

Chapter 2

Space Segment

This chapter describes orbital mechanics and their significance with regard to satellite use for Mobile Satellite Communications (MSC). The space segment of an artificial satellite system is one of its three major operational components, the others being the ground and user segments. It comprises the different satellite constellations and the satellite uplink and downlink. Namely, here are introduced the fundamental laws governing satellite orbits and the principal parameters that describe the motion of the Earth's artificial satellites. The types of satellite orbits are also classified, presented, and compared from the MSC system viewpoint in terms of coverage and link performances. The satellite bus and payloads for communications, broadcasting, and navigation (GNSS) are discussed. The chapter concludes with a brief overview of satellite launch vehicles and orbit insertion. Types of satellite orbits and perturbations are also classified and compared from the communication and navigation systems viewpoint in terms of coverage and link performances.

During the last four decades, military and commercial MSC networks have utilized GEO extensively to the point where orbital portions have become quite crowded and coordination between satellites is becoming constrained and could never solve the problem of polar coverage. On the other hand, non-GEO MSC solutions have recently grown in importance because of their orbit characteristics and coverage capabilities in high latitudes and polar regions.

2.1 Platforms and Orbital Mechanics

The platform is an artificial object located in orbit around the Earth at a minimum altitude of about 20 km in the stratosphere and a maximum distance of about 36,000 km in the Space. The artificial platforms can have a different shape and designation but usually they have the form of aircraft, airship, or spacecraft. In addition, there are special space stations and space ships, which are serving on more

distant locations from the Earth's surface for scientific exploration and research and for cosmic expeditions.

Orbital mechanics is a specific discipline describing planetary and satellite motion in the solar system, which can solve the problems of calculating and determining the position, speed, path, perturbation, and other orbital parameters of planets and satellites. In fact, a space platform is defined as an unattended object revolving about a larger one. Although it was used to denote a planet's Moon, since 1957 it also means a man-made object put into orbit around a large body (planet), when the former USSR launched its first spacecraft Sputnik-1. Accordingly, man-made satellites are sometimes called artificial satellites.

Orbital mechanics support a communications satellites project in the phases of orbital design and operations. The orbital design is based on a generic survey of orbits and at an early stage in the MSC project is tasked to identify the most suitable orbit for the objective MSC service. The orbital operation is based on rather short-term knowledge of the orbital motion of the satellite and starts with TT&C maintenances after the satellite is located in orbit. In effect, only a few types of satellite orbits are well suited for MSC and navigation systems.

2.1.1 Space Environment

The satellite service begins when a spacecraft is located as a space platform in the desired orbital position in a space environment around the Earth. This space environment is a very specific part of the Universe, where many factors and determined elements affect the planet and satellite motions. The Earth is surrounded by a thick layer of many different gasses and parts known as the atmosphere, whose density decreases as the altitude increases. Hence, there is no air and the atmosphere disappears at about 180 km above the Earth, where the Cosmos begins. The endless environment in space is not very friendly and is extremely destructive, mainly because there is no atmosphere, the cosmic radiation is very powerful, the vacuum creates very high pressure on spacecraft or other bodies, and there is the negative influence of very low temperatures.

The Earth's gravity keeps everything on its surface. All the heavenly bodies such as the Sun, Moon, planets, and stars have gravity and reciprocal reactions. Any object flying in the atmosphere continues to travel until it meets forces due to the Earth's gravity or until it has enough speed to surpass gravity and to hover in the stratosphere. However, to send an object into space, it first has to overcome gravity and then travel at least at a particular minimum speed to stay in space. In this case, an object traveling at about 5 miles/s can circle around the Earth and become an artificial spacecraft.

An enormous amount of energy is necessary to put a satellite into orbit and this is realized by using a powerful rockets or launchers, which are defined as an apparatus consisting of a case containing a propellant (fuel) and reagents by the combustion of which it is projected into the space. As the payload is carried on the

top, the rocket is usually separated and drops each stage after burnout and brings a payload up to the required velocity and leaves it in orbit. A rocket is also known as a booster, as a rocket starts with a low velocity and attains some required height, where air drag decreases and it attains a higher velocity.

2.1.2 Laws of Satellite Motion

A satellite is an artificial object located by rocket in space orbit following the same laws in its motion as the planets rotating around the Sun. Thus, Johannes Kepler, a German mathematician, has contributed a great deal to the field of astronomy and astrology. The Laws of Planetary Motion formulated by Kepler proves that the orbits of the planets are ellipses and not circles, as believed by many. The ellipse is a geometrical shape that has two foci, such that the sum of the distance from the focus to any point on the surface of the ellipse is constant. The orbits of planets have small eccentricities (flattening of ellipse), and so, they appear as circles. Based on the properties of ellipses, Johannes Kepler devised three laws that explain the motion of planets around the Sun.

A satellite is an artificial object launched and located by rocket in orbit follows the same laws in its motion as the planets rotating around the Sun. Thus, three important laws for planetary motion were derived by Johannes Kepler, as follows:

1. **First Law**—The first law is also known as The Law of Orbits. As stated, an ellipse has two foci. While studying the motion of planets around the Sun, Kepler explained that the path followed was elliptical, with the Sun as one of the two foci. In simple terms, the law is stated as: The orbit of each planet follows an elliptical path or all planets move in elliptical orbits, with the Sun at one focus, shown in Fig. 2.1a. This indicates that the Sun is one focus, while the other focus is known as the vacant or empty focus.

As shown in the diagram, the Sun and the empty focus lie on the major axis of the ellipse, and the planet lies on the surface of the ellipse. As the planet is continuously moving around the Sun, and as the Sun is not at the center of the ellipse, the Planet–Sun distance will always keep on changing. The Law of Orbits proves that planet motion lies in the plane around the Sun (1602).

