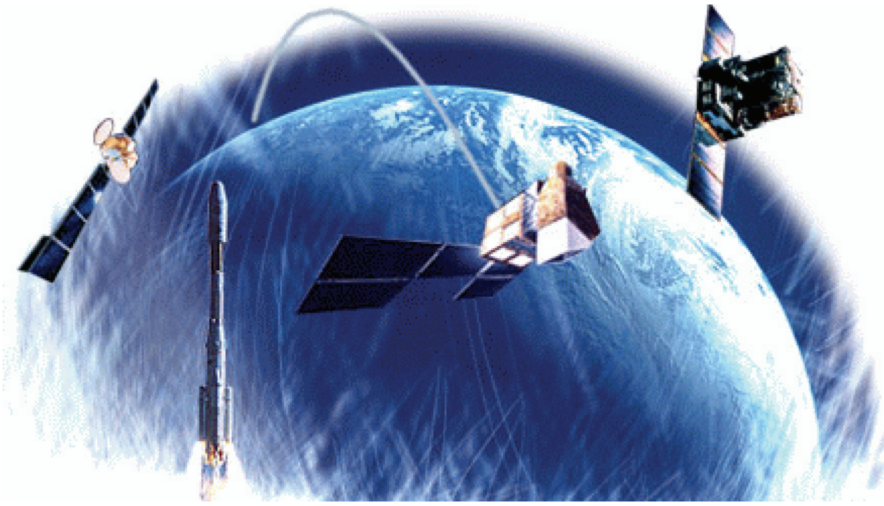
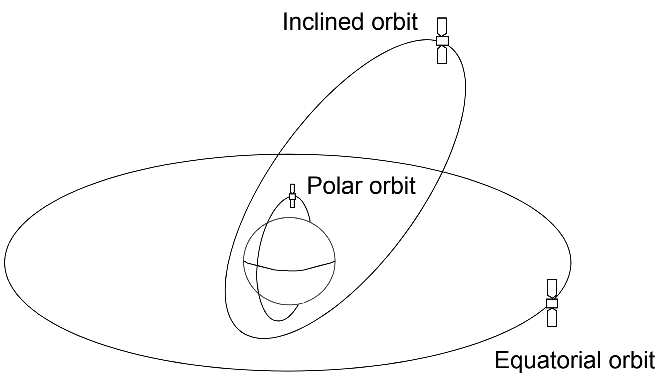


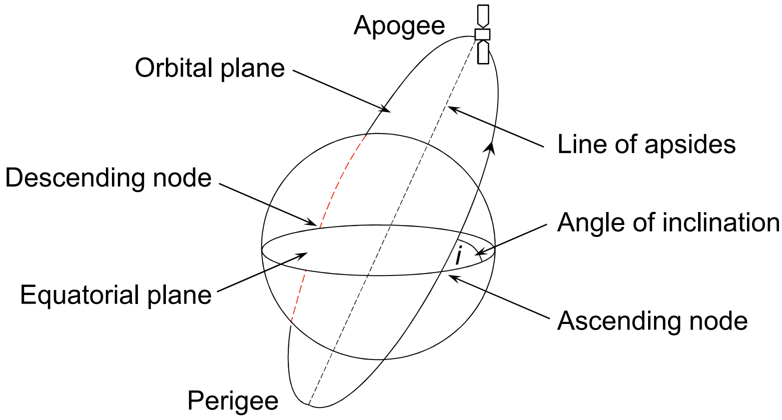
SATELLITE ORBITS

Classification of Orbits

- Orbits of a satellite are classified as *equatorial*, *polar*, or *inclined*, defined by the path on Earth traced out by the *sub-satellite point* during the orbit. The sub-satellite point is the point on the Earth's surface that lies on the line between the centre of the Earth and the satellite.



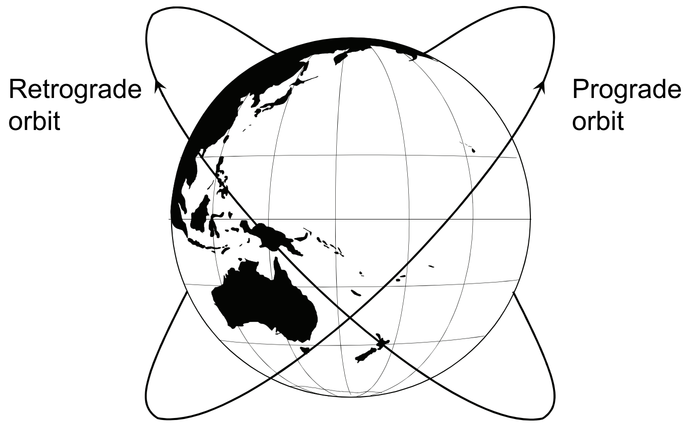
Orbital Parameters



Classification of Orbits

- The altitude selected for the satellite above the Earth directly determines both the *period* of the orbit (time to complete one cycle around the Earth) and the velocity that must be imparted to the vehicle to achieve that orbit.
- *Non-synchronous satellites* rotate around the Earth in a low-altitude elliptical or circular pattern. Orbits are classified as either *prograde* or *retrograde.*
  - If the satellite is orbiting in the same direction as Earth's rotation and at an angular velocity greater than that of Earth, the orbit is called a *prograde orbit.* A prograde orbit has an *angle of inclination* of less than 90°.
  - If the satellite is orbiting in the opposite direction as Earth's rotation or in the same direction but at an angular velocity less than that of Earth, the orbit is called a *retrograde orbit*. A retrograde orbit has an angle of inclination of greater than 90°.

Prograde and Retrograde Orbits



Geostationary Orbit

- *Geosynchronous* satellites have a period of rotation that is synchronised to that of the Earth or some multiple of it.
- The *geostationary* orbit is a unique geosynchronous one, located over the equator with a 0° inclination. The satellite in geostationary orbit has a height and velocity such that it appears stationary to Earth-bound observation.
- The height above the Earth's surface required for geostationary orbit is 35 786 km with a velocity of 3.073 km/s or 11 069 km/hr.

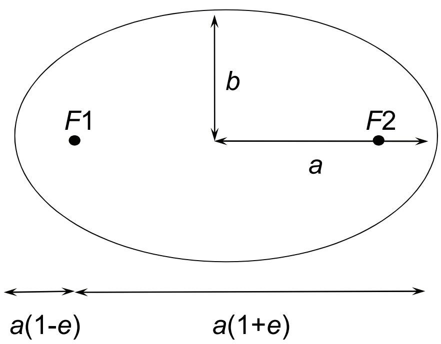
Kepler’s Laws

- Johann Kepler’s laws apply to the motion of satellites in elliptical orbits.
- Kepler developed these laws empirically, based on conclusions drawn from the extensive observations of Mars by Tycho Brahe (taken around the year 1600).
- They were originally defined in terms of the motion of the planets about the Sun, but apply equally to the motion of natural or artificial satellites about the Earth.
- The first two laws were published in 1609 in *Astronomia Nova*, and the third was announced in 1619 in *Harmonie Mundi*.

Kepler’s First Law

The satellite will follow an elliptical path around the Earth, which is located at one of the foci.

- The largest mass is called the *primary* body, or *barycentre*; the smaller mass is called the *secondary* body.

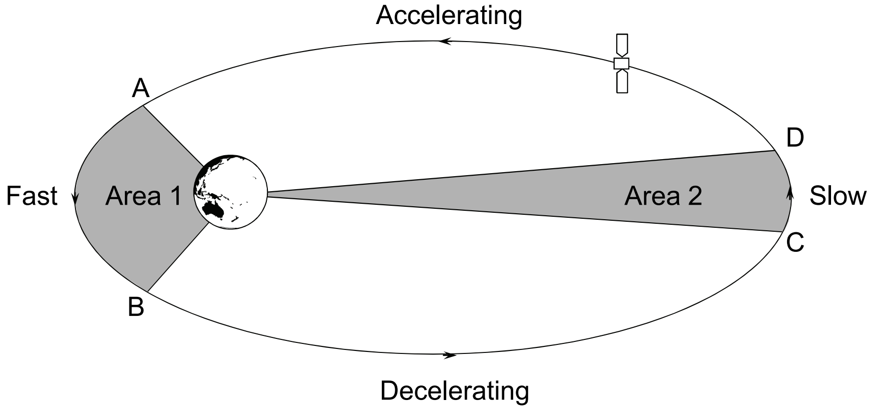


Eccentricity (e):

$$e = \frac{\sqrt{a^2 - b^2}}{a}$$

Kepler's Second Law

The line joining the satellite with the centre of the Earth sweeps out equal areas in equal times.



Kepler's Second Law

- The velocity of a satellite in elliptical orbit is:

$$v = \sqrt{Gm_e\left(\frac{2}{r} - \frac{1}{a}\right)} = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

- where r is the distance of the satellite above the centre of the Earth, a is the semi-major axis of the ellipse, the gravitational constant $G \approx 6.672\times10^{-11}$ Nm^2/kg^2, and the Earth's mass $m_e \approx 5.98\times10^{24}$ kg. The product G_{me} is known as *Kepler's constant*, μ, and has the value 3.986004418×10^5 km^3/s^2.

Kepler's Second Law

- Note that the velocity of the satellite is independent of the mass of the satellite. Therefore all satellites have the same acceleration at the same distance, regardless of their masses (provided that the satellite masses are negligible in comparison with the Earth). This is an extension of Galilleo's discovery that, in the absence of air resistance, all masses fall with the same acceleration at the surface of the Earth.
- Of course, when the orbit is circular, the satellite has uniform velocity because *r*=*a*, which is a constant.

$$v = \sqrt{Gm_e\left(\frac{2}{r} - \frac{1}{a}\right)} = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \quad \Longrightarrow \quad v = \sqrt{\frac{\mu}{r}}$$

Kepler's Third Law

The cube of the mean distance of the satellite from the Earth is proportional to the square of its period.

$$T^2 \propto a^3$$

- A satellite's orbit therefore increases as its height increases so that a satellite in low orbit might circle the Earth a number of times each day while a higher satellite might circle the Earth once or twice each day. It can be shown that the precise relationship is:

$$T = 2\pi\sqrt{\frac{a^3}{Gm_e}} = 2\pi\sqrt{\frac{a^3}{\mu}}$$

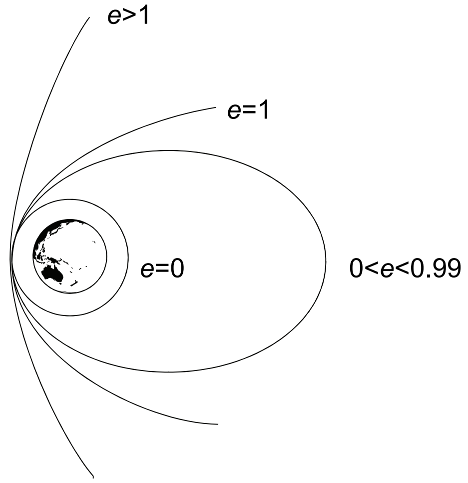
- This relationship applies for a perfectly spherical Earth with the satellite as the only other body involved. Later we discuss orbital variations from the ideal, caused by a range of factors.

Satellite Orbits

Types of orbit

- Hyperbolic
- Parabolic
- Elliptical

$$e = \frac{\sqrt{a^2 - b^2}}{a}$$



Choice of Orbit

- The utility of a communications satellite depends on its orbit, the selection of which depends on a careful balance of the following orbital parameters:
  - *Height.* The influence of satellite height on orbital period is in accordance with Kepler's Third Law and defines the satellite's period of rotation.
  - *Ellipticity.* The *eccentricity* or *ellipticity*, *e*, describes the amount by which the orbit deviates from circular. Useful orbits for satellite communications have eccentricities between 0 (circular) and 0.99 (highly elliptical). Other orbits are the *hyperbolic orbit* ($e > 1$) and the *parabolic orbit* ($e = 1$) which have little use in communications since the orbit is not closed.
  - *Inclination.* The inclination of the orbit determines the track of the satellite over the Earth.

Circular Orbits

- Communications satellites commonly use a circular orbit, particularly the circular geostationary orbit. A circular orbit is a special case of an elliptical orbit, in which the two foci of the ellipse are coincident.
- As the satellite orbits the Earth, its position is governed by a balance of forces. The satellite moves tangentially to the Earth's surface with a velocity that was imparted at launch. If no other force were involved, the satellite would continue on out into space. However, the Earth's gravitational force acts in opposition to the satellite's tangential velocity, pulling the satellite towards its centre. This balance of forces results in a smoothly curved orbit path.

Circular Orbits

- A satellite moving around the circumference of a circle of radius r with an angular velocity ω will have a velocity $v=\omega r$. The velocity has a constant magnitude but is continually changing direction from a straight line as it travels around the curved path. This change in velocity means that the satellite is undergoing a continual acceleration away from the centre of the Earth of $\omega^2 r$ or v^2/r. This outward, or *centrifugal* (literally *centre-fleeing*), force on the satellite, F_{out}, is the product of the satellite mass times its acceleration:

$$F_{out} = \frac{m_s v^2}{r}$$

Circular Orbits

- The centripetal (literally, *centre-seeking*) force required to restrain the satellite from flying off at a tangent is provided by the Earth's gravitational pull. Newton's Law of Universal Gravity states that the force of attraction between any two objects is proportional to the product of their masses and inversely proportional to the square of the distance between them. So, the centripetal force on the satellite due to gravity, F_{in}, is:

$$F_{in} = \frac{Gm_e m_s}{r^2} = \frac{\mu m_s}{r^2}$$

Circular Orbits

- So, if the satellite is orbiting in a circle about the Earth, then the centripetal force on the satellite due to gravity, F_{in}, is exactly balanced by the centrifugal force, F_{out}. Equating F_{in} to F_{out} we have:

$$\frac{\mu m_s}{r^2} = \frac{m_s v^2}{r}$$

or

$$v = \sqrt{\frac{\mu}{r}}$$

Injection into Circular Orbit

- This equation gives the unique velocity at which a satellite must be injected to give a circular orbit of radius *r*. Note that it is independent of the mass of the satellite. Thus a satellite may be launched into a circular orbit at any desired height above the Earth, so long as, at the point of injection, its velocity is adjusted so that it is:
  – in a direction at right angles to the radius; and
  – of magnitude as given by the equation

$$v = \sqrt{\frac{\mu}{r}}$$

Geostationary Satellites

- A geostationary satellite must have a period equal to the period of rotation of the Earth, which is 24 hours—the time it takes for the Sun to reappear over the local meridian after a 360° rotation—or a *solar day*.
- However, during that time the Earth has moved a further 0.986° along its orbit around the Sun.
- Consequently, if we look at a day from the point of view of the stars, the time it takes for the Earth to rotate 360° around its axis (called a *sidereal day*, since it is an Earth day as viewed from the stars) is slightly shorter than 24 hours.
- Viewed slightly differently, the Earth takes 365.25 days, or a *solar year*, to travel around the Sun from our perspective.
- However, as far as the Sun is concerned, the Earth also travels once around the Sun, which means that the Earth rotates a total of 366.25 times.

