.012 units	384,500 km	27.3 earth days
Earth	6,378.14 km	1.0 units	149,597,870 km	1 earth day

The orbit radius refers to the center of the home galaxy (Milky Way) for the sun, center of earth for the moon, and center of the sun for the earth, respectively.

geostationary satellite is to change the plane of the orbit at an initial average rate of change of 0.85°/year from the equatorial plane[1].

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 the plane of the satellite's orbit will be less than average. Examples of maximum years are 1988 and 2006 (0.94°/year) and examples of minimum years are 1997 and 2015 (0.75°/year)[1]. These rates of change are neither constant with time nor with inclination. They are at a maximum when the inclination is zero and they are zero when the inclination is 14.67°. From an initial zero inclination, the plane of the geostationary orbit will change to a maximum inclination of 14.67° over 26.6 years. The acceleration forces will then change direction at this maximum inclination and the orbit inclination will move back to zero in another 26.6 years and out to $-14.67°$ over a further 26.6 years, and so on.

In some cases, to increase the orbital maneuver lifetime of a satellite for a given fuel load, mission planners deliberately place a satellite planned for geostationary orbit into an initial orbit with an inclination that is substantially larger than the nominal 0.05° for a geostationary satellite. The launch is specifically timed, however, so as to set up the necessary precessional forces that will automatically reduce the inclination "error" to close to zero over the required period without the use of any thruster firings on the spacecraft. This will increase the maneuvering lifetime of the satellite at the expense of requiring greater tracking by the larger earth terminals accessing the satellite for the first year or so of the satellite's operational life.

Under normal operations, ground controllers command spacecraft maneuvers to correct for both the in-plane changes (longitudinal drifts) and out-of-plane changes (inclination changes) of a satellite so that it remains in the correct orbit. For a geostationary satellite, this means that the inclination, ellipticity, and longitudinal position are controlled so that the satellite appears to stay within a "box" in the sky that is bounded by $\pm 0.05°$ in latitude and longitude over the subsatellite point. Some maneuvers are designed to correct for both inclination and longitude drifts simultaneously in the one burn of the maneuvering rockets on the satellite. In others, the two maneuvers are kept separate: one burn will correct for ellipticity and longitude drift; another will correct for inclination changes. The latter situation of separated maneuvers is becoming more common for two reasons. The first is due to the much larger velocity increment needed to change the plane of an orbit (the so-called north–south maneuver) as compared with the longitude/ellipticity of an orbit (the so-called east–west maneuver). The difference in energy requirement is about 10:1. By alternately correcting for inclination changes and in-plane changes, the attitude of the satellite can be held constant and different sets of thrusters exercised for the required maneuver.

The second reason is the increasing use of two completely different types of thrusters to control N–S maneuvers on the one hand and E–W maneuvers on the other. In the mid-1990s, one of the heaviest items that was carried into orbit on a large satellite was the fuel to raise and control the orbit. About 90% of this fuel load, once on orbit, was to control the inclination of the satellite. Newer rocket motors, particularly arc jets and ion thrusters, offer increased efficiency with lighter mass. In general, these low thrust, high efficiency rocket motors are used for N–S maneuvers leaving the liquid propellant thrusters, with their inherently higher thrust (but lower efficiency) for orbit raising and in-plane changes. In order to be able to calculate the required orbit maneuver for a given satellite, the controllers must have an accurate knowledge of the satellite's orbit. Orbit determination is a major aspect of satellite control.

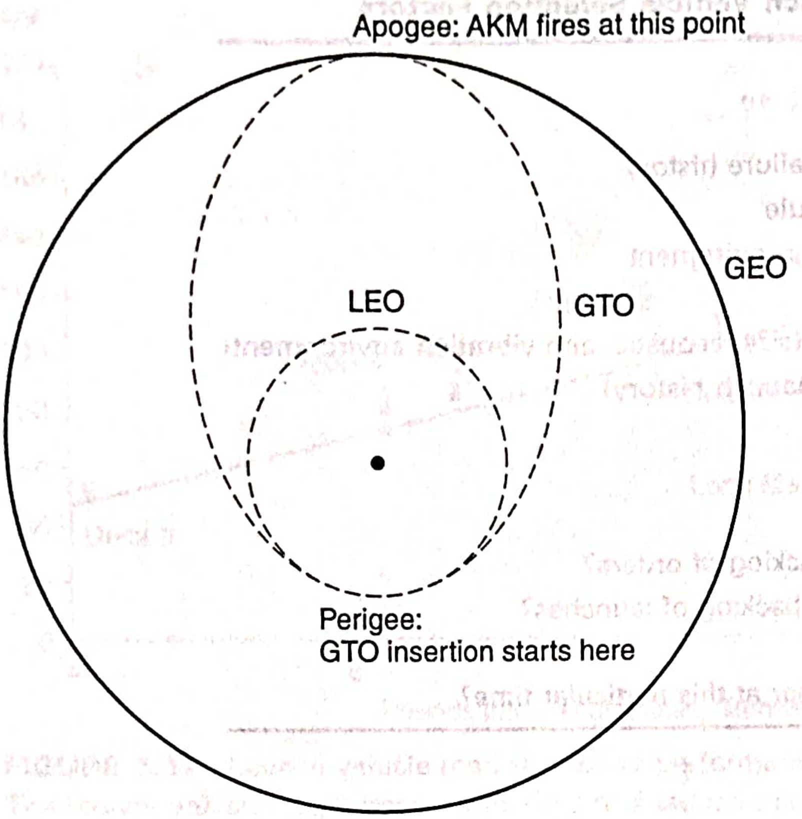

FIGURE 2.19 Illustration of the GTO/AKM approach to geostationary orbit (not to scale). The combined spacecraft and final rocket stage are placed into low earth orbit (LEO) around the earth. After careful orbit determination measurements, the final stage is ignited in LEO and the spacecraft inserted into a transfer orbit that lies between the LEO and the geostationary orbit altitude: the so-called geostationary transfer orbit or GTO. Again, after more careful orbit determination, the apogee kick motor (AKM) is fired on the satellite and the orbit is both circularized at geostationary altitude and the inclination reduced to close to zero. The satellite is then in GEO.