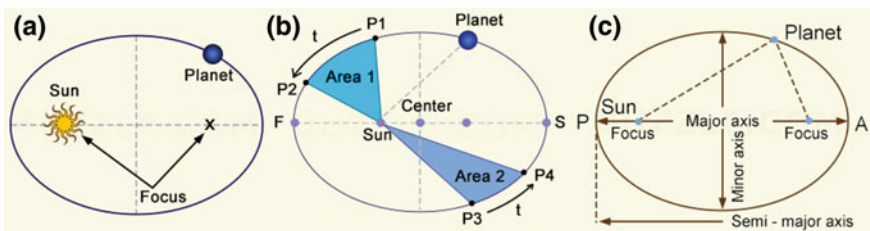


Fig. 2.1 Kepler's laws of satellite motion—Courtesy of Manual: by Ilcev

2. Second Law—The second law is also known as The Law of Equal Areas, shown in Fig. 2.1b. As the Sun is one of the foci, it is clear that the Planet–Sun distance will be changing. But, the planet covers up for the increase in the distance by moving faster when it is closer to the Sun. This indicates that planets do not move at a uniform speed. This law states that: The line from the Sun to orbital planet or radius vector (r) sweeps out equal areas in equal intervals of time (t) as the planet travels around the ellipse. The point at which the speed of the planet is fastest is known as Perihelion or Perigee indicated with F (Fastest motion), while the distance with slowest speed is known as Aphelion (Apogee) indicated by S (Slowest motion). The distance measured from the Perihelion to the position of the Sun is known as Perihelion distance, while the distance from the Sun to the Aphelion is known as the Aphelion distance. The law says that while moving in an elliptical path, the planet moves faster when it is closer to the Sun. This way, the radius sweeps equal areas in equal amount of time. If the planet is observed at successive times (P1, P2, P3, P4), it draws the radius vector during the first second observations, showing that the two radius vectors having the same area. So, the area swept during the time (t) by the planet to move from P1 to P2 is the same as the area swept while moving from P3 to P4. This is the Law of Equal Areas (1605).

3. Third Law—The third law of planetary motion in ellipse with Perigee (P) and Apogee (A) is alternatively known as The Law of Periods and Harmonic Law, see Fig. 2.1c. This law relates the time required by a planet to make a complete trip around the Sun to its mean distance from the Sun. It can be simply stated as: The square of the planet orbital period is directly proportional to the cube of the semi-major axis of its orbit. The square of the planet's orbital period around the Sun (T) is proportional to the cube of the semi-major axis (a = distance from the Sun) of the ellipse for all planets in the Solar system (1618).

Kepler's laws only describe the planetary motion if the mass of central body insofar as it is considered to be concentrated in its center and when its orbits are not affected by other systems. However, these conditions are not completely fulfilled in the case of Earth's motion and its artificial satellites. Namely, the Earth does not have an ideal spherical shape and the different layers of mass are not equally concentrated inside of the Earth's body. Because of this, the satellite motions are not ideally synchronized and stable, and the motions are namely slower or faster at particular orbital sectors, which present certain exceptions to the rule of Kepler's laws.

Furthermore, in distinction from natural satellites, whose orbits are almost elliptical, the artificial satellites can also have circular orbits, for which the basic relation can be obtained by the equalizing the centrifugal and centripetal Earth forces.

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Kepler's laws were based on observational records and only described the planetary motion without attempting an additional theoretical or mathematical explanation of why the motion takes place in that manner.

In 1687, the English physicist British Sir Isaac Newton published his breakthrough work "Principia Mathematica" with own syntheses, known as the Three Laws of Motion, such as follows:

1. **Law I**—Every body continues in its state of rest or uniform motion in a straight line, unless it is compelled to change that state by forces impressed on it.
2. **Law II**—The change of momentum per unit time of a body is proportional to the force impressed on it and is in the same direction as that force.
3. **Law III**—To every action, there is always an equal and opposite reaction.

On the basis of Law II, Newton also formulated the Law of Universal Gravitation, which states that any two bodies attract one another with a force proportional to the products of their masses and inversely proportional to the square of the distance between them. This law may be expressed mathematically for a circular orbit with the relations:

$$F = m(2\pi/t)^2(R + h) = G \left[M \cdot m / (R + h)^2 \right] \quad (2.1)$$

where parameter m = mass of the satellite body; t = time of satellite orbit; R = equatorial radius of the Earth (6.37816×10^6 m); h = altitude of satellite above the Earth's surface; G = Universal gravitational constant (6.67×10^{-11} N m²/kg⁻²); M = Mass of the Earth body (5.976032×10^{24} kg) and finally, F = force of mass (m) due to mass (M). Force of mass can be also presented by the following relation:

$$F = ma = dv/dt \quad (2.2)$$

where a = acceleration and v = velocity of satellite orbit.

The force of attraction between two distant point masses m_1 and m_2 separated by a distance r is giving the following relation:

$$F = Gm_1m_2/r^2 \quad (2.3)$$

where G = Newtonian (or universal) gravitation constant.

Consider the simple circular orbit and assuming that the Earth is a sphere, it is possible that can be treated a point mass. The centripetal force F_c required to keep the satellite in a circular orbit $= mv^2/r$, where v = orbital velocity of the satellite.

The force of gravity that supplies this centripetal force is GMm/r^2 , where M = mass of the Earth, and m is the mass of the satellite. Equating the two forces gives relation:

$$F_c = mv^2/r = GMm/r^2 \quad (2.4)$$

Division by m eliminates the mass of the satellite from the equation, which means that the orbit of a satellite is independent of its mass. Thus, the period of the satellite is the orbit circumference divided by the velocity: $T = 2\pi r/v$. Substituting in Eq. 3.3 gives the following relation:

$$T^2 = (4\pi^2/GM)r^3 \quad (2.5)$$

The first-generation NOAA meteorological satellites orbit at approximately 850 km above the Earth's surface. Since the equatorial radius of the Earth is about 6378 km, the orbit radius is about 7228 km. Substituting in Eq. 3.4 shows that the NOAA satellites have a period of about 102 min.