Geostationary Satellites

- So, the sidereal day is slightly less than 24 hours and the period of rotation required for a geostationary satellite is:

$$T = \frac{365.25}{366.25} \times 24\text{hr} = 23.935\ \text{hr}$$

or, 23 hours, 56 minutes, 4.1 seconds.

- Expressing the velocity as the product of radius and angular velocity we have:

$$r\omega = \sqrt{\frac{\mu}{r}} \quad \text{or} \quad r = \left(\frac{\mu}{\omega^2}\right)^{1/3}$$

Geostationary Satellites

- Now, since the satellite is geostationary, its angular velocity is the same as that of the Earth—2π radians per sidereal day.

$$\omega = \frac{2\pi}{23.9345 \times 60 \times 60} = 7.29 \times 10^{-5}\ \text{rad/s}$$

Therefore, substituting for all values:

$$r = \left(\frac{6.67 \times 5.98 \times 10^{24}}{\left(7.29 \times 10^{-5}\right)^2}\right)^{1/3} \ \text{(m)}$$

$$= 42\ 164\ \text{km} \quad \text{above the centre of the Earth.}$$

Geostationary Satellites

- Since the Earth has an average[1] radius of 6 378 km (equatorial radius), this corresponds to an orbital height, h, above the Earth's surface of:

  h = 35 786 km

- Since $v=\omega r$, the velocity of the geostationary satellite is:

$$v = \frac{2\pi \times 42.164 \times 10^6}{23.9345 \times 60 \times 60} = 3.073\ \text{km/s} = 11\,063\ \text{km/hr}$$

1. The Earth is oblate—that is, it is wider around the Equator than around the poles. The Earth's mean radius is 6 371 km.

Orbital Perturbations

- So far, our discussion of satellite orbits has ignored the presence of the Sun and Moon and assumed the Earth to be a perfect sphere.
- The Earth is not a perfect sphere and the Sun and Moon exert a gravitational pull on an orbiting satellite.
- These effects give rise to orbital perturbations and discrepancies between the motion predicted earlier and the actual motion of the satellite.

Oblateness of the Earth

- Kepler's laws apply for bodies that are perfectly spherical. The Earth, however, is oblate (the diameter north-south is less than the diameter east-west by about 42.76 km[1]) and the equatorial circumference is not quite circular.
- These imperfections mean that the direction of the force of gravity acting on an orbiting satellite is not precisely towards the centre of the Earth but is slightly displaced causing the satellite to drift from its geostationary position.
- Station-keeping adjustments are therefore required every few months.

1. The Earth has an equatorial radius of 6 378 137.0m, a polar radius of 6 356 752.3142m, and a mean radius of 6 371 008.7714m.

Oblateness of the Earth

- Because the Earth's equatorial plane is elliptical, rather than circular, it has a semi-major axis along the line 165°E and 15°W, and a semi-minor axis along the line 105°W and 75°E.
- These points represent stable points in a geostationary orbit and satellites therefore tend to drift eastward toward the nearest stable point unless their progress is corrected by firing thrusters.
- These points are sometimes referred to as satellite graveyards, because satellites that are out of station-keeping fuel tend to drift eastwards towards these positions as their final resting place.

Oblateness of the Earth

- The effect of the oblateness of the Earth is also significant for satellites in inclined orbits where the ellipticity of the orbit is affected by the uneven distribution of the Earth's mass. The perturbations in the orbital plane cause a rotation of the perigee of an elliptical orbit—called the *rotation of the line of apsides*. We discuss these effects in more detail when we consider the Molniya orbit.
- The Earth's equatorial bulge also causes a rotation of the orbital plane around the Earth's north-south axis. The precession, in the opposite direction to the satellite's movement, for an orbital plane at inclination i, is:

$$\text{rate of precession} = 2.0617 \times r^{-3.5} \times \cos(i) \times 10^{14} \quad \text{(degrees per day)}$$

- which is approximately 4.9° per year for a geostationary satellite.

Oblateness of the Earth

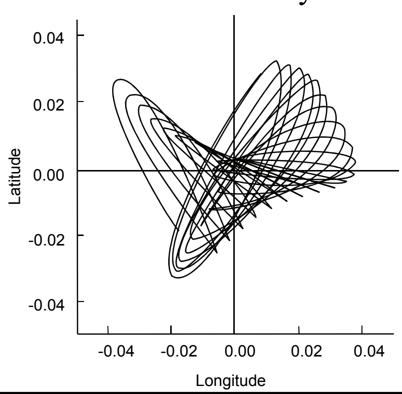
- Reconnaissance and remote-sensing satellites make use of this precession to operate in a Sun-synchronous orbit. Because the Earth itself makes a 360° rotation every 365.25 days, an orbital plane fixed with respect to the Earth rotates about 0.986° per day.
- Using the rate of precession equation, the height of the satellite and the angle of inclination can be chosen such that the precession of the satellite compensates for the orbital rotation due to the Earth's movement around the Sun and the satellite maintains a constant angular relationship with the Sun.
- This is very useful for Earth-observing satellites because they are then be over the same spot on the Earth's surface at the same time each day. Consequently, shadows are always in the same place and temporal variations are much easier to identify.

Lunar-solar Perturbations

- The most significant gravitational effect on a satellite comes from the Earth in the manner as described earlier. Although the Moon and, to a lesser extent the Sun, have a much smaller gravitational effect than the Earth, these other bodies do cause noticeable perturbations to the angle of inclination of the orbit of a geostationary satellite. This effect is not constant due to the fact that the Moon's own orbit is affected by the Sun.
- The drift in angle of inclination due to the Moon varies cyclically, increasing from approximately 0.48° to a maximum of 0.67° per year in a period of about 18.6 years. The drift due to the Sun is reasonably constant 0.27° per year so that, due to the combined effect of the two, the inclination of a geostationary satellite varies between 0.75° and 0.94° per year.
- If left uncorrected, the orbit inclination would vary between ±15° with 55-year cycle.

Lunar-solar Perturbations

- For satellites operating in C band (6/4 GHz), the drift must be kept within ±0.1°, which means that there is an uncertainty in true satellite location of about ±40 km. For Ku band (14/12 GHz) satellites the drift must be kept within ±0.05°. To reduce the requirement for tracking the satellite, north-south (*station-keeping*) manoeuvres are carried out once every few weeks by means of thruster jets.



Lunar-solar Perturbations

- The geostationary satellite also suffers longitudinal drift. Since longitudinal tolerance for satellites is of a similar magnitude to latitudinal tolerance, east-west station-keeping manoeuvres are therefore also carried out every few weeks, but require considerably less fuel than the north-south manoeuvres.
- Since the extra weight needed for station-keeping fuel is a major factor in the cost of launching a geostationary satellite, small geostationary satellites take an alternative approach where the orbit is initially inclined at 2° to 3°. The effect of the lunar-solar perturbation is to move inclination through 0° and back to the initial angle in a period of four to five years.
- The angle of inclination therefore remains acceptably small over the expected life of the satellite.

Lunar-solar Perturbations

- The lunar-solar gravitational pulls are in the opposite direction to the force resulting from the oblateness of the Earth and the two cancel out at approximately 7.5°. Consequently, without correction, the inclination of a geostationary satellite oscillates around that stable inclination to a maximum of 15° with a period of 53 years.
- Lunar-solar perturbations also affect satellites in elliptical orbit, although the effects are minor compared to the effects due to the Earth's oblateness.

Solar Radiation Pressure

- The effects of solar radiation pressure must also be considered for geostationary satellites, particularly for large satellites that require large solar arrays.
- Solar radiation pressure increases the orbital eccentricity of a geostationary satellite by up to 0.002 by introducing a disturbing torque resulting in an east-west movement about the mean longitude of up to about 0.25° per day.
- The extent of the disturbance depends on the surface area of the satellite presented to the Sun.
- Such perturbations are corrected periodically.
- Solar radiation pressure has some effect on satellites in elliptical orbit, although the effects are minor compared to the effects due to the Earth's oblateness.

Atmospheric Drag

- In lower orbits, a satellite suffers drag due to the friction caused by collision with atoms and ions in the Earth's atmosphere. This reduction in velocity in perigee causes the satellite to lose height in apogee on successive orbits. At altitudes below about 180 km, the effect of friction is so great that the excessive heat on the satellite causes it to burn up. The effect of atmospheric drag is significant out to orbital altitudes of at least 1 000 km and does not become negligible until 3 000 km.
- Atmospheric drag reduces the orbital lifetime of a satellite the orbit shape, its initial height, atmospheric conditions and mass of the satellite. Typically, a small satellite in very low-Earth orbit (say around 400 km) may only last a few months while a satellite of the same size in a circular orbit at 800 km could have an orbital life of several decades (much longer than an expected operational life of 10–15 years).

Doppler Effect

- Johann Doppler (1803–53) explained the *Doppler effect*, or the *Doppler shift*, which refers to the apparent change in frequency of sound when a transmitter moves with respect to a receiver—a higher pitch when the transmitter is moving towards the receiver compared with a lower pitch when it is moving away.
- The Doppler effect is also observed at RF, so that the frequency of satellite transmissions received by an Earth station increases as the satellite is approaching the Earth station and reduces as the satellite moves away.
- When the orbital parameters of the satellite are known precisely, Doppler shift can also be used to estimate the position of an observer or, alternatively, when the observer's position in known, Doppler shift can be used to estimate the orbital parameters of a satellite.

Doppler Effect

- The Doppler shift at a frequency f_0 (wavelength λ_0) is given by:

$$\Delta f = \pm \frac{v_r}{c} f_0 = \pm \frac{v_r}{\lambda_0}$$

- where v_r is relative radial velocity between the observer and the transmitter; and c is the velocity of light. The Doppler shift is positive when the satellite is approaching the Earth station, and negative when it is moving away.
- Formulae for estimating the Doppler shift are summarised in Appendix B of Richharia[1].

1. M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

Doppler Effect

- While Doppler shift is non-existent in ideal geostationary orbits and is negligible in paths to real geostationary satellites, it is significant in low-Earth orbits and highly elliptical orbits.
- The effect of Doppler shift is to increase the frequency stability of the carrier, which complicates the process of demodulation in the receiver.
- Additionally, distortion is introduced into the demodulated baseband since the modulated sidefrequencies are shifted by Δf on one side of the carrier and by the same amount in the opposite direction on the other side of the carrier.
- The sign of the Doppler shift is positive when the satellite is approaching the observer.

Other Orbital Effects

- In addition to the major effects in the preceding sections, there are a number of much more minor effects on a satellite's orbit.
- For example, the satellite is affected as it passes through the Earth's magnetic field, the platform is bombarded by micrometeorites, and RF radiation from large antennas cause pressure and torques on the platform.
- Each of these minor effects is generally negligible and any impact is corrected for when accommodating the major perturbations.
- We will not consider these effects any further.

Orbit Height

- In addition to their shape, orbits can also be grouped by their altitude:
  - *Low Earth orbit (LEO)* satellites have altitudes of 150 km to 1 500 km.
  - *Medium Earth orbit (MEO)* satellites have altitudes of 1 500 km to 35 786 km.
  - At 35 786 km above the equator the satellite is said to be in *Geostationary Earth orbit (GEO)*.

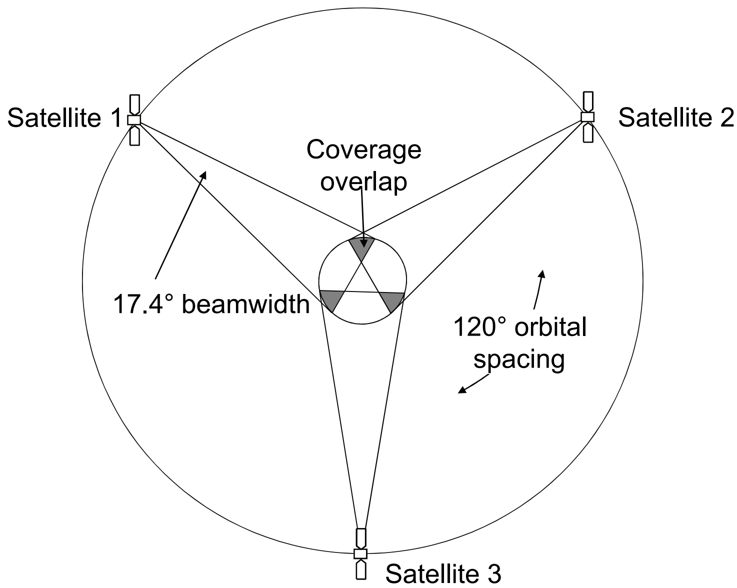
Geostationary Earth Orbits (GEO)

- As we have seen, a geostationary satellite has a period of rotation equal to that of the Earth so that it remains in a fixed position with respect to a given Earth station. The geostationary orbit is the orbit most commonly used for communications satellites.
- Each satellite must be placed carefully into predefined 'slots', or parking orbits, to ensure that the signals they transmit and receive do not interfere with nearby satellites.
- The satellite is kept in its slot by the use of its thrusters through the process of station keeping. When satellites are running low on station-keeping fuel, they are often moved into a much higher 'disposal' orbit and shut down.