Some of the launch vehicles deliver the spacecraft directly to geostationary orbit (called a direct-insertion launch) while others inject the spacecraft into a geostationary transfer orbit (GTO). Spacecraft launched into GTO must carry additional rocket motors and/or propellant to enable the vehicle to reach the geostationary orbit. There are three basic ways to achieve geostationary orbit.

Placing Satellites into Geostationary Orbit

Geostationary Transfer Orbit and AKM The initial approach to launching geostationary satellites was to place the spacecraft, with the final rocket stage still attached, into low earth orbit. After a couple of orbits, during which the orbital elements are measured, the final stage is reignited and the spacecraft is launched into a geostationary transfer orbit. The GTO has a perigee that is the original LEO orbit altitude and an apogee that is the GEO altitude. Figure 2.19 illustrates the process. The position of the apogee point is close to the orbital longitude that would be the in-orbit test location of the satellite prior to it being moved to its operational position. Again, after a few orbits in the GTO while the orbital elements are measured, a rocket motor (usually contained within the satellite itself) is ignited at apogee and the GTO is raised until it is a circular, geostationary orbit. Since the rocket motor fires at apogee, it is commonly referred to as the apogee kick motor

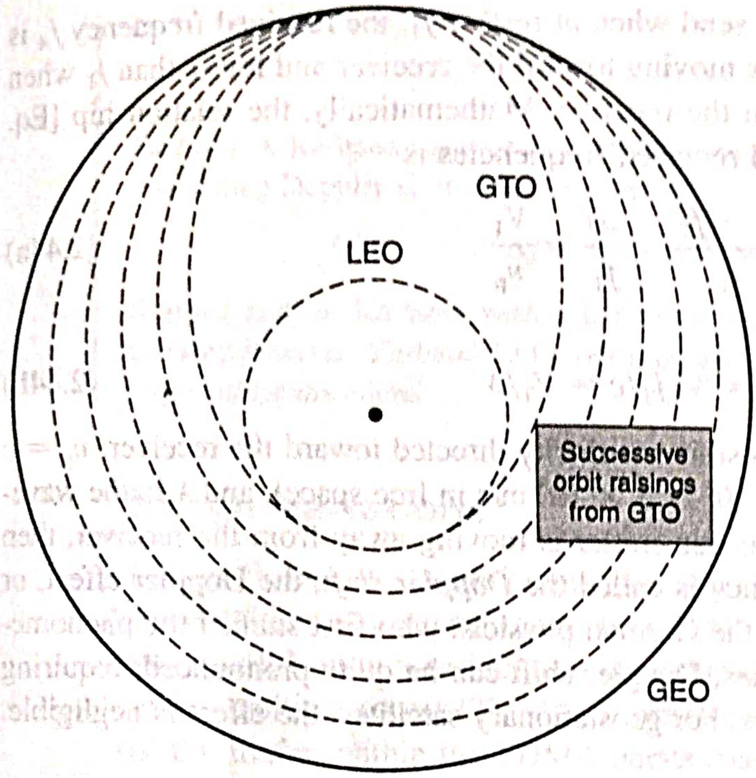

FIGURE 2.20 Illustration of slow orbit raising to geostationary orbit (not to scale). The combined spacecraft and final rocket stage are placed into low earth orbit (LEO) around the earth. As before (see Figure 2.19), the spacecraft is injected into GTO but, in this case, once the satellite is ejected from the final rocket stage, it deploys many of the elements that it will later use in GEO (solar panels, etc.) and stabilizes its attitude using thrusters and momentum wheels, rather than being spin-stabilized. The higher power thrusters are then used around the apogee to raise the perigee of the orbit until the orbit is circular at the GEO altitude. At the same time as the orbit is being raised, the thruster firings will be designed gradually to reduce the inclination to close to zero.

(AKM). The AKM is used both to circularize the orbit at GEO and to remove any inclination error so that the final orbit of the satellite is very close to geostationary.

Geostationary Transfer Orbit with Slow Orbit Raising In this procedure, rather than employ an apogee kick motor that imparts a vigorous acceleration over a few minutes, the spacecraft thrusters are used to raise the orbit from GTO to GEO over a number of burns. Since the spacecraft cannot be spin-stabilized during the GTO (so as not to infringe the Hughes patent), many of the satellite elements are deployed while in GTO, including the solar panels. The satellite has two power levels of thrusters: one for more powerful orbit raising maneuvers and one for on-orbit (low thrust) maneuvers. Since the thrusters take many hours of operation to achieve the geostationary orbit, the perigee of the orbit is gradually raised over successive thruster firings. The thruster firings occur symmetrically about the apogee although they could occur at the perigee as well. The burns are typically 60 to 80 min long on successive orbits and up to six orbits can be used. Figure 2.20 illustrates the process.

In the first two cases, AKM and slow orbit raising, the GTO may be a modified orbit with the apogee well above the required altitude for GEO. The excess energy of the orbit due to the higher-than-necessary altitude at apogee can be traded for energy required to raise the perigee. The net energy to circularize the orbit at GEO is therefore less and the satellite can retain more fuel for on-orbit operations.

Direct Insertion to GEO This is similar to the GTO technique but, in this case, the launch service provider contracts to place the satellite into GEO. The final stages of the rocket are used to place the satellite directly into GEO rather than the satellite using its own propulsion system to go from GTO to GEO.