However, radius required for a satellite in GEO has the same angular velocity as the Earth, so the angular velocity mean motion constant of a satellite shows the following:

$$\xi = 2\pi/T \quad (2.6)$$

Substituting Eq. 2.6 in Eq. 2.5 is giving the following formula:

$$r^3 = GM/\xi^2 \quad (2.7)$$

Inserting the angular velocity of the Earth, the required radius for a GEO is 42,164 km or about 35,786 km above the Earth's surface.

2.1.2.1 Geometry of Elliptical Orbit

The satellite in circular orbit undergoes its revolution at a fixed altitude and with fixed velocity, while a satellite in an elliptical orbit can drastically vary its altitude and velocity during one revolution. The elliptical orbit is also subjected to Kepler's Three Laws of satellite motion.

Therefore, the characteristics of elliptical orbit can be determined from elements of the ellipse of the satellite plane with the perigee (Π) and apogee (A) and its position in relation to the Earth, see Fig. 2.2 (Left). The parameters of elliptical orbit are presented as follows:

$$\begin{aligned} e &= c/a = \sqrt{1 - (b/a)^2} \quad \text{or} \quad e = (\sqrt{a^2 - b^2}/a) \quad p = a(1 - e^2) \quad \text{or} \quad p = b^2/a \\ c &= \sqrt{a^2 - b^2} \quad a = p/1 - e^2 \quad b = a\sqrt{1 - e^2} \end{aligned} \quad (2.8)$$

where e = eccentricity, which determines the type of conical section; a = large semi-major axis of elliptical orbit; b = small semi-major axis of elliptical orbit;

c = axis between center of the Earth and center of ellipse, and p = focal parameter. The equation of ellipse derived from polar coordinates can be presented with the resulting trajectory equation as follows:

$$r = p / (1 + e \cos \theta) \quad (2.9)$$

where r = distance of the satellites from the center of the Earth ($r = R + h$) or radius of path; Θ = true anomaly and E = eccentric anomaly. In this case, the position of the satellite will be determined by the angle called “the true anomaly,” which can be counted positively in the direction of movement of the satellite from 0° to 360° , between the direction of the perigee and the direction of the satellite (S).

The position of the satellite can also be defined by eccentric anomaly, which is the argument of the image in the mapping, which transforms the elliptical trajectory into its principal circle, an angle counted positively in the direction of movement of the satellite from 0 to 360° , between the direction of the perigee and the direction of the satellite. The relations for both mentioned anomalies are given by the following equations:

$$\cos \theta = \cos E - e / (1 - e \cos E) \quad \cos E = \cos \theta + e / (1 + e \cos \theta) \quad (2.10)$$

The total mechanical energy of a satellite in elliptical orbit is constant, although there is an interchange between the potential and the kinetic energies. As a result, a satellite slows down when it moves up and gains speed as it loses height. Thus, considering the termed gravitation parameter $\mu = GM$ (Kepler’s Constant $\mu = 3.99 \times 10^5 \text{ km}^3/\text{s}^2$), the velocity of a satellite in an elliptical orbit may be obtained from the following relation:

$$v = \sqrt{GM(2/r) - (1/a)} = \sqrt{\mu(2/r) - (1/a)} \quad (2.11)$$

Applying Kepler’s Third Law, the sidereal time of one revolution of the satellite in elliptical orbit is as follows:

$$\begin{aligned} t &= 2\pi\sqrt{(a^3/GM)} = 2\pi\sqrt{(a^3/\mu)} \\ t &= 3.147099647\sqrt{(26,628.16 \cdot 10^3)^3 \cdot 10^{-7}} = 43,243.64 \text{ [s]} \end{aligned} \quad (2.12)$$

Therefore, the last equation is the calculated period of sidereal day for the elliptical orbit of Russian-based satellite Molniya with apogee = 40,000 km, perigee = 500 km, revolution time = 719 min and $a = 0.5 (40,000 + 500 + 2 \times 6378.16) = 26628.15 \text{ km}$.

2.1.2.2 Geometry of Circular Orbit

The circular orbit is a special case of elliptical orbit, which is formed from the relations $a = b = r$ and $e = 0$, see Fig. 2.2 (Right). According to Kepler’s Third

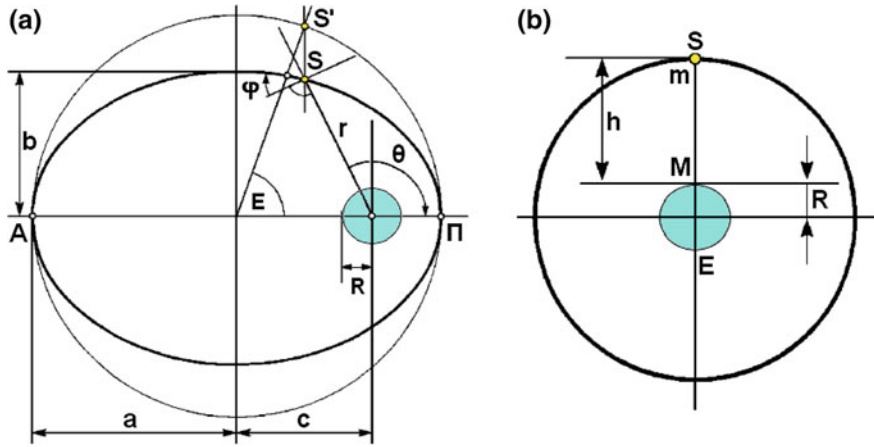


Fig. 2.2 Elliptical and circular satellite orbits—Courtesy of Book: by Galic

Law, the solar time (τ) in relation to the right ascension of an ascending node angle (Ω), the sidereal time (t) with the consideration that $\mu = GM$, and satellite altitude (h), for a satellite in circular orbit, will have the following relations:

$$\begin{aligned}\tau &= t / (1 - \Omega t / 2\pi) \\ t &= 2\pi \sqrt{(r^3 / \mu)} = 3.147099647 \sqrt{(r^3 \cdot 10^{-7})} \quad [\text{s}] \\ h &= [\sqrt[3]{(\mu t^2 / 4\pi^2)}] - R = 2.1613562 \cdot 10^4 (\sqrt[3]{t^2}) - 6.37816 \cdot 10^6 [\text{m}]\end{aligned} \quad (2.13)$$