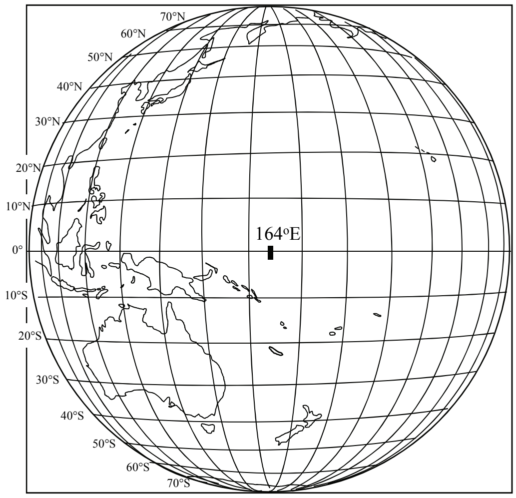
Geostationary Earth Orbit (GEO)

- GEO satellite are normally separated by 2°, providing 180 geostationary slots, although a number of advances now allow a single orbital slot to be shared by a number of satellites.
- GEO orbits can also be used for other purposes such as surveillance and signals intelligence, although acceptable image resolution and signal gain normally militates against such applications.
- At geostationary height, three satellites with a spacing of 120° (and a beamwidth of 17.4°) can provide world-wide coverage

Geostationary Orbit

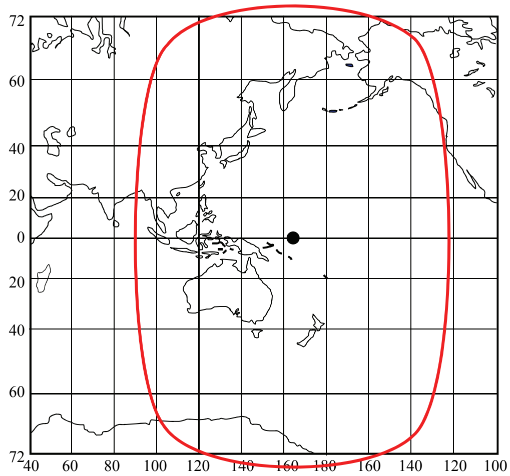


Geostationary Orbit - Earth View



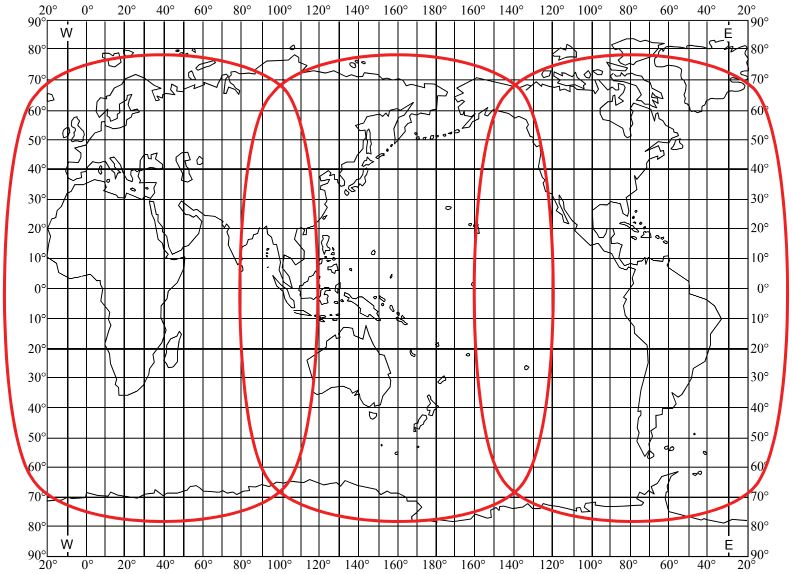
View from Optus B satellite at 164° E.
Other Optus satellites are at 152°, 156°, and 160° E.

Geostationary Orbit - Earth View



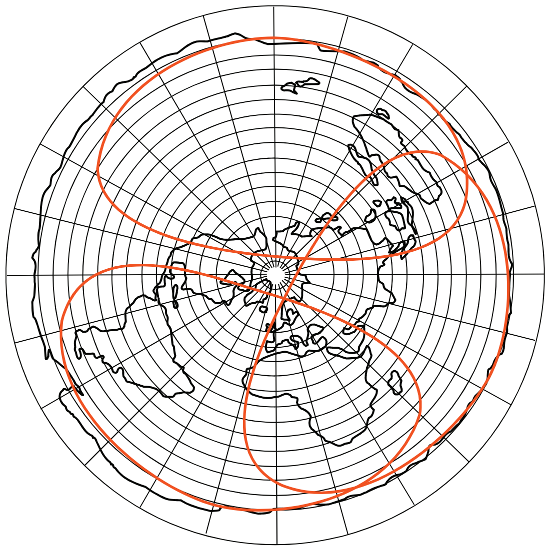
Area covered from Optus B satellite at 164° East

Geostationary Orbit - Earth Coverage



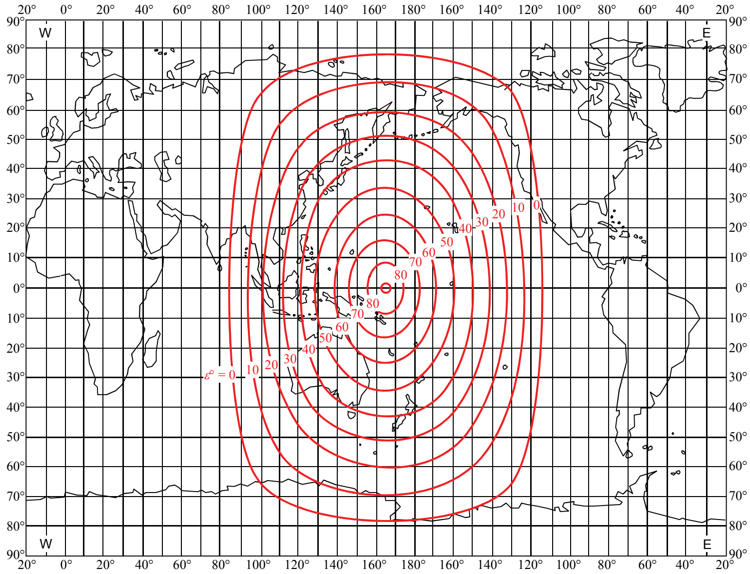
Area visible from three notional geostationary satellites (evenly spaced)

Geostationary Orbit - Earth View



Area visible from three geostationary satellites (unevenly spaced)

Effect of Elevation Angle



GEO Advantages

- As launch vehicles improved, GEO satellites became the most popular for telecommunications satellites because they have the following advantages:
  - only three satellites are needed to provide global coverage;
  - each satellite has a large footprint—at a geostationary altitude, the satellite has coverage of 17.4° cone angle, or about 42.4% of the Earth's surface;
  - each satellite remains effectively stationary with respect to the Earth (with the minor variations we discussed earlier), so that Earth stations can be of a simpler design because they do not require complex mechanisms to track the satellite with their antennas, and the need for handover is minimised in terminals used for personal communications systems;
  - …

GEO Advantages

- …
- the transmitted signal has only a small negligible Doppler shift caused by satellite movement;
- due to the constant geometry of transmission paths, interference between a particular geostationary satellite system and other terrestrial and satellite radio systems is predictable; and
- after many years of operating geostationary satellites, the technology and techniques required are well known.

GEO Disadvantages

- However, GEO systems have the following major problems:
  - high latency (delay)—the round-trip delay of 250 ms leads to a total of 500 ms for a two-way conversation (this is bearable for a telephone conversation over a single satellite hop, but a two-hop telephone conversation is very difficult—data communications are also very difficult with such delay);
  - high precision and sophisticated control are required to place a GEO satellite into orbit and to keep it there;
  - propulsion engines and sufficient fuel are required on board the satellites to overcome orbital perturbations and keep them in their respective orbits (station keeping);
  - …

GEO Disadvantages

- …
- poor coverage in the polar regions with no coverage beyond approximately 80° north and south of the equator;
- the large distance for the propagation path leads to high path attenuation requiring high-power transmitters, sensitive receivers, and large antennas at both the satellite and the Earth station;
- large losses due to shadowing in the higher latitudes as well as by buildings in the urban environment; and
- the limited number of orbital slots available above each country (although each slot can now be shared by a number of satellites).

Low Earth Orbits (LEO)

- The first LEO orbits were the low polar orbits that were originally used for navigation, remote-sensing and weather-forecasting systems. These satellites rotate north-south as the Earth spins beneath them so that they cover the Earth's surface in a series of scanning strips. LEO orbits are also popular for intelligence and reconnaissance systems because they facilitate the use of high-resolution sensors.
- The altitude of a LEO should be low enough to avoid the inner Van Allen belt and but high enough to avoid atmospheric drag. Altitudes for communications satellites are between 780 km and 1 400 km, having orbital periods between 100 and 113 minutes with velocities of 6–8 km/s. Consequently, approximately 60–70 satellites are required to provide continuous coverage.

Low Earth Orbits (LEO)

- LEO systems have a negligible round-trip delay of between 0.002–0.019s and have much lower attenuation than GEO or MEO systems which means that they are much better suited to the use of low-power, handheld terminals in mobile satellite service (MSS) systems.
- LEO satellites are now commonly used for personal communications systems (PCS) communications in such networks as Iridium LLC's *Iridium*, Loral-Qualcomm's *Globalstar*, Constellation Communications Inc's *ECCO*, and Ellipsat's *Ellipso*.
- LEO orbits are also popular for intelligence and reconnaissance systems, as they give good resolution. The CIA's KH-11 reconnaissance satellites reportedly have a 150-mm ground resolution[1].

1. J. Richelson, *America's Secret Eyes in Space*, Harper and Row, New York, 1990.

LEO Advantages

- LEO systems are more suited to personal communications services as they:
  - they require less power, which means that they can make use of small handheld terminals as and use small omni-directional antennas;
  - frequencies can be re-used more efficiently because of the smaller footprint;
  - even at a maximum LEO height of 1 400 km, the minimal propagation delay is less than 4.7 ms for each uplink and downlink leading to a total of 18.7 ms for a two-way conversation;
  - …

LEO Advantages

- …
- the satellite is overhead each user, which means that LEO systems are less subject to shadowing than GEO systems;
- a LEO system can provide communications coverage of the entire Earth (including polar regions); and
- satellite launching costs are lower as the satellites can be injected directly into orbit and several satellites can be launched at a time on the same launch vehicle.

LEO Disadvantages

- However, the use of LEO systems does have some difficulties:
  - the ability to cover the whole Earth comes at a cost of some 60–70 satellites, compared with three for GEO systems;
  - because satellites have such a short orbit period, frequent handover is required between satellites to provide uninterrupted communications;
  - although the whole Earth is covered, satellites spend considerable time covering uninhabited spaces such as oceans and deserts;
  - the proximity to the atmosphere means that satellites have a shorter orbital lifetime due to orbital decay as a result of atmospheric drag; and
  - satellite operational lifetimes are limited to 5–7.5 years because the satellite spends time in Earth eclipse, placing a significant demand on battery power.

Medium Earth Orbits (MEO)

- MEO—also called *intermediate circular orbits* (*ICO*)—have larger footprints than LEO and require fewer satellites to provide the same coverage. MEO altitude is between 5 000 km and 15 000 km above the Earth's surface with periods of 8 to 12 hours, requiring approximately 10 satellites for complete coverage of the Earth.
- Early launch vehicles could only reach the lower altitudes of MEO (up to 10 000 km) so they were the first orbits used for communications. As launch vehicles became more capable and could carry the greater payloads required, satellite communications systems almost exclusively used GEOs, despite the drawbacks of large Earth stations and long delays.

Medium Earth Orbits (MEO)

- Although GEO will continue their domination broadcast satellite systems, the MEO orbits are particularly attractive for mobile satellite systems because the lower transmission path loss allows the use of handheld terminals with much lower power and simple omnidirectional antennas.
- The proposed ICO system of ICO Global Communications is an example of a MEO system.

MEO Advantages

- MEO systems are generally better suited to personal communications than GEO and have the following advantages :
  - MEO orbits are in the slot between the two Van Allen belts;
  - a delay of 17–50 ms for each uplink and downlink leads to an acceptable total of 67–200 ms for a two-way conversation;
  - each satellite has a relatively large footprint so that only a few (~10) satellites are required to cover the whole Earth;
  - fewer satellites means that inter-satellite handover is not as frequent as for LEO systems;
  - battery lifetimes of longer than seven years are possible due to fewer eclipse cycles than LEO;
  - …

MEO Advantages

  - …
  - lower cosmic radiation leads to longer operational lifetimes;
  - communication between the terminal and the satellite occurs more often at a higher average elevation angle, reducing shadowing; and
  - shorter slant ranges require approximately 10 dB less power than GEO systems requiring smaller satellites and smaller terminals (particularly useful for mobile communications).

MEO Disadvantages

- However, the MEO systems also have some disadvantages:
  - each satellite can service a much smaller area than GEO, providing a less-satisfactory repeater for telecommunications networks;
  - Doppler frequency offsets are lower than for LEO but are larger than GEO due to higher relative satellite motion;
  - frequency coordination is more difficult and highly unlikely to be able to achieve the frequency re-use of GEO;
  - although fewer satellites are required than LEO, the trade-off between number of satellites and latency is generally considered sub-optimal;
  - handover is still required between satellites; and
  - satellites still spend considerable time covering uninhabited regions of the Earth's surface.

Highly Elliptical Orbits (HEO)

- HEO satellites have an apogee that may be beyond GEO at which the satellite appears *pseudo-stationary*, remaining within a beamwidth of less than ±15º for more than eight hours over the service area.
- Unlike GEO, HEO orbits (sometimes called *super-synchronous* orbits) also cover the polar regions, which is why a HEO orbit was chosen for the *Molniya* system to provide coverage of the USSR.

HEO Advantages

- HEO systems have the advantages of:
  - lower launching costs than GEO due to the lower energy required,
  - inclined orbits can provide visibility to the higher northern and southern latitudes, and
  - lower atmospheric loss in higher latitudes.

HEO Disadvantages

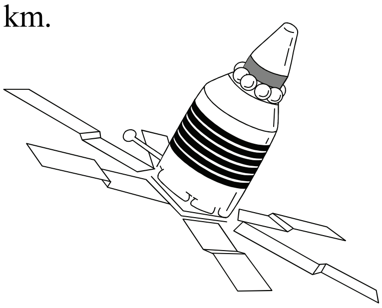
- However, HEO have significant disadvantages such as:
  - satellites must pass through the Van Allen belts twice each orbit which increases the protection required,
  - the requirement for continual tracking of the satellite by the Earth station,
  - a possible requirement for handover between satellites,
  - severe Doppler effects,
  - higher path loss than GEO due to longer distance in apogee,
  - inefficient use of spectrum compared to the frequency re-use of GEO,
  - …

HEO Disadvantages

- ….
- higher system costs due to the need for three or four satellites for continuous coverage,
- extensive eclipse periods,
- operational lifetimes are limited by atmospheric drag near perigee,
- signal fading, and
- more complex control of satellites and Earth stations.

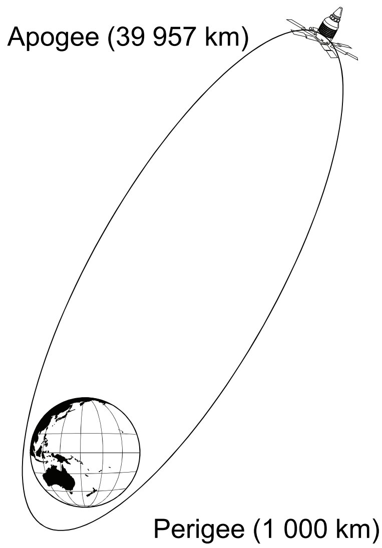
Elliptical HEO - Molniya

- One of the difficulties with geostationary satellites is the lack of coverage of high and low latitudes. One of the more interesting orbital satellite systems is the Soviet *Molniya* system.
- This is also spelled *Molnya* and *Molnia*, which means “lightning” in Russian (in colloquial Russian it means “news flash”).
- The Molniya satellites are used for television broadcasting.
- Molniya uses a highly elliptical orbit with apogee at about 40 000 km and perigee at about 1 000 km.