The time is measured with reference to the Sun by solar and sidereal day. Thus, a solar day is defined as the time between the successive passages of the Sun over a local meridian. In fact, a solar day is a little bit longer than a sidereal day, because the Earth revolves by more than 360° for successive passages of the Sun over a point 0.986° further. On the other hand, a sidereal day is the time required for the Earth to rotate one circle of 360° around its axis: $t_E = 23 \text{ h } 56 \text{ min } 4.09 \text{ s}$. Therefore, a geostationary satellite must have an orbital period of one sidereal day in order to appear stationary to an observer on Earth. During rotation, the duration of sidereal day $t = 85,164,091 \text{ (s)}$ and is considered in such a way for synchronous orbit that $h = 35786.04 \times 10^3 \text{ (m)}$. The speed is conversely proportional to the radius of the path ($R + h$) and for the satellite in circular orbit, it can be calculated from the following relation:

$$v = \sqrt{(MG / R + h)} = \sqrt{(\mu / r)} = 1.996502 \cdot 10^{-7} / \sqrt{r} = 631.65 \sqrt{r} [\text{m/s}] \quad (2.14)$$

From Eq. (2.8) and using the duration of sidereal day (t_E) gives the relation for the radius of synchronous or geostationary orbits:

Table 2.1 The values of times different than the synchronous time of orbit

Parameter	Values of time					Unit
t	86164.00	43082.05	21541.23	10770.61	6052.00	s
h	35786.00	20183.62	10354.71	4162.89	800.00	km
(R + h)	42164.00	26561.78	16732.87	10541.05	7178.00	km
v	3075.00	3873.83	4880.72	5584.12	7450.00	km/s ⁻¹

$$r = \sqrt[3]{(\mu t)/(2\pi)^2} \quad (2.15)$$

The satellite trajectory can have any angle of orbital planes in relation to the equatorial plane: in the range from PEO up to GEO plane. Namely, if the satellite is rotating in the same direction of Earth's motion, where (t_E) is the period of the Earth's orbit, the apparent orbiting time (t_a) is calculated by the following relation:

$$t_a = t_E \cdot t/t_E - t \quad (2.16)$$

This means, inasmuch as $t = t_E$ the satellite is geostationary ($t_a = \infty$ or $\tau = 0$). In Table 2.1, several values for times different from synchronous orbital time are presented.

According to Table 2.1 and Eq. (2.9), it is evident that a satellite does not depend so much on its mass but decreases with higher altitude. In addition, satellites in circular orbits with altitudes of a 1700, 10,400 and 36,000 km will have t/τ values 2/2, 18, 6/8, and 24/zero, respectively. In this case, it is evident that only a satellite constellation at altitudes of about 36,000 km can be synchronous or geostationary.

2.1.3 Horizon and Geographic Satellite Coordinates

The horizon system is a type of orbital coordinate parameters that can be used to locate the position of objects in the space. In a satellite, orbits usually use local geographic coordinates, which rotate with the Earth. The horizon and geographical coordinates are very important to find out many satellite parameters and equations for better understanding the problems of orbital plane, satellite distance, visibility of the satellite, coverage areas, etc. The coverage areas of a satellite are illustrated in Fig. 2.3a with the following geometrical parameters: actual altitude (h), radius of Earth (R), angle of elevation (ϵ), angle of azimuth (A), distance between satellite and the Earth's surface (d), and central angle (Ψ) or subsatellite angle, which is similar to the angle of antenna radiation (δ).

The geographical and horizon coordinates of a satellite are presented in Fig. 2.3b with the following, not yet mentioned, main parameters: angular speed of the Earth's rotation (v), argument of the perigee (ω), moment of satellite pass across any point on the orbit (t_0), which can be perigee (Π), projection of the perigee point

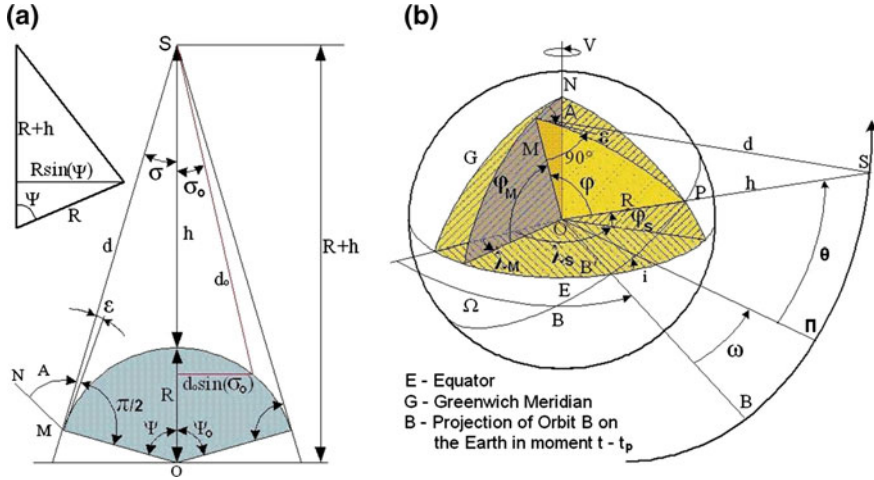


Fig. 2.3 Geometric projection of satellite orbits—Courtesy of Book: by Zhilin

on the Earth's surface (Π'), spherical triangle ($B'TP$), satellite (S), the Point of the Observer or Mobile (M), latitudes of observer and satellite (φ_M and φ_S), longitudes of observer and satellite (λ_M and λ_S), inclination angle (i) of the orbital plane measured between the equatorial and orbital plane, and the right ascension of an ascending node angle in the moment of t_0 (Ω_0).