Elliptical HEO - Molniya

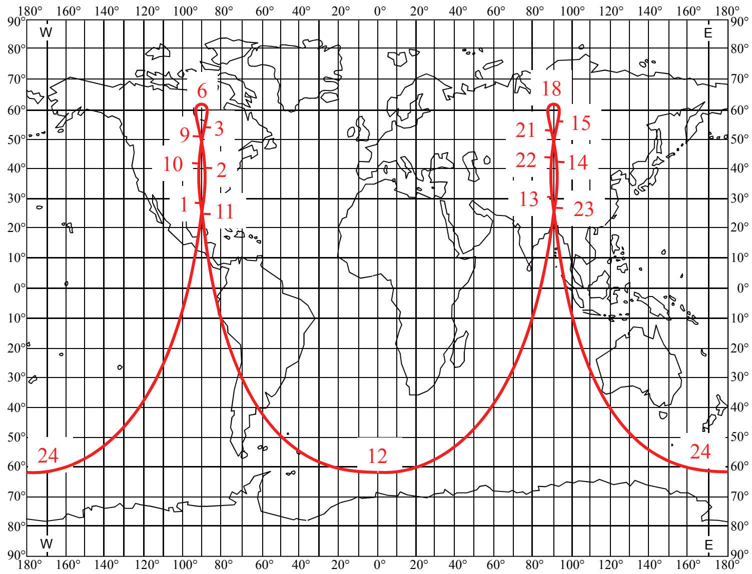
- The Molniya apogee is reached over the Northern Hemisphere and the perigee over the Southern Hemisphere.
- The orbit is inclined at 63.4° to the equatorial plane so that the satellites pass over the landmass of the Russia.
- The size of the ellipse was chosen with an eccentricity of *e*=0.75, to make its period exactly one-half of a sidereal day (the time it takes the Earth to rotate back to the same constellation).



Elliptical HEO - Molniya

- The orbit inclination is chosen for a particular reason. One effect of the Earth's equatorial bulge is to cause the orbit to rotate, such that the apogee and perigee move around the Earth - called *rotation of the line of apsides*. However, at an inclination of 63.4°, the rotation is zero, meaning that the satellites have an apogee that remains fixed over a particular region.
- Because of its unique orbital pattern, the *Molniya* satellite is synchronous with the rotation of the Earth. During its 12-hour orbit, it spends about 11 hours over the Northern Hemisphere.
- Continuous communications coverage of the operational region is possible by spacing three or four satellites. Each satellite is normally only turned on for operation at altitudes above 15 000–20 000 km.

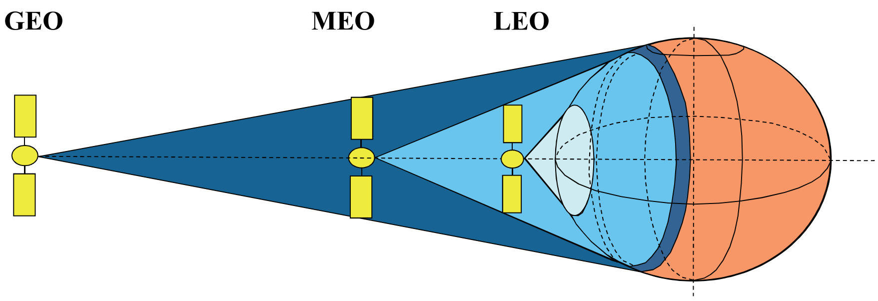
Elliptical HEO - Molniya



Orbit Selection

- The selection of orbit is a trade-off of a number of characteristics, the solution to which is effectively dictated by the specifications of the ground terminal.
- If a small handheld terminal is required, satellites need to be in LEO—or at least MEO—so that the terminal can use low powers and a small omnidirectional antenna.
- However, at these lower altitudes, each satellite's footprint is significantly reduced and a number of satellites are required depending on the altitude. If low system cost and wide-area coverage is required, GEO systems are probably best suited.

Circular Orbit



Orbit Selection

Characteristic	LEO	MEO	GEO
Satellite height (km)	600–1 500	5 000–15 000	35 786
Orbital period (hr)	1.5–2	3–8	24
Number of satellites	60–80	8–20	2–4
Two-way delay (ms)	8–20	67–200	500
Satellite life (years)	5–7.5	10-15	10–15
Elevation angle	medium	best	good
Visibility of satellite	short	medium	permanent
Handheld terminal	small	small–medium	medium
Handover rate	frequent	infrequent	none
Space segment cost	maximum	minimum	medium
Gateway cost	highest	medium	lowest
Network complexity	complex	medium	simplest
RF output power	low	medium	high
Propagation loss	low	medium	high

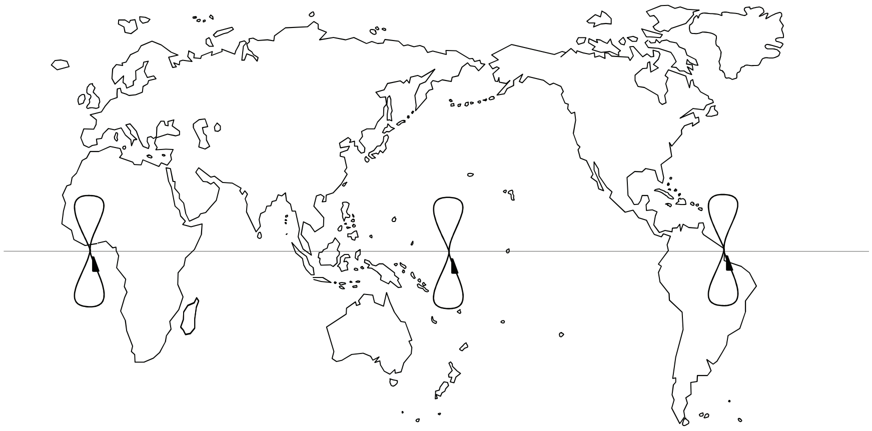
Satellite Ground Traces

- The ground trace of a satellite is the path traced out by the line of apsides on the Earth's surface (that is, traced out by the sub-satellite point).
- The ground trace is therefore produced by a combination of two motions: the Earth's rotation and the satellite's orbit.

GEO Ground Trace

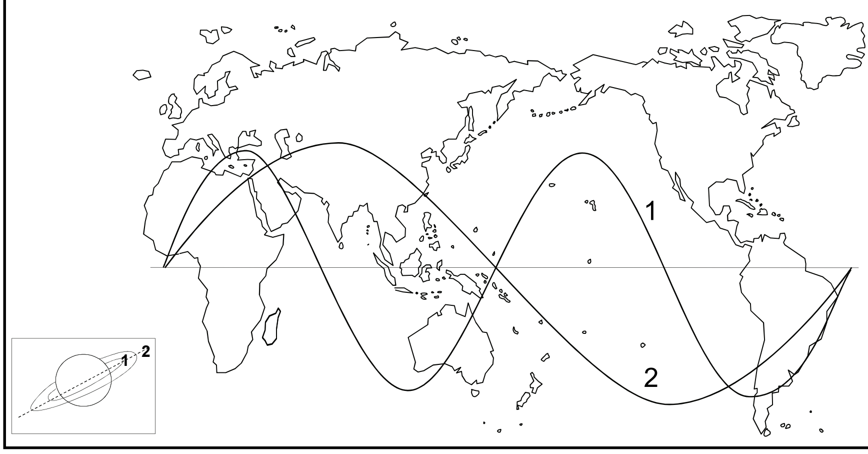
- A true geostationary satellite would remain stationary over a fixed point on the equator (0° inclination).
- The ground trace of a real GEO satellite looks like a figure-of-eight, the latitude extremes of which are determined by the inclination of the orbit.
- Even if the inclination of a GEO is exactly zero, the satellite will still wander a little due the orbital perturbations discussed earlier.

GEO Ground Trace



Effect of Period

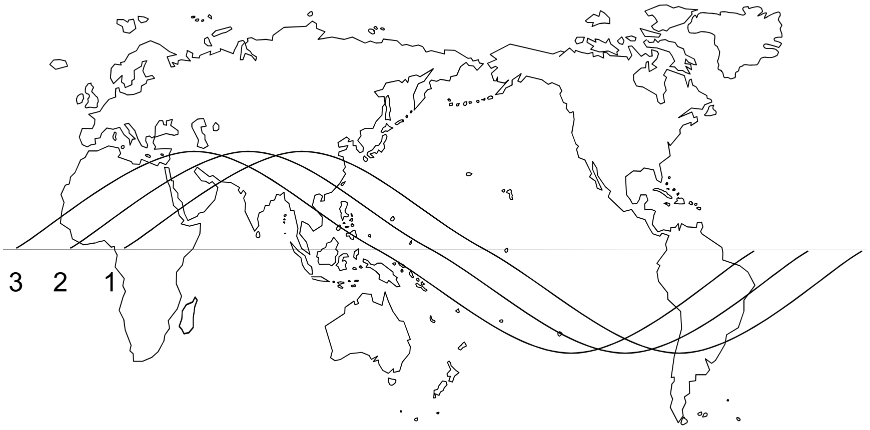
- As the altitude of the orbit increases, so does its period. Satellite 2 has the higher altitude is travelling more slowly and will therefore cover a smaller angular distance in a given time. Consequently the higher satellite's ground trace will have the lower frequency.



Apparent Regression of Nodes

- If the satellite orbit is higher than GEO, the period is longer than twenty four hours and the ground trace is displaced westward by the difference between the number of degrees the Earth and the satellite rotate during the period of the orbit.
- Eventually the ground traces of the satellite will cover a band of the Earth's surface bounded by the latitudes equal to the orbit inclination.
- Careful selection of the orbit altitude ensures that the satellite cyclically repeats the same ground trace sequence, which is particularly useful for surveillance and remote-sensing satellites that can then observe temporal change.
- For orbits lower than GEO the period is shorter than twenty four hours and the ground trace moves eastward.

Apparent Regression of Nodes

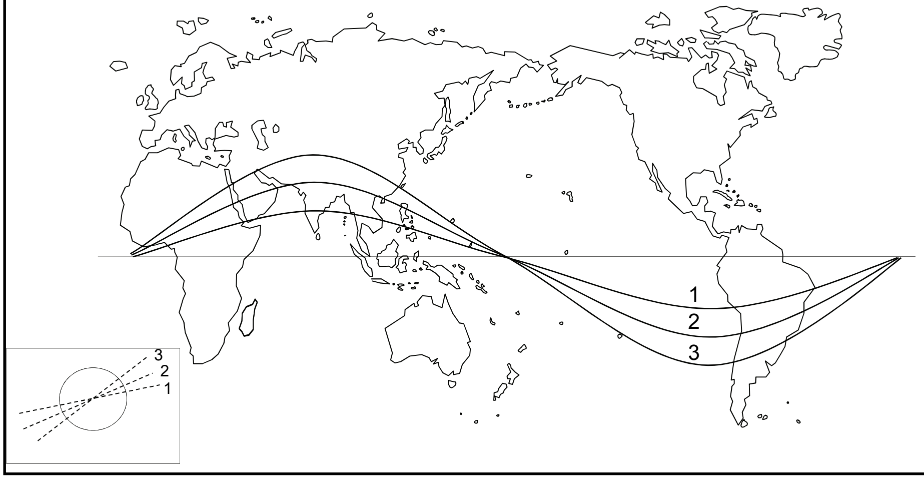


Effect of Inclination

- The satellite's orbital inclination can be measured by measuring the latitude of the northern-most turning point of the ground trace.
- The inclination will equal the measured latitude for prograde orbits and will be 180° minus the measured latitude for retrograde orbits.
- Note that inclination has no effect on period, which is determined solely by the altitude of the satellite.

Effect of Inclination

- This figure shows the ground traces of three satellites with identical periods, but different inclinations (20°, 35° and 50°).

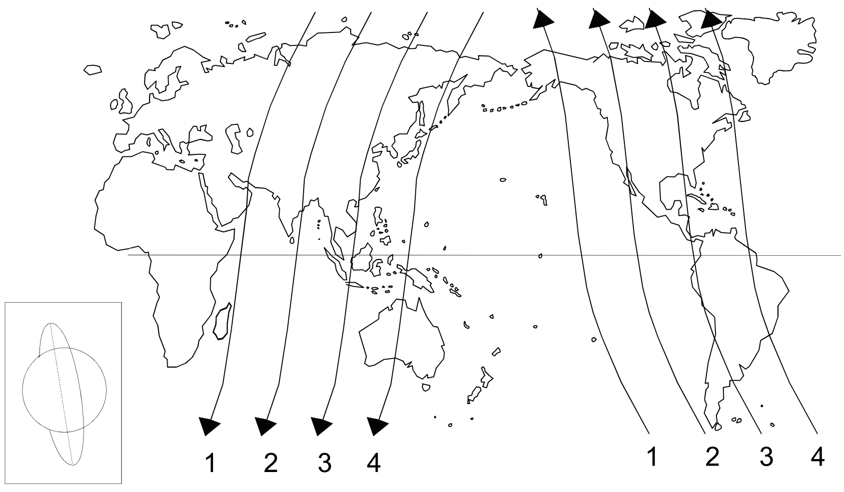


Polar Orbit

- Polar orbits have inclinations at or near 90°. If the inclination is exactly 90°, then the satellite will transverse both poles during orbit. If the inclination is slightly less than 90°, then the ground trace will turn at the latitude of inclination as discussed earlier.
- A satellite in polar orbit inclined at 90° will eventually cover the entire Earth's surface, whilst a satellite inclined more or less than 90° will cover most of the surface (less the extremes of latitude).
- The polar orbit is most often used for weather, reconnaissance and remote sensing satellites, where total, or near total, coverage is required.

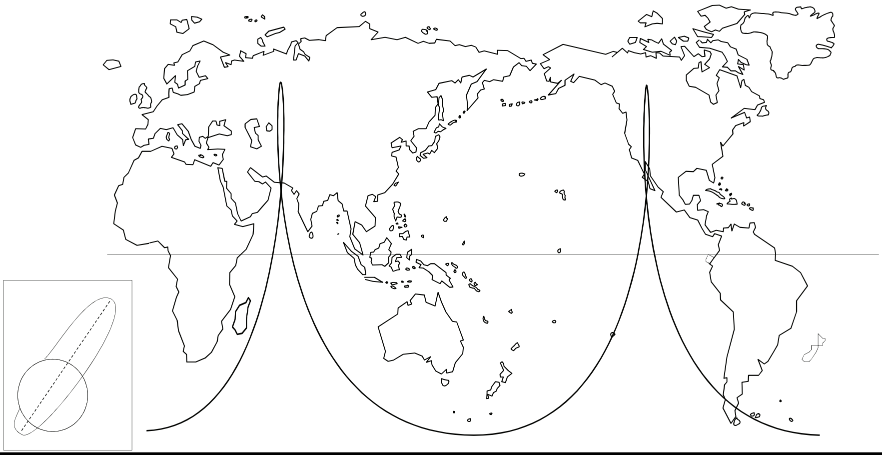
Polar Orbit

- This figure shows a polar orbit that is slightly retrograde: that is, it has an inclination greater than 90° so that the ground trace moves east with each orbit.



The Effect of Eccentricity

- This figure shows the ground trace of a highly eccentric orbit. Perigee (where the satellite is moving fastest) is over the southern hemisphere, while apogee (where the satellite is moving slowest) is over the northern hemisphere.



Satellite Launch

- We saw earlier that an elliptical orbit can be defined by the velocity of a satellite in that orbit through the relationship:

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

- Which, for circular orbits, reduces to:

$$v = \sqrt{\frac{\mu}{r}}$$

- so that geostationary satellites require a velocity of 3 073 m/s at 42 164 km from the Earth's centre

Satellite Launch

- A satellite can therefore be launched into any orbit at any desired height above the Earth, so long as, at the point of injection into the orbit, its velocity is adjusted so that it is of the magnitude given by

$$v = \sqrt{\frac{\mu}{r}}$$

- and in a direction at right angles to the line joining the satellite and the centre of the Earth.

Satellite Launch

- To launch a satellite into any orbit, the launch vehicle must perform three chief functions:
  - The launch vehicle must be able to carry the satellite to the required orbit height. Since a large amount of energy is required to escape the Earth's gravitational field, normally the satellite is launched to perigee to minimise fuel usage.
  - Once at the desired height, the launch vehicle must align its direction at right angles to the radius of the orbit. While the alignment can be made at any point for circular orbits, apogee and perigee are preferred points for elliptical orbits to save fuel. Normally, to minimise atmospheric drag, the vehicle is launched vertically and gradually rotated so that it is at right angles at the point of injection.
  - Finally, the satellite must be given an incremental boost to bring its final velocity to that appropriate to the orbit.

Satellite Launch

- The maximum velocity increment imparted by a launch vehicle is:

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

- where m_o is the total mass, v_g is the effective exhaust velocity of the propellant; and m_f is the mass of the fuel.
- The ratio of m_f to m_o should be maximised to maximise Δv.
- Therefore it is usual for rockets to have multiple stages, each stage being jettisoned after imparting a given thrust, so that as m_o is reduced, succeeding stages of rockets need to impart progressively lower thrust to achieve the desired orbit.
- The final velocity of the spacecraft is the sum of the velocity increments of all the stages.

Satellite Launch

- Consideration of the required velocity starts as soon as the satellite is launched. Since the Earth rotates to the east, a satellite launched towards the east will be given a boost in that direction.
- Since the Earth rotates at a velocity of 467 m/s at the equator and 0 m/s at the poles, the largest velocity boost can be obtained from launching in an easterly direction at the equator.

Satellite Launch

- Launching from the equator has another advantage for geostationary satellites since the resultant orbital inclination of a satellite, i, is defined by the latitude of the launch site, L_e, and the azimuth of launch, ξ:

$$\cos(i) = \sin(\xi)\cos(L_e)$$

- Note that a satellite cannot be launched into an orbit with an inclination less than the launch latitude. Additionally, since an inclination of 0° is required for a geostationary orbit, the desired orbit can be obtained by launching eastwards (ξ=90°) from a launch site on the equator (L_e=0). Launching with any other azimuth and inclination leads to an orbital inclination other than 0° from which a correction must be made (and fuel expended) by adding a velocity increment perpendicular to the orbital plane to bring the inclination back to 0°.

Satellite Launch

- Additionally, the time of the launch is important since the position of the satellite needs to favourable with respect to the Sun (for power supply and thermal control) and must be visible to the control station during the most critical manoeuvres. These factors dictate a *launch window*, of which there is one per day for direct insertion into orbit when the launch latitude equals the orbital inclination and two per day for a launch latitude that is less than the desired inclination.
- There are no land-based launch sites at the equator—the closest is the French site of Kourou in French Guiana in South America, which has a latitude of 5°N. Cape Kennedy in the US has a latitude of 28°N. The *SeaLaunch* system has been developed to sail a launch platform out to sea to obtain the ideal launch conditions launching east on the equator.

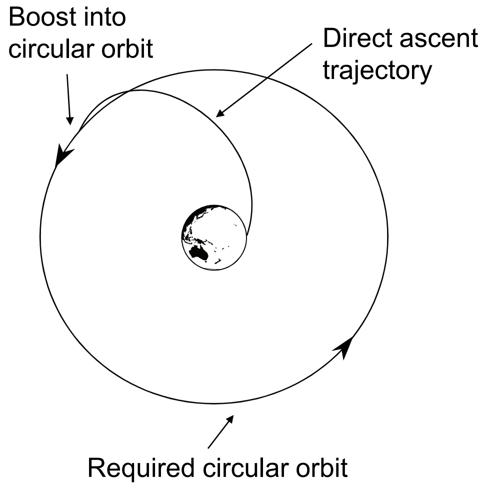
Geostationary Satellite Launch

- There are two methods used to launch a satellite to the required height:
  - Direct Ascent
  - Hohmann Transfer Ellipse

Direct Ascent

- The thrust of the launch vehicle is used to carry the payload in a trajectory with turning point just above the required height.
- As the satellite falls back it passes through the desired orbit.
- At this point the final stage of the launch vehicle is ignited or re-ignited to achieve the necessary velocity increase required for circular orbit.
- The direct ascent represents a 'head-on' approach to the problem of satellite launch. It is inefficient for all except very low orbits and is not used for the launch of satellites to geostationary height because of the prohibitive quantity of fuel that would be needed.

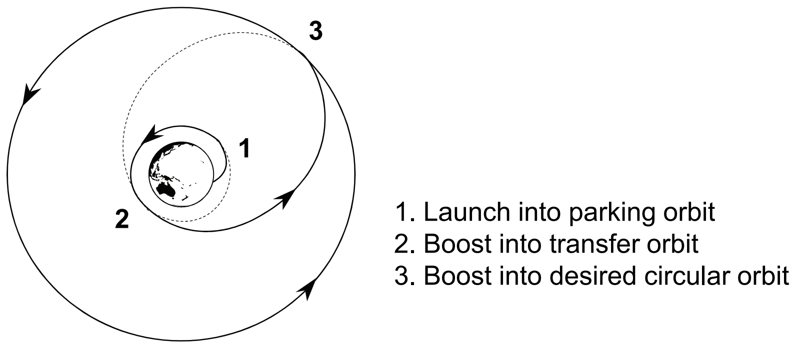
Direct Ascent



Hohmann Transfer

- For higher orbits such as GEO, the payload is first launched into an initial low parking orbit (185–250 km).
- The launch vehicle is then boosted into an elliptical transfer orbit, whose perigee is where the boost was applied and whose apogee is in the desired final circular orbit.
- Once in apogee, the satellite is then boosted again to move into circular orbit.
- This technique of transferring between two circular orbits via an elliptical transfer orbit is called a *Hohmann transfer*.

Hohmann Transfer



Hohmann Transfer

- The satellite is normally boosted into elliptical transfer orbit using the final stage of the launch vehicle's rockets. The final stage is then discarded and satellite takes some five and a half hours to drift to apogee where its own apogee kick motor (AKM), or apogee boost motor (ABM) is used to transfer to circular orbit.
- This does not occur on the first apogee—a number of orbits are undertaken giving controllers time to determine the parameters required to identify precisely the velocity increment required to circularise the orbit.
- At the same time as the orbit is circularised, a velocity increment may be applied in a direction perpendicular to the orbital to correct for the initial inclination of the orbit (which is equal to the latitude of the launch location).
- The satellite is then corrected gradually for position and altitude before being made available for operations.

Hohmann Transfer

- There are four distinct phases involved:
  - *Powered ascent.* A rocket or the Space Shuttle is used to position the satellite in a circular parking orbit (of height typically 88 km). Depending upon particular requirements some adjustment of the inclination of this orbit is sometimes made, but such adjustments are better made at geostationary height where the satellite is moving more slowly.

Hohmann Transfer

  - *Transfer orbit.* The third stage motor of the launch vehicle is re-ignited to boost the satellite into an elliptical orbit of perigee equal to the radius of the parking orbit and apogee equal to the geostationary orbital height. The third stage is then jettisoned, leaving the satellite and its integral *apogee-boost motor (ABM)*, or *apogee-kick motor (AKM)*, to drift through space to the required height. This transfer takes approximately 5.45 hours. Injection into circular orbit is not usually performed on the first apogee of the transfer orbit. This enables data to be recorded to enable an accurate computation of the velocity increment to be made, and the ABM is commonly fired on the fifth or seventh pass of the satellite through the transfer orbit's apogee. This ABM firing increases the satellites velocity so that its orbit is circularised.

Hohmann Transfer

– *Drift orbit.* The satellite is now allowed to drift in space while corrections are made to its altitude and position using the on-board thrust jets. Corrections are made in small steps once per orbit.
– *Final orbit.* The satellite is then halted when on station and its sub-systems are tested before being made available for service.

- There is a trade-off between fuel consumed and time taken for the launch sequence, but 21 days is usually sufficient for the drift and station acquisition phases.

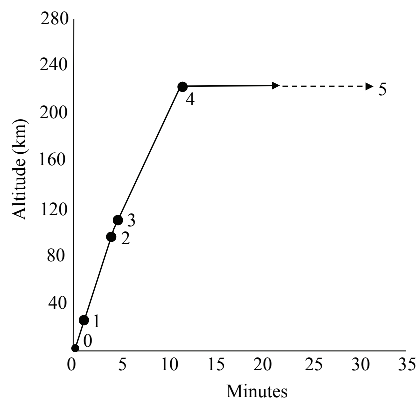
Launch Into LEO, MEO and HEO

- Other satellite orbits are achieved in a similar manner to GEO. LEO satellites are first launched into a parking orbit and then manoeuvred into the desired final orbit using the techniques described for the Hohmann transfer.
- MEO satellites have circular orbits that are not as high as GEO, so they are launched in the same manner, but are transferred into circular orbit considerably earlier.
- HEO satellites are launched like GEO except that the final circularisation manoeuvres are not performed.

Launch Vehicles

- Satellites can be launched either by expendable rockets or by the reusable systems such as the Space Shuttle, although following the Challenger explosion in 1986, use of the Shuttle has been reserved for government and scientific activities.
- Expendable systems can be used for direct-ascent or transfer launches, reusable systems can only be used to place satellites into low-Earth orbit.
- Reusable systems are also able to retrieve satellites in low orbit for repair.
- The current lift capability of the larger rockets is similar to the Shuttle, which can carry some 25 000 kg into low eastern orbits and approximately half that amount into polar orbits.

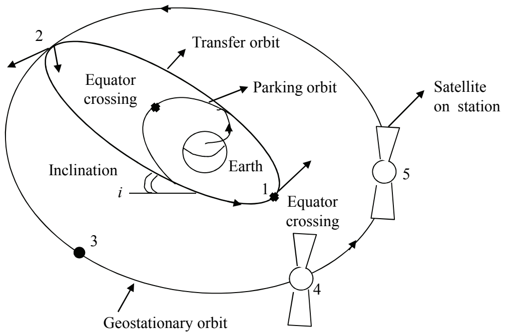
Expendable Launchers



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

M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

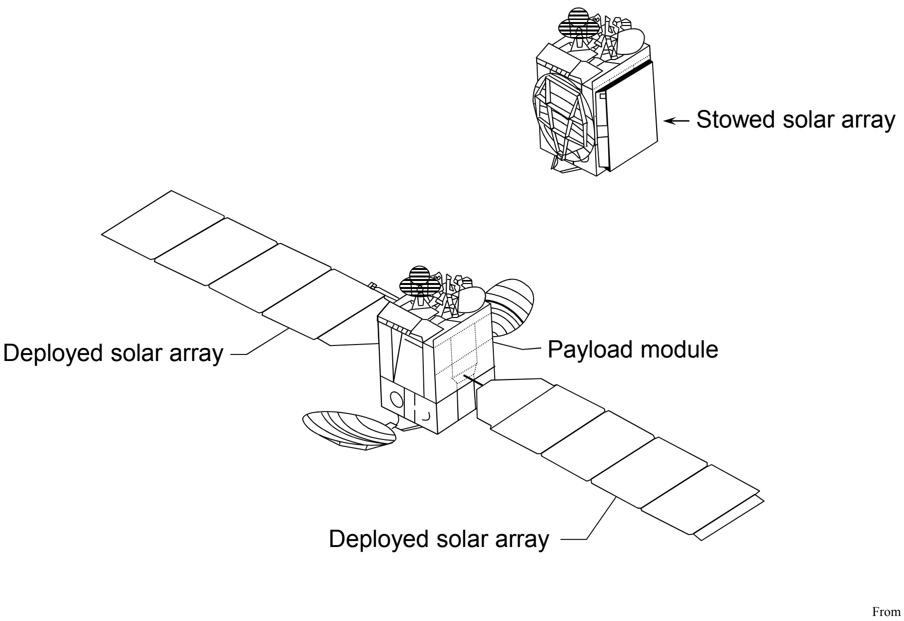
Expendable Launchers



Event number	Event
1	Velocity increment to acquire transfer orbit; satellite spun for stabilisation; attitude manoeuvres done before ABM firing
2	ABM fired to give necessary velocity increment; orbit circularised and inclination reduced to near zero
3	Satellite despun
4	Three-axis stabilisation acquired
5	Minor orbit corrections performed to correct residual orbital errors in orbit tests and position satellite on station

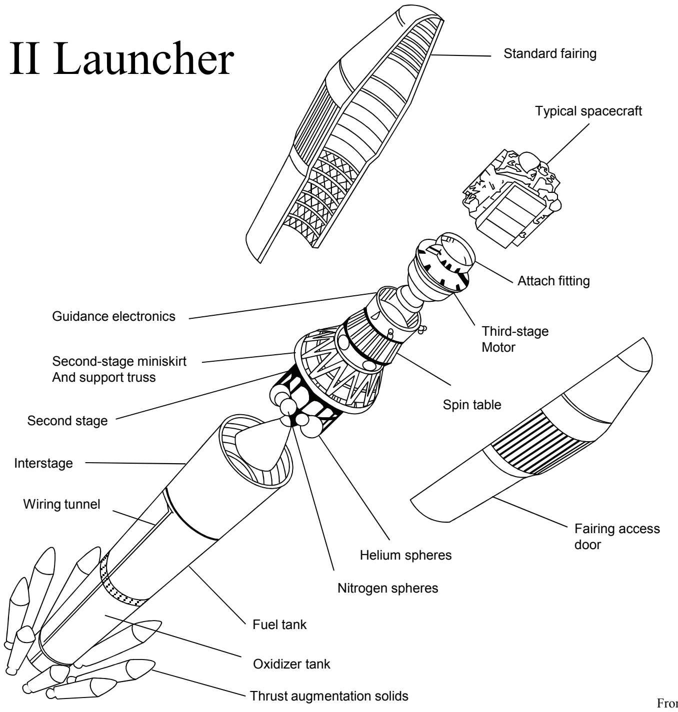
M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

Intelsat VIII – Launch & Orbit State



From Intelsat

Delta II Launcher



From McDonnell Douglas

Orbital Manoeuvring

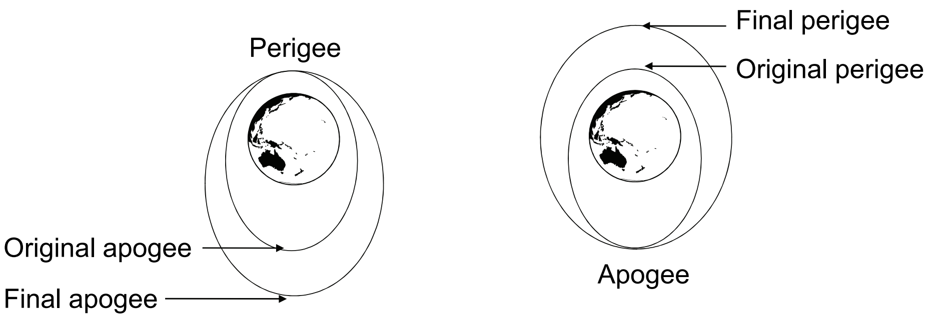
- We have already seen that, after launch, satellites may require a series of orbital manoeuvres to attain the required orbit. Since a satellite's orbit is effectively defined by its velocity, by changing velocity through the application of thrust it is possible to change orbits, either in the same orbital plane, or between planes. Such manoeuvres are not undertaken lightly, however, as satellites travel at very high velocities that require the expenditure of significant amounts of fuel to effect any changes. Major orbital manoeuvring therefore reduces the satellite's operational life by consuming station-keeping fuel.
- An orbit can therefore be modified by the application of thrust in the same or opposite direction to the satellite's velocity. If thrust is applied in the direction of travel, the orbit increases in size; if applied in the opposite direction, the orbit decreases in size. The point where the application of thrust was terminated is a point in both the old and new orbits.