Otherwise, the right ascension of an ascending node angle (Ω) is the angle in the equatorial plane measured counterclockwise from the direction of the vernal equinox to that of the ascending node, while the argument of the perigee (ω) is the angle between the direction of the ascending node and the direction of the perigee.

2.1.3.1 Satellite Distance and Coverage Area

The area coverage or angle of view for each type of satellite depends on orbital parameters, its position in relation to the Land Earth Station (LES), and geographic coordinates. This relation is very simple in the case where the subsatellite point is in the center of coverage, while all other samples are more complicated. Thus, the angle of a GEO satellite inside its range has the following regular reciprocal relation:

$$\delta + \varepsilon + \Psi = 90^\circ \quad (2.17)$$

The circular sector radius can be determined by the following relation:

$$R_s = R \sin \Psi \quad (2.18)$$

When the altitude of orbit h is the distance between satellite and subsatellite point (SP), the relation for the altitude of the circular sector can be written as:

$$h_s = R(1 - \cos \Psi) \quad (2.19)$$

From a satellite communications point of view, there are three key parameters associated with an orbiting satellite, such as (1) Coverage area or the portion of the Earth's surface that can receive the satellite's transmissions with an elevation angle larger than a prescribed minimum angle; (2) The slant range (actual LOS distance from a fixed point on the Earth to the satellite), and (3) The length of time a satellite is visible with a prescribed elevation angle.

Elevation angle is an important parameter, since communications can be significantly impaired if the satellite has to be viewed at a low-elevation angle, that is, an angle too close to the horizon line. In this case, a satellite close to synchronous orbit covers about 40 % of the Earth's surface. Thus, from the diagram in Fig. 2.3a a covered area expressed with central angle (2δ or 2Ψ) or with arc ($MP \approx R\Psi$) as a part of Earth's surface can be derived with the following relation:

$$C = \pi(R_s^2 + h_s^2) = 2\pi R^2(1 - \cos \Psi) \quad (2.20)$$

Since the Earth's total surface area is $4\pi R^2$, it is easy to rewrite C as a fraction of the Earth's total surface:

$$C/4\pi R^2 = 0.5(1 - \cos \Psi) \quad (2.21)$$

The slant range between a point on Earth and a satellite at altitude (h) and elevation angle can be defined in this way:

$$z = \left[(R \sin \epsilon)^2 + 2Rh + h^2 \right]^{1/2} - R \sin \epsilon \quad (2.22)$$

This determines the direct propagation length between LES, (h) and (ϵ) and will also find the total propagation power loss from LES to satellite. In addition, (z) establishes the propagation time (time delay) over the path, which will take an electromagnetic field as:

$$t_d = (3.33)z \quad [\mu \text{ sec}] \quad (2.23)$$

To propagate over a path of length (z) km, it takes about 100 ms to transmit to GEO. If the location of the satellite is uncertain ± 40 km, a time delay of about ± 133 μ sec is always present in the Earth-to-satellite propagation path. When the satellite is in orbit at altitude (h), it will pass over a point on Earth with an elevation angle (ϵ) for a time period:

$$t_p = (2\Psi/360)(t/1 \pm t/t_E) \quad (2.24)$$

The quotations for right ascension of the ascending node angle (Ω) and argument of the perigee (ω) are as follows:

$$\begin{aligned}\Omega &= 9,95(R/r)^{3.5} \cos i \quad \text{or} \quad \Omega = \Omega_0 + v(t-t_0) \\ \omega &= 4,97(R/a)^{3.5} \left[5 \cos^2 i - 1 / (1-e^2)^2 \right]\end{aligned}\quad (2.25)$$

The limit of the coverage area is defined by the elevation angle from LES above the horizon with angle of view $\varepsilon = 0^\circ$. In this case, the satellite is visible and its maximal central angle for GEO will be as follows:

$$\begin{aligned}\Psi &= \arccos(R \cos \varepsilon / r) - \varepsilon \quad \text{or} \\ \Psi &= \pi/2 - \arcsin(R/r) = \arccos(R/r) - \varepsilon = \arccos k - \varepsilon \\ \Psi &= \arccos 6,376.16/42,164.20 = \arccos 0.15126956 = 81^\circ 17' 58.18'' \\ C_{\max} &= 255.61 \cdot 10^6 (1 - 0.15126956) = 216.94 \cdot 10^6 (\text{km}^2)\end{aligned}\quad (2.26)$$

Therefore, all MES and LES with a position above $\Psi = 81^\circ$ will be not covered by GEO satellites. Since the Earth's square area is $510,100,933.5 \text{ km}^2$ and the extent of the equator is $40,076.6 \text{ km}$, only with three GEO mutually moved apart in the orbit by 120° it is possible to cover a great area of the Earth's surface, see Fig. 2.4a, which shows AOR (Atlantic), IOR (Indian), and POR (Pacific) satellite coverages. The zero angles of elevation have to be avoided, even to get maximum coverage, because this increases the noise temperature of the receiving antenna. Owing to this problem, an equation for the central angle with minimum angle of view between 5° and 30° will be calculated with:

$$\Psi_s = \arccos(k \cos \varepsilon) - \varepsilon \quad (2.27)$$

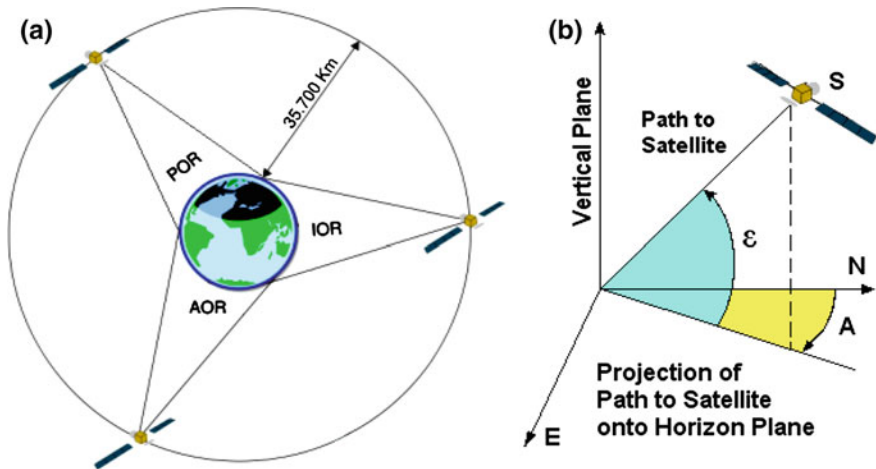


Fig. 2.4 GEO coverage and look angle parameters—Courtesy of Book: by Pratt

The arch length or the maximum distant point in the area of coverage can be determined in the following way:

$$l = 2\pi R(2\Psi/360 = 222.64\Psi \text{ [km]}) \quad (2.28)$$

The real altitude of satellite over subsatellite point is as follows:

$$h = r - R = 42,162 - 6,378 = 35,784 \text{ [km]} \quad (2.29)$$

The view angle under which a GEO satellite can see LES/MES is called the “subsatellite angle.” More distant points in the coverage area for GEO satellites are limited around $\varphi = 70^\circ$ of North and South geographical latitudes and around $\lambda = 70^\circ$ of East and West geographical longitudes, viewed from the subsatellite point. Theoretically, all Earth stations around these positions are able to see satellites by a minimum angle of elevation of $\varepsilon = 5^\circ$. Such access is very easy to calculate, using simple trigonometry relations:

$$\delta_{\varepsilon=0} = \arcsin k \approx 9^\circ \quad (2.30)$$

At any rate, the angle (Ψ) is in correlation with angle (δ), which can determine the aperture radiation beam. For example, the aperture radiation beam of satellite antenna for global coverage has a radiation beam of $2\delta = 17.3^\circ$. According to Fig. 2.3a, it will be easy to find out relations for GEO satellites as follows:

$$\operatorname{tg} \delta = k \sin \Psi / 1 - k \cos \Psi = 0.15126956 \sin \Psi / 1 - \cos \Psi / 1 - 0.15126956 \cos \Psi \quad (2.31)$$

$$\delta_s = 90^\circ - \Psi_s = 8^\circ 42' \cdot 1.82''$$

Differently to say, the width of the beam aperture ($2\delta_s$) is providing the maximum possible coverage for synchronous circular orbit. The distance of LES and MES with regard to the satellite can be calculated using Fig. 2.3a and Eqs. (2.13) and (2.22) by:

$$d = R \sin \Psi / \sin \delta = r \sin / \cos \varepsilon \quad (2.32)$$

The parameter (d) is quite important for transmitter power regulation of LES, which can be calculated by the following equation:

$$\begin{aligned} d &= \sqrt{(R+r)^2 - 2Rr \cos \Psi} \quad \text{or} \\ d &= h \sqrt{1 + 2(1/k)(R/h)^2 (1 - \cos \varphi \cos \Delta \lambda)} \quad \text{or} \\ d &= r [1 - (R \cos \varepsilon / r)^2]^{1/2} - R \sin \varepsilon \end{aligned} \quad (2.33)$$

Accordingly, when the position of any MES is near the equator in subsatellite point (P) or right under the GEO satellite, its distance is equal to the satellite altitude and takes out value for $d = H$ of 35,786 km. Thus, every MES will have a further position from (P) when the central angle exceeds $\Psi = 81^\circ$, when $d_{\max} = 41,643$ km.

2.1.3.2 Satellite Look Angles (Elevation and Azimuth)

The horizon coordinates are considered to determine satellite position in correlation with an Earth observer, LES, and MES terminals. These specific and important horizon coordinates are angles of satellite elevation and azimuth, illustrated in Figs. 2.3a, b and 2.4b, respectively.

The satellite elevation (ε) is the angle composed upward from the horizon to the vertical satellite direction on the vertical plane at the observer point. From point (M) shown in Fig. 2.3a, the look angle of ε value can be calculated by the following relation:

$$\operatorname{tg} \varepsilon = \cos \Psi - k / \sin \Psi \quad (2.34)$$

Figure 2.5a illustrates the Mercator chart of the First-Generation Inmarsat space segment, using three-ocean coverage areas with the projection of elevation angles and with one example of a plotted position of a hypothetical ship (may also be aircraft or any mobile). Thus, it can be concluded that Mobile Earth Station (MES) at designated position ($\varepsilon = 25^\circ$ for IOR and $\varepsilon = 16^\circ$ for AOR) has the possibility to use either GEO satellites over IOR or AOR to communicate with any LES inside the coverage areas of both satellites.

The satellite azimuth (A) is the angle measured eastward from the geographical North line to the projection of the satellite path on the horizontal plane at the observer point. This angle varies between 0 and 360° as a function of the relative positions of the satellite and the point is considered. The azimuth value of the satellite and subsatellite point looking from the point (M) or the hypothetical position of MES can be calculated as follows:

$$\operatorname{tg} A' = \operatorname{tg} \Delta \lambda_M - k / \sin \Psi \quad (2.35)$$

Otherwise, the azimuth value, looking from subsatellite point (P), can be calculated as:

$$\operatorname{tg} A = \sin \Delta \lambda / \operatorname{tg} \varphi \text{ or } \sin A = \cos \varphi \sin \Delta \lambda \operatorname{cosec} \Psi \quad (2.36)$$

Figure 2.5b illustrates the Mercator chart of First-Generation Inmarsat 3-satellite or ocean coverage areas with the projection of azimuth angles, with one example for the plotted position of a hypothetical ship ($\varepsilon = 47^\circ$ for IOR and $\varepsilon = 303^\circ$ for AOR).