Orbital Manoeuvring

- The orbit can be modified by the application of thrust.
- If thrust is applied in the direction of travel, the orbit will increase in size.
- If the satellite is turned and thrust is applied in the opposite direction (called retro-firing), the orbit will decrease in size.
- The simple rule concerning changing orbital parameters is that: *a satellite will always return to the point at which the application of thrust was terminated*.

Orbital Manoeuvring—Changing Eccentricity

- Eccentricity can be changed by adding energy to the orbit. If the satellite is boosted at perigee, the orbit becomes more elliptical. Note that the point of perigee (where the thrust was applied) is the same in both orbits.
- If the satellite is boosted at apogee, the orbit also increases in size, but becomes less elliptical.



Orbital Manoeuvring — Changing Altitude

- Changes in altitude can be achieved simply by boosting to transfer from any point in the lower orbit into a temporary elliptical orbit that will intersect the desired higher orbit and then boosting again as the satellite approaches the point of intersection with the new orbit.
- Although this simple transfer technique is quick, it consumes significant quantities of fuel.
- Much less fuel can be expended using a Hohmann transfer where the altitude is increased with a first boost at the original perigee to increase the eccentricity of the old orbit more eccentric, and a second boost at the new apogee to circularise the orbit.
- Decreasing orbital altitude can be achieved by the same process using retro-firings instead of boosts.

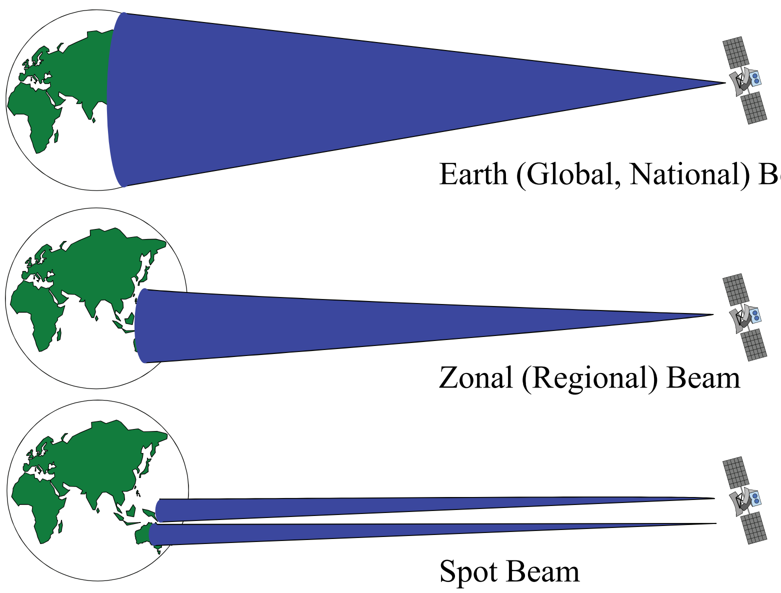
Orbital Manoeuvring — Changing Plane

- Changes in orbital inclination can be achieved without affecting other orbital parameters by a boost perpendicular to the orbital plane at either the ascending or descending node of the original orbit.
- Again, the point of boost is common to both orbits.

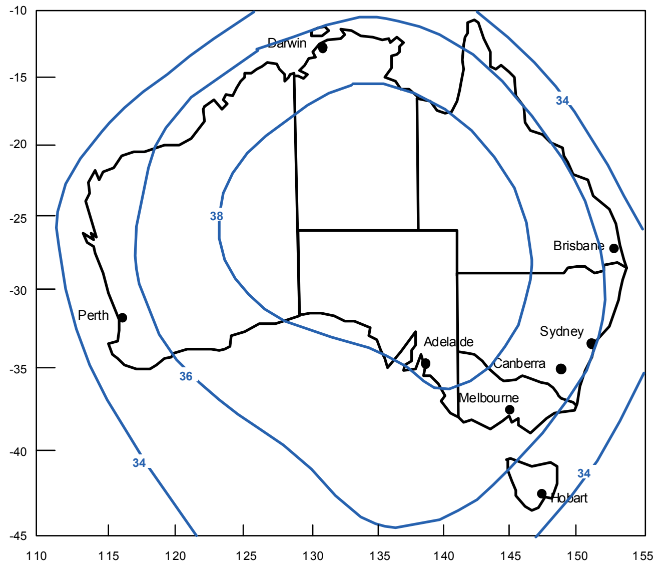
Footprint

- The area of the Earth covered by the radiation from a satellite is called its *footprint*. The size of the footprint depends on the location of the satellite in its orbit and the shape and size of beam produced by its antennas.
- The radiation pattern from a satellite antenna may be categorised as either *spot*, *zonal*, or *Earth*.
- The radiation patterns of Earth-coverage antennas have a beamwidth of approximately 17° and include coverage of approximately 40% of the Earth's surface.
- Zonal coverage includes an area of less than 40% of the Earth's surface. Sometimes large zone beams (approximately 20% of the Earth's surface) are called *hemisphere* beams.
- Spot beams concentrate the radiated power in a very small geographic area.

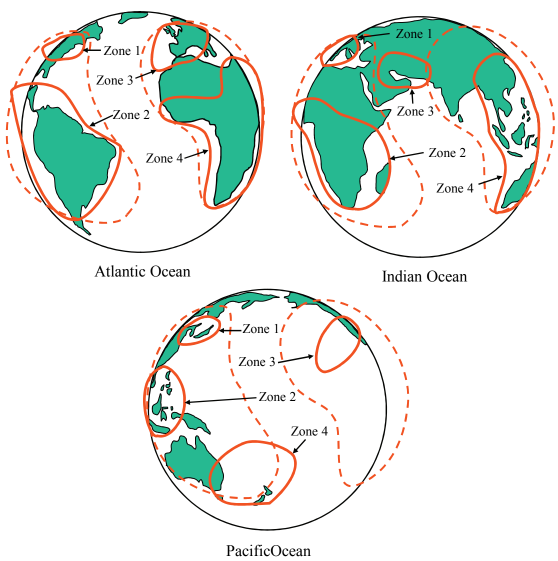
Ground Coverage Beams



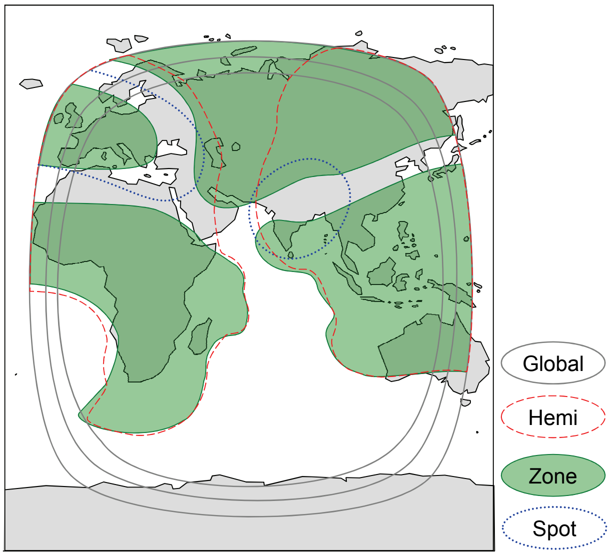
Optus - Earth Beam



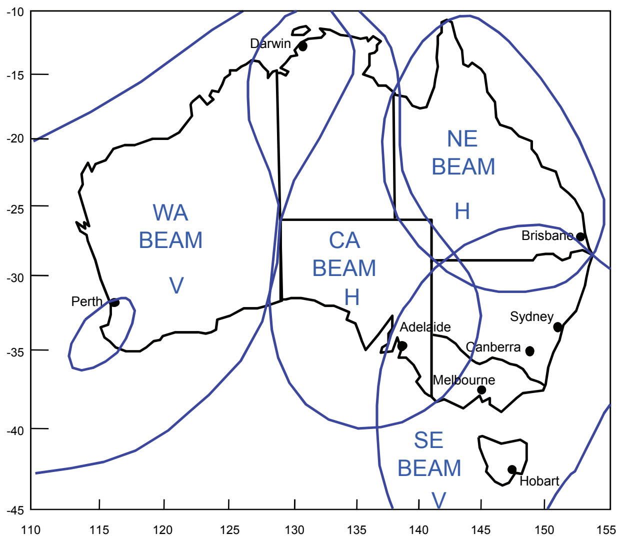
Zonal Beams - Intelsat Coverage



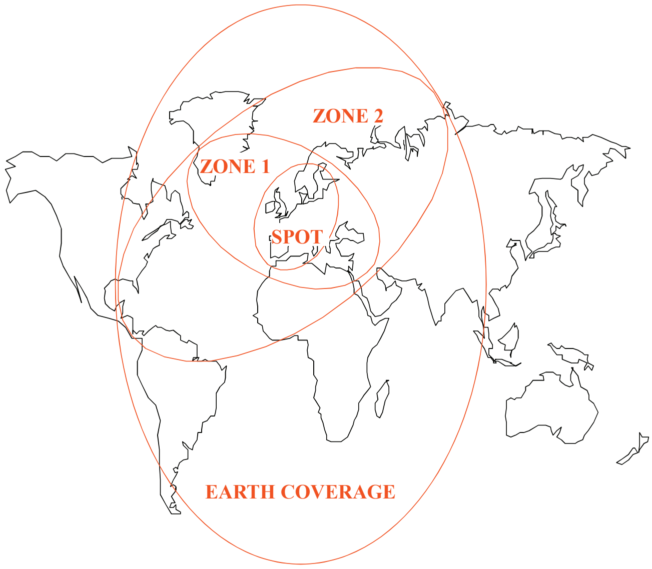
All Beams - Intelsat IX (64°E) Coverage



Spot Beams - Optus



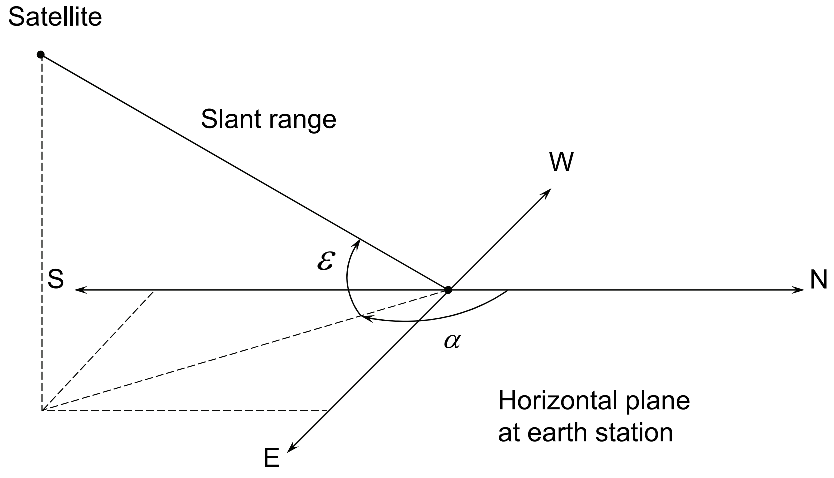
Earth, Zonal and Spot Beams - UK Skynet IV



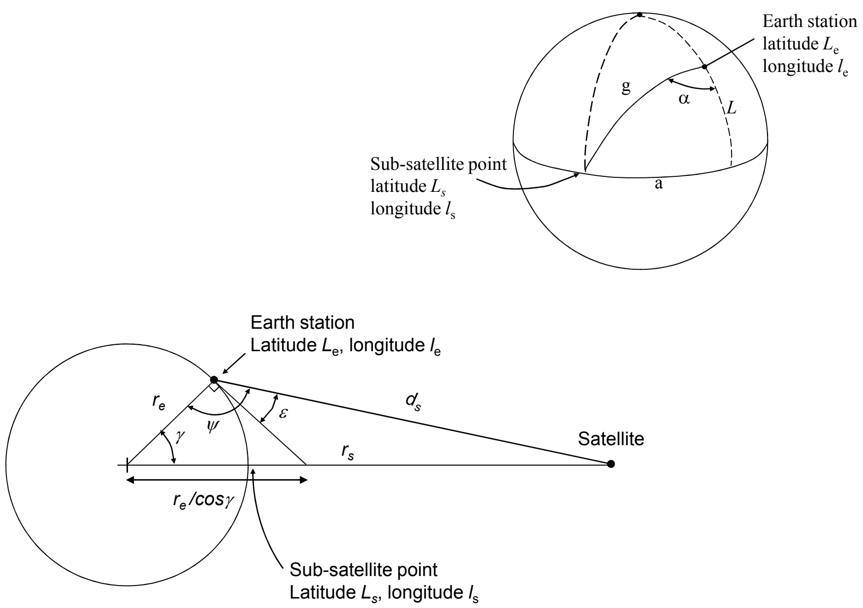
Look Angles

- The point at which a line between the satellite and the centre of the Earth intersects the Earth's surface is called the *sub-satellite point*.
- The distance from the Earth station to the satellite is called the *slant range*.
- To orient an Earth station toward a satellite, it is necessary to know the *look angles* of the satellite relative to the Earth station:
  - The *elevation* is the angle of the satellite above the horizon from the Earth station.
  - The *azimuth* is the angle between the line of longitude through the Earth and the direction of the sub-satellite point.

Look Angles



Look Angles

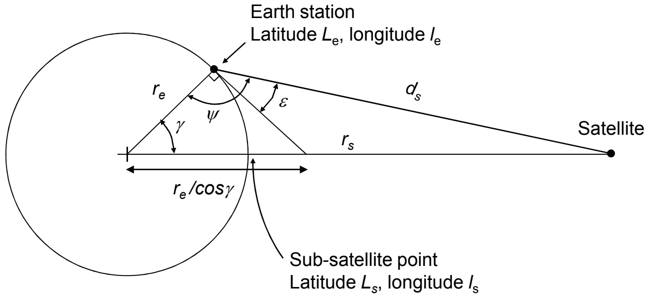


Look Angles

- The angle γ at the centre of the Earth is given by:

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

- where L_e is the latitude of the Earth station, L_s is the latitude of the satellite, and l_s–l_e is the difference in longitude between the Earth station and the sub-satellite point.



Look Angles

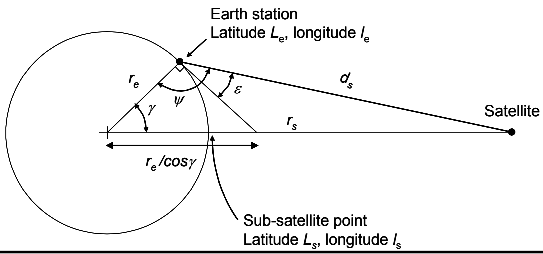
- In the case of a geostationary satellite, Ls =0, the equation simplifies to :

$$\cos\gamma = \cos L_e \cos(l_s - l_e)$$

- Applying the cosine rule ($A^2=B^2+C^2-2BC\cos a$):

$$d_s^2 = r_e^2 + r_s^2 - 2r_e r_s \cos\gamma$$

- where r_e is the radius of the Earth and r_s is the radius of the satellite (from the centre of the Earth).



Slant Range

- The slant range is therefore:

$$d_s = \left[r_e^2 + r_s^2 - 2r_e r_s \cos\gamma\right]^{1/2}$$

- Substituting for r_s and r_e gives:

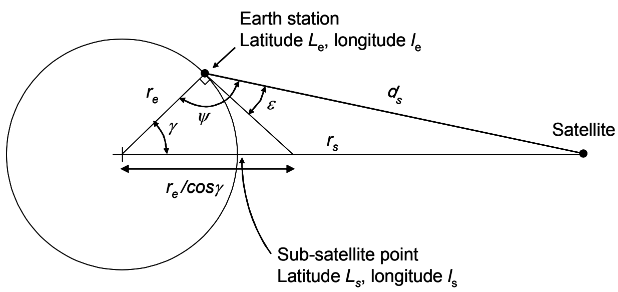
$$d_s = \left[18.1848178 - 5.3784398\cos\gamma\right]^{1/2} \times 10^4 \quad \text{(km)}$$

- The distance from the Earth station to the satellite is 36 786 km in zenith (90° elevation) and increases to a maximum slant range of 41 679 km at 0° elevation.

Elevation Angle

- From the sine rule $\left(\frac{\sin a}{A} = \frac{\sin b}{B} = \frac{\sin c}{C}\right)$:

$$\frac{r_s}{\sin\psi} = \frac{d_s}{\sin\gamma}$$



- giving:

$$\sin\psi = \frac{r_s \sin\gamma}{d_s}$$

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

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

Elevation Angle

- And, since, $\varepsilon = \psi - 90$,

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

- Using r_e=6 378 km and r_s=42 164 km gives:

$$\cos\varepsilon = \frac{\sin\gamma}{\left[1.02288 - 0.30253\cos\gamma\right]^{1/2}}$$

Elevation Angle

- The angle of elevation depends on the angle γ, which depends on the latitude of the Earth station. The elevation angle is therefore large towards the equator and small towards the poles.

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

- For the Earth station to be able to see the satellite, the elevation angle must be positive so that:

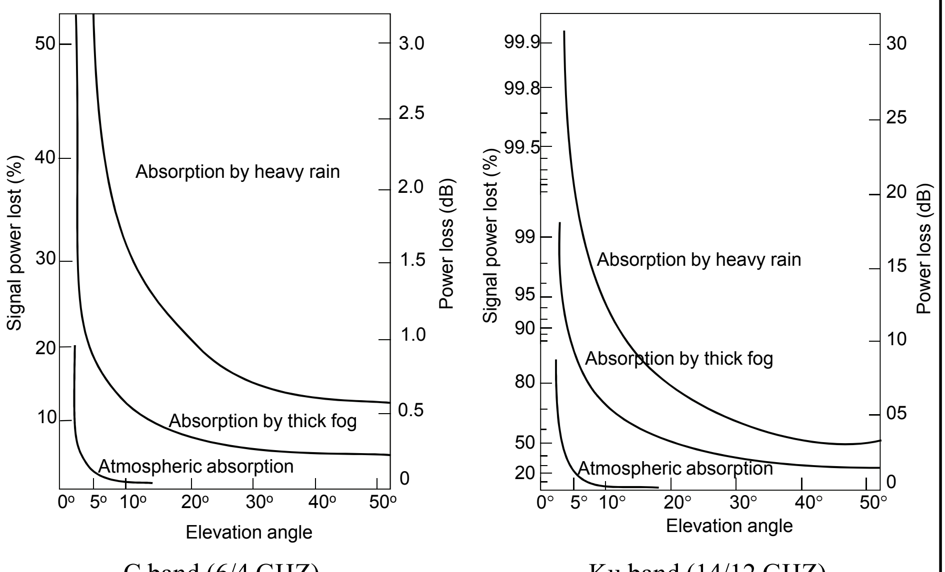
$$r_s \geq \frac{r_e}{\cos\gamma} \quad \Rightarrow \quad \gamma \leq \cos^{-1}\left(\frac{r_e}{r_s}\right)$$

- For a satellite in geostationary orbit, $\gamma \leq$ 81.3°. That is, a geostationary satellite is not visible above 81.3°N and 81.3°S.

Elevation Angle

- At small angles of elevation, the line-of-sight to the satellite is most likely occluded by foliage, buildings, small variations in terrain, and so on.
- Additionally, the receiver's sensitivity is reduced by the increase in thermal noise due to the close proximity of the Earth.
- Finally, the smaller the angle of elevation, the greater the distance a propagated wave must travel through the atmosphere with greater absorption and noise.
- Consequently, there is a restriction on the minimum angle of elevation as $\varepsilon \rightarrow 0$—generally, considered to be 5° in C band and 10° in Ku band.

Elevation Angle



C band (6/4 GHZ) Ku band (14/12 GHZ)

W. Tomasi, *Electronic Communications Systems*, Prentice Hall, Englewood Cliffs, New Jersey, 1994.

Limits of Visibility

- At a minimum angle of elevation, ε_{min}, the limits of visibility for an Earth station at longitude l_e, are:

$$l_e - l_s = \cos^{-1}\left(\frac{\cos\left(\cos^{-1}\left(\frac{r_e}{r_s}\cos\varepsilon_{\min}\right)-\varepsilon_{\min}\right)}{\cos L_s}\right)$$

- For a geostationary satellite (L_s=0) and ε_{min}= 0, and the equation reduces to the one we saw before:

$$r_s \geq \frac{r_e}{\cos\gamma} \quad \Rightarrow \quad \gamma \leq \cos^{-1}\left(\frac{r_e}{r_s}\right)$$

Area of Coverage

$$\frac{\cos\varepsilon}{r_s} = \frac{\cos(\varepsilon+\gamma)}{r_e} \quad \Longrightarrow \quad \gamma = \cos^{-1}\left(\frac{r_e\cos\varepsilon}{r_s}\right)-\varepsilon$$

- The area of the Earth's surface visible from a satellite is:

$$A_{coverage} = 2\pi r_e^2(1-\cos\gamma)$$

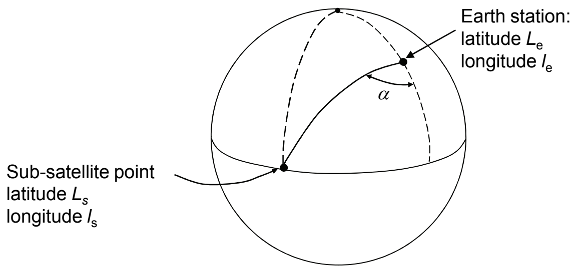
- Rearranging as a percentage of the Earth's surface:

$$A_{\%} = \frac{A_{coverage}}{4\pi r_e^2}\times 100\% = 50(1-\cos\gamma)\%$$

- So, from geostationary height, with a minimum elevation angle of 0°, γ=81.3° and the satellite can see 42.44% of the Earth's surface. From a LEO height of 780 km, with the same minimum elevation, the satellite can see 5.45% of the Earth's surface.

Azimuth

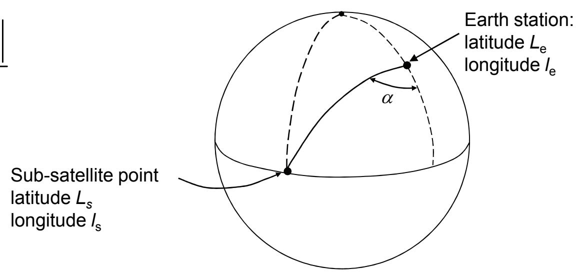
- The azimuth angle from the Earth station to the satellite is the same as the azimuth angle from the Earth station to the sub-satellite point (because the Earth station, the centre of the Earth, the satellite, and the sub-satellite point are all in the same plane). The azimuth angle is much more difficult to compute than the elevation because the geometry depends on whether the satellite is east or west of the Earth station, and in which of the hemispheres the Earth station and the sub-satellite point are located.



Azimuth

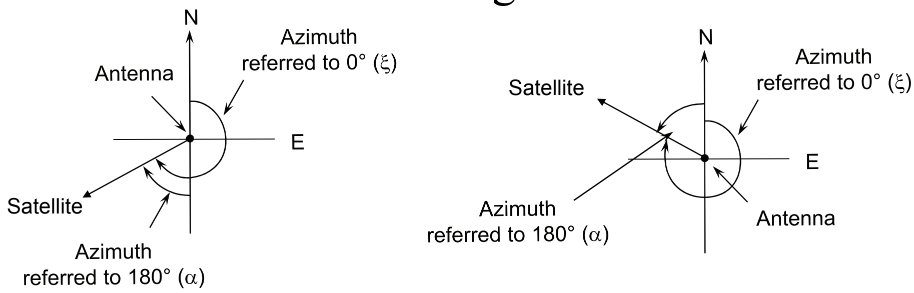
- The problem becomes much simpler for a geostationary satellite where the azimuth angle can be related to the latitude and longitude of the Earth station, L_e and l_e, and the longitude of the sub-satellite point l_s by:

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



- The azimuth can be referred to 0° (north), or to 180° (south). The azimuth is given the symbol ξ when referred to 0° and α when referred to 180°.

Look Angles



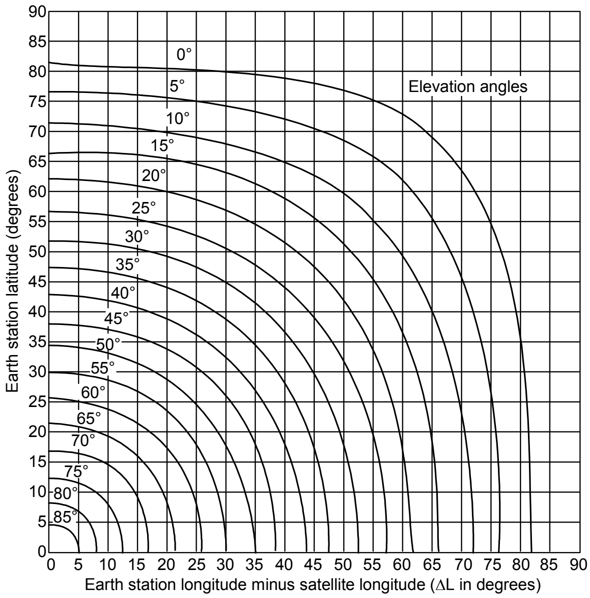
Northern hemisphere   Southern hemisphere

- In the northern hemisphere, the azimuth referred to 0° is:
  - $\xi = 180° + \alpha$, when the satellite is to the west of the Earth station
  - $\xi = 180° - \alpha$, when the satellite is to the east of the Earth station
- In the southern hemisphere, the azimuth referred to 0° is:
  - $\xi = 360° - \alpha$, when the satellite is to the west of the Earth station
  - $\xi = \alpha$, when the satellite is to the east of the Earth station

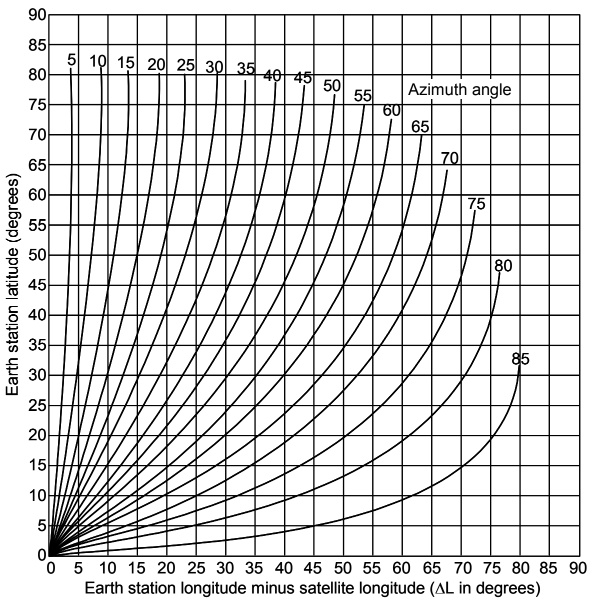
Look Angles

- A simpler approach to finding elevation and azimuth is to determine the latitude and longitude of the Earth station and establish the longitude of the satellite of interest.
- Calculate the difference, in degrees (ΔL), between the longitude of the satellite and the longitude of the Earth station.
- Then, from the following figure, determine the azimuth and elevation angle for the antenna.

Look Angles—Elevation (ε)



Look Angles—Azimuth (α)



Look Angles—Example

- An Earth station at Houston, Texas (longitude 95.5°W, latitude of 29.5°N) is to communicate to a satellite with longitude 135°W. Determine the azimuth and elevation.

  $\Delta L = 135° - 95.5° = 39.5°$

- From the figure, the angle of elevation is approximately 35° and the azimuth is approximately 59° west of south.