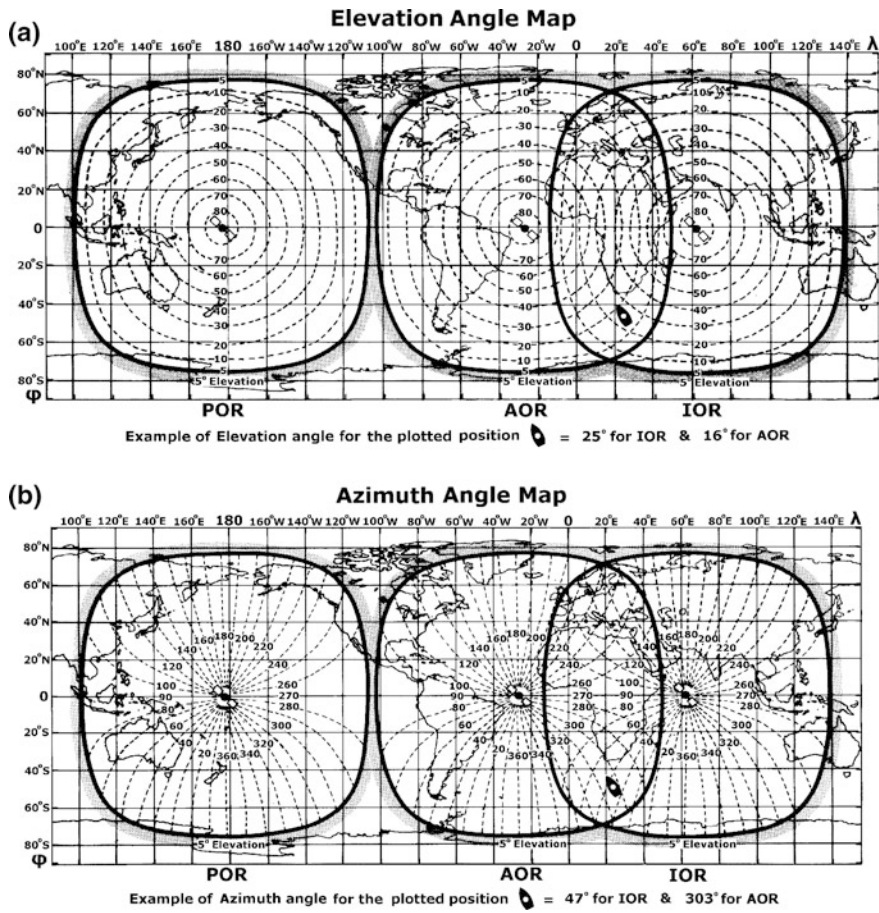


Fig. 2.5 Elevation and Azimuth angle maps—Courtesy of Manual: by EB Communications

Any mobile inside of both satellites’ coverage can establish a radio link to the subscribers onshore via any LES.

However, parameter (A') is the angle between the meridian plane of point (M) and the plane of a big circle crossing this point and subsatellite point (P), while the parameter (A) is the angle between a big circle and the meridian plane of point (P). Thus, the elevation and azimuth are, respectively, vertical or horizontal look angles, or angles of view, in which range the satellite can be seen.

Figure 2.6a presents a correlation of the look angle for three basic parameters (δ , Ψ , d) in relation to the altitude of the satellite. Inasmuch as the altitude of the satellite is increasing as the values of central angle (Ψ), distance between satellite and the Earth’s surface (d) and duration of communication (t_c) or time length of signals are increasing, while the value of subsatellite angle (δ) is indirectly proportional. An important increase of look angle and duration of communication can

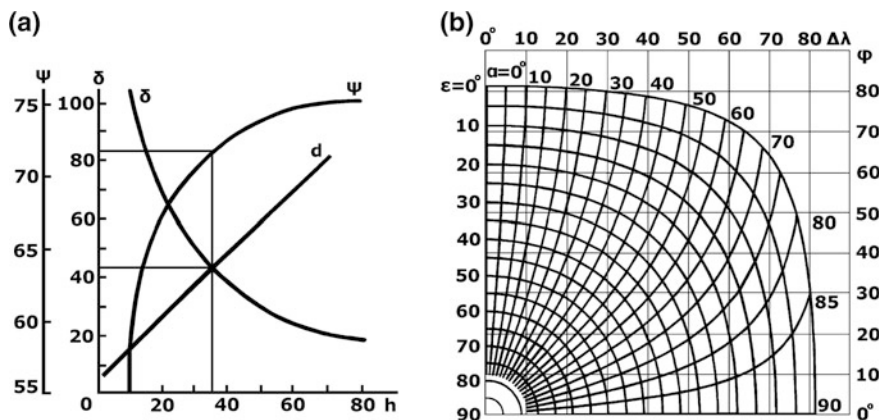


Fig. 2.6 Look angle parameters and graphic of geometric coordinates for GEO—Courtesy of Book: by Zhilin

be realized by increasing the altitude to 30 or 35,000 km, while an increase in look angle is unimportant for altitudes of more than 50,000 km. The duration of communication is affected by the direction's displacement from the center of look angle, which will have maximum value in the case when the direction is passing across the zenith of the LES. The single angle of the satellite in circular orbit depends on the $t/2$ value, which in area of satellite look angle, can be found in the duration of the time and is determined as:

$$t_c = \Psi t / \pi \quad (2.37)$$

Practical determination of the geometric parameters of a satellite is possible by using many kinds of plans, graphs, and tables. It is possible to use tables for positions of MES (ϕ , λ), by the aid of which longitudinal differences can be determined between MES and satellite for four feasible ship's positions: N/W, S/W, N/E, and S/E in relation to GEO.