Look Angles—Example

- To check using the equations:
- Azimuth (referenced to 180°) can be calculated:

$$\alpha = \tan^{-1}\left(\frac{\tan\left|(l_s - l_e)\right|}{\sin L_e}\right) = \tan^{-1}\left(\frac{\tan 39.5°}{\sin 29.5°}\right) = \tan^{-1}\left(\frac{0.82}{0.49}\right) = 59°$$

- The angle at the centre of the Earth:

$$\gamma = \cos^{-1}\left[\cos L_e \cos(l_s - l_e)\right] = \cos^{-1}\left[\cos 29.5° \cos 39.5°\right]$$
$$= \cos^{-1}\left[0.87 \times 0.77\right] = 47.8°$$

Look Angles—Example

- So, the elevation is:

$$\varepsilon = \cos^{-1}\left[\frac{\sin\gamma}{\left[1.02288 - 0.30253\cos\gamma\right]^{1/2}}\right]$$
$$= \cos^{-1}\left[\frac{0.741}{\left[1.02288 - 0.30253 \times 0.67\right]^{1/2}}\right] = 35°$$

- The slant range to the satellite is:

$$d_s = \left[18.1848178 - 5.3784398\cos\gamma\right]^{1/2} \times 10^4 \quad \text{(km)}$$
$$= \left[18.1848178 - 5.3784398 \times 0.672\right]^{1/2} \times 10^4 \quad \text{(km)}$$
$$= \left[14.57050635\right]^{1/2} \times 10^4 \quad \text{(km)}$$
$$= 38\ 171\,\text{km}$$

Solar Eclipses

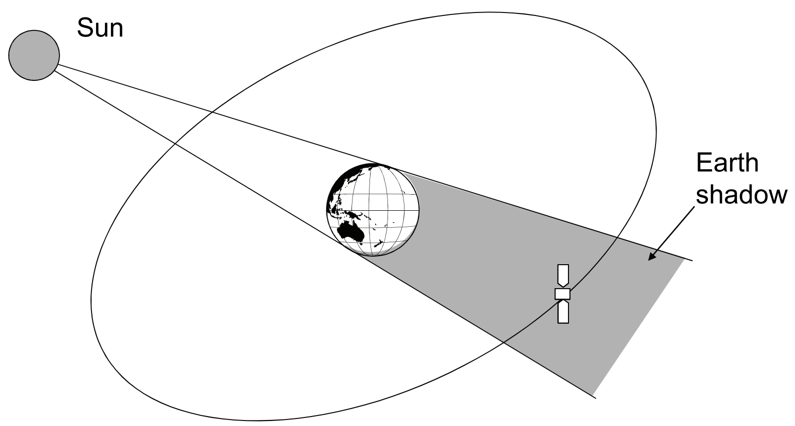
- Occasionally the path between a geostationary satellite and the Sun will be occluded by the Earth or Moon.
- Since a geostationary satellite uses solar cells for its primary power, any loss of the Sun's radiation requires the use of its battery back-up to provide a secondary power source during the period of the eclipse to ensure that service is not lost.
- Even so, there is a high cost associated with carrying sufficient batteries for full operations and it is therefore preferable to reduce the service during the eclipse period.
- This becomes particularly important at the end of a satellite's operational life when battery output power is reduced.

Solar Eclipses

- The rapid change in temperature during a solar eclipse places severe thermal stress on the spacecraft, whose design must accommodate the fact that Earth sensors may give false readings during an eclipse, which may confuse attitudinal control.
- The design must also protect against rapid power fluctuations at the start and end of an eclipse since system failures are much more likely to happen during these periods than during any other.

Eclipse due to Earth

- A satellite is said to be in eclipse when the Earth prevents sunlight from reaching it, that is, when the satellite is in the shadow of the Earth.

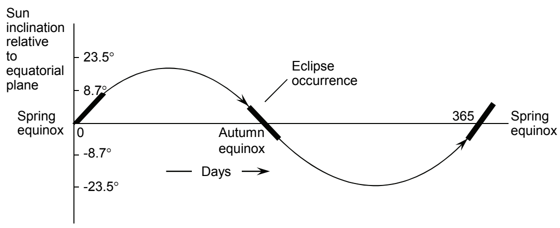


Eclipse due to Earth

- The Sun makes a sinusoidal motion relative to the equatorial plane with a period of one year and can be approximated by[1]:

$$\chi \approx 23 \sin\left(\frac{2\pi t_d}{T_Y}\right)$$

- With χ the Sun inclination, t_d the time in days, and T_Y = 365 days.



Movement of the Sun relative to the equatorial plane[2]

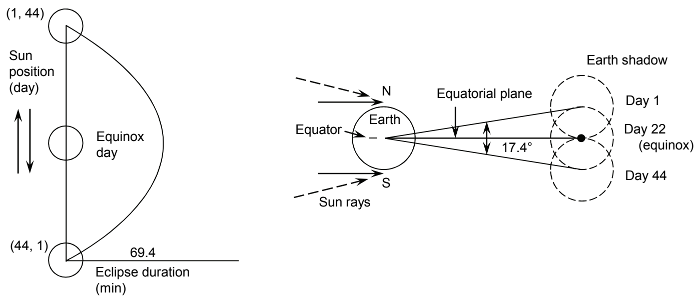
1. G. Maral and M. Bousquet, *Satellite Communications Systems*, John Wiley and Sons, Chichester, 1986.
2. M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

Eclipse due to Earth

- For geostationary satellites, eclipses occur during autumnal and vernal equinox (about 21 March and about 23 September).
- They occur near the equinoxes because these are the times when the Sun, the Earth, and the spacecraft are nearly in the same plane.
- The eclipse duration is determined by the time it takes the satellite to traverse through the 17.4° of shadowed arc (~69.4 minutes).

Eclipse due to Earth

- Substituting 17.2/2 into the previous equation shows that eclipses occur of approximately 22 days before and after equinox.

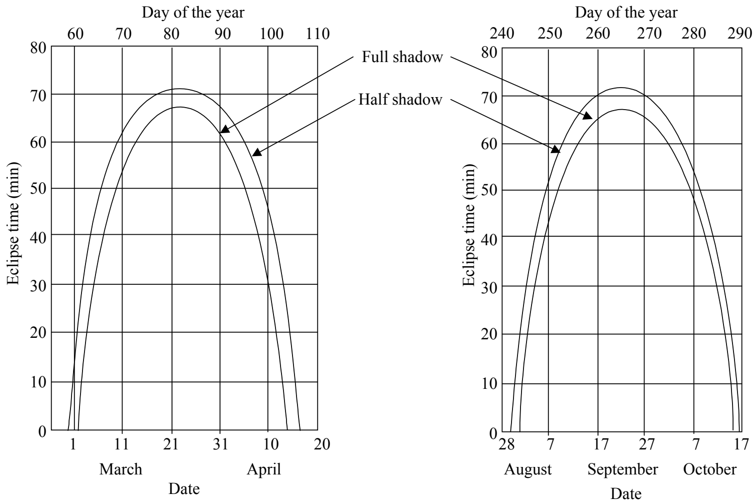


Earth-induced eclipse of a geostationary satellite[1]

1. M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

Eclipse due to Earth

- Because of diffraction effects and the Sun is not a point source, eclipse duration exceeds theoretical by about 2 minutes.



Eclipse due to Earth

- Since the timing of a solar eclipse predictable, orbital locations can be chosen so that the eclipses occur during off-peak traffic loads. The occurrence of the eclipse can be suitably timed by selecting the location of the satellite to the west of the coverage area according to[1]:

$$T_l = 23.38 - (1/15)(\phi_s - \phi_{tz})$$

- Where T_l is local time when eclipse begins, ϕ_s is eastern limit of orbital arc, and ϕ_{tz} is longitude of the time zone of the coverage area.
- Satellites in orbits other than geostationary orbit also experience eclipse more frequently depending on the nature of their orbit.
- For many LEO, the satellite is eclipsed daily.

1. M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

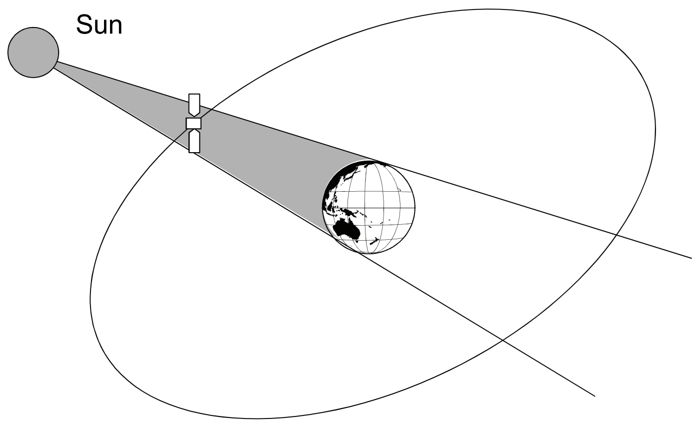
Eclipse due to Moon

- Eclipse can also occur when the satellite is in shadow because the Moon blocks the Sun's rays.
- For geostationary satellites, lunar eclipses occur twice a year for about forty minutes.
- Eclipses in other orbits are shorter but occur more frequently.
- Lunar orbits are a little more unpredictable than solar eclipses and orbital slots cannot be chosen to accommodate the Moon's obstruction.
- Battery capacity is particularly stressed when a lunar and solar eclipse happen in close proximity as the battery may not have sufficient time to re-charge.

Sun-transit Outages

- The Sun radiates a broad spectrum of RF energy that will appear as noise at the front end of communications receivers.
- For most terrestrial communication systems this solar noise is generally insignificant since the power levels are low in comparison to the signal strength, the solar noise increases with frequency, and the receive antenna beamwidth is not pointed directly at the Sun.
- In satellite communications systems, however, the received signal is very weak, the frequencies used tend to be higher than terrestrial systems, and the high-gain sensitive receiver is pointed skyward.
- Particularly strong interference occurs when the geometry of an orbit is such that the beam of an Earth station antenna is pointed directly at the Sun. For example, the solar noise interference at 4 GHz can be 20 dB in excess of the signal level for a satellite broadcast TV signal.

Sun-transit Outages



Sun-transit Outages

- The increase in antenna noise temperature, ΔT_a, depends on the polarisation of the solar noise, the Sun's equivalent noise temperature, the antenna beamwidth and efficiency.

$$\Delta T_a = pT_s\eta D_s^2$$

- Where:
  - p = 0.5 (a factor to account for the ransom polarisation of the noise),
  - T_s is the Sun's equivalent noise temperature = 120 000 $f^{-0.75}$ (where f is frequency of operation in GHz)
  - D_s is the optical diameter of the Sun/θ_{hp} (where θ_{hp} is the half-power beamwidth of the antenna)
    - D_s=0.48/ θ_{hp} (for θ_{hp}>0.48) and D_s=1 (for θ_{hp}<0.48)

1. M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

Sun-transit Outages

- For an antenna of beamwidth θ_b, solar interference occurs for a maximum number of days of:

$$\frac{(\theta_b + 0.48)}{0.4}$$

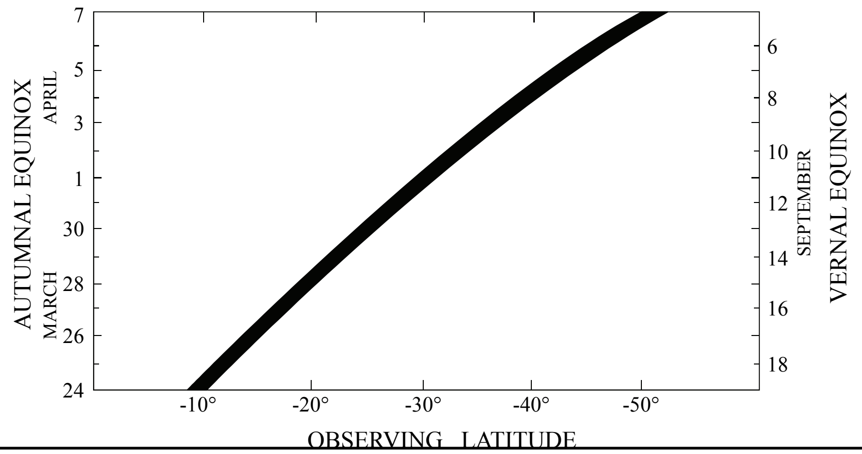
- The maximum duration of the Sun transit on the peak day is:

$$\frac{(\theta_b + 0.48)}{0.25}$$

M. Richharia, *Satellite Communication Systems*, McGraw Hill, New York, 1995.

Sun-transit Outages

- Depending on the system parameters, disturbance may be such that it results in a *Sun-transit outage*. The graph below indicates the days on which maximum solar interference (a sun-transit outage) will occur.

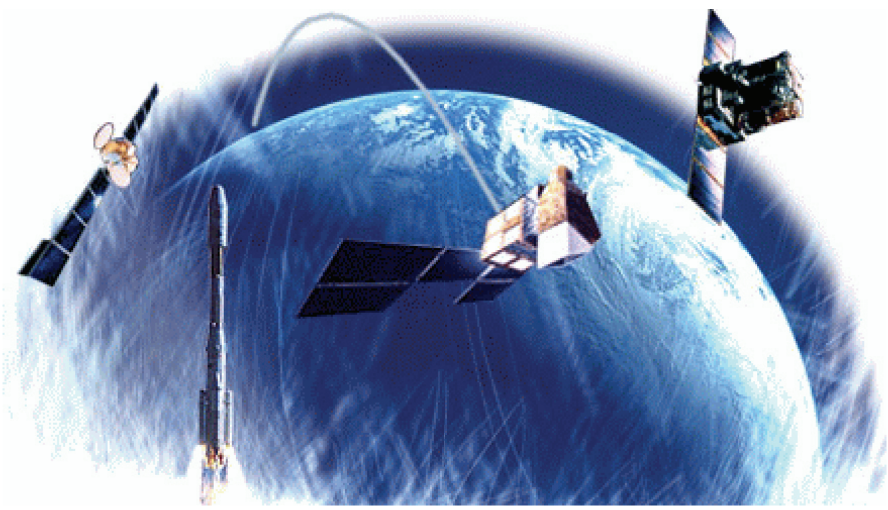


Sun-transit Outages

- The days on which maximum solar interference occurs varies according to the latitude of the Earth station; the time of day on which the interference occurs depends on the longitude of the Earth station.
- Noticeable interference may be experienced for as long as ten minutes a day for several days, during which time the receiving Earth station can do nothing except wait for the Sun to move out of the antenna’s main lobe.
- Although the time lost may seem insignificant (approximately 0.01% of the year), Sun-transit outages always occur in the daytime during peak traffic loads.

Selection of Orbit

- The selection of an orbital location depends on the application and the associated radio regulations, the desired coverage area and the orbital crowding in the region of interest. Some of the basic points considered are:
  - The service area should be served at as high an elevation angle as possible. This applies in particular to MSS, for reasons we will discuss later.
  - Satellite eclipse should occur as late at night as possible to minimise the need for storage batteries on the satellite.
  - The need to maintain a separation of 2–3° from the adjacent satellite to permit coexistence and to co-ordinate with other agencies who might be affected or might have an existing satellite.
  - Some services such as DBS have an existing plan that reserves an orbital slot to each country.



SATELLITE ORBITS