One of the most important practical pieces of information about a communications satellite is whether it can be seen from a particular location on the Earth's surface. In Fig. 2.6b, a graphic design is shown which can approximately determine limited zones of satellite visibility from the Earth (MES) by using elevation and azimuth angles under the condition that $\delta = 0$. This graphic contains two groups of crossing curves, which are used to compare (ϕ) and ($\Delta\lambda$) coordinates of mobile positions. Thus, the first group of parallel concentric curves shows the geometric positions where elevation has the constant value ($\epsilon = 0$), while the second group of fan-shaped curves starting from the center shows the geometric positions where the difference in azimuth has the constant value ($\alpha = 0$). This diagram can be used in accordance with Fig. 2.3b in the following order:

Table 2.2 The form for calculation of Azimuth values

The GEO direction in relation to MES	Calculating of Azimuth angles
Course of MES toward S & W	$A = a$
Course of MES toward N & W	$A = 180^\circ - a$
Course of MES toward N & E	$A = 180^\circ + a$
Course of MES toward S & E	$A = 360^\circ - a$

- (1) First, it is necessary to note the longitude values of satellite (λ_S) and mobile (λ_M) and the latitude of the mobile (ϕ_M), then calculate the difference in longitude ($\Delta\lambda$), and plot the point into the graphic with both coordinates (ϕ_M & $\Delta\lambda$).
- (2) The value of elevation angle (ϵ) can then be determined by a plotted point from the group of parallel concentric curves.
- (3) The difference value of azimuth (a) can be determined by a plotted point from the group of fan-shaped curves starting from the center.
- (4) Finally, depending on the mobile position, the value of azimuth (A) can be determined on the basis of the relations presented in Table 2.2.

Inasmuch as the position of Ship Earth Station (SES) or any MES is of significant or greater height above sea level (if the bridge or ship's antenna is in a very high position) or according to the flight altitude of Aircraft Earth Station (AES), the elevation angle will be compensated by the following parameter:

$$x = \arccos(1 - H/R) \quad (2.38)$$

where H = height above sea level of observer or MES. Let us say, if the position of LES is a height of $H = 1000$ m above sea level, the value of $x \approx 1^\circ$. This example can be used for the determination of AES compensation parameters, depending on actual aircraft altitude. In such a way, the estimated value of elevation angle has to be subtracted for the value of the compensation parameter (x).

2.1.3.3 Satellite Track and Geometry (Longitude and Latitude)

The satellite track on the Earth's surface and the presentation of a satellite's position in correlation with the MES results from a spherical coordinate system, whose center is the middle of Earth, is illustrated in Fig. 2.3b. In this way, the satellite position in any time can be decided by the geographic coordinates, subsatellite point, and range of radius. Thus, the subsatellite point is a determined position on the Earth's surface; above, it is the satellite at its zenith.

The longitude and latitude are geographic coordinates of the subsatellite point, which can be calculated from the spherical triangle ($B'TP$), using the following relations:

$$\begin{aligned}\sin \varphi &= \sin(\theta + \omega) \sin i \\ \operatorname{tg}(\lambda_S - \Omega) &= \operatorname{tg}(\theta + \omega) \cos i\end{aligned}\quad (2.39)$$

With the presented equation in previous relation, it is possible to calculate the satellite path or trajectory of subsatellite points on the Earth's surface. The GEO track breaks out at the point of coordinates $\varphi = 0$ and $\lambda = \text{const}$.

Furthermore, considering geographic latitude (φ_M) and longitude (λ_M) of the point (M) on the Earth's surface presented in Fig. 2.3b, what can be the position of the MES, taking into consideration the arc (MP) of the angle illustrated in Fig. 2.3a, the central angle can be calculated by the following relations:

$$\begin{aligned}\cos \Psi &= \cos \varphi_S \cos \Delta \lambda \cos \varphi_M + \sin \varphi_S \sin \varphi_M \text{ or} \\ \cos \Psi &= \cos \text{arc MP} = \cos \varphi_M \cos \Delta \lambda\end{aligned}\quad (2.40)$$

The transition calculations from geographic to spherical coordinates and vice versa can be computed with the following equations:

$$\begin{aligned}\cos \Psi &= \cos \varphi \cos \Delta \lambda \text{ and } \operatorname{tg} A = \sin \Delta / \lambda \operatorname{tg} \varphi, \text{ respectively} \\ \sin \varphi &= \sin \Psi \cos A \text{ and } \operatorname{tg} \Delta \lambda = \operatorname{tg} \Psi \sin A\end{aligned}\quad (2.41)$$

These relations are useful for any point or area of coverage on the Earth's surface, then for a center of the area if it exists, as well as for spot beam and global area coverage for MSC systems. The optimum number of GEO satellites for global coverage can be determined by:

$$n = 180^\circ / \Psi \quad (2.42)$$

For instance, if $\delta = 0$ and $\Psi = 81^\circ$, it will be necessary to put into orbit only 3 GEO, and to get a global coverage from 75° N to 75° S geographic latitude. Hence, in a similar way the number of satellites can be calculated for other types of satellite orbits.

The trajectory of radio waves on a link between an MES and a satellite at distance (d) and the velocity of light ($c = 3 \times 10^8$ m/s) requires a propagation time equal to:

$$T = d/c \text{ (s)} \quad (2.43)$$

The phenomenon of apparent change in frequency of signal waves at the receiver when the signal source moves with respect to the receivers (Earth) was explained and quantified by Johann Doppler (1803–53). The frequency of the satellite transmission received on the ground increases as the satellite is approaching the ground observer and reduces as the satellite is moving away. This change in frequency is called Doppler effect or shift, which occurs on both the uplink and the downlink. This effect is quite pronounced for LEO and compensating for it requires

frequency tracking in a narrowband receiver, while its effects are negligible for GEO satellites. The Doppler shift at a transmitting frequency (f) and radial velocity (v_r) between the ground observer and the transmitter can be calculated by the following relation:

$$\Delta f_D = f v_r / c \text{ where } v_r = dR/dt \quad (2.44)$$

For an elliptical orbit, assuming that $R = r$, the radial velocity is given by:

$$v_r = dr/dt = (dr/\theta)(d\theta/dt) \quad (2.45)$$

The sign of the Doppler shift is positive when the satellite is approaching the observer and vice versa. Doppler effect can also be used to estimate the position of an observer provided that the orbital parameters of the satellite are precisely known. This is very important for the development of Doppler satellite tracking and determination systems.

2.2 Spacecraft Launching and Station-Keeping Techniques

The launch of the satellite and the controlling support services are a very critical point in the creation of space communications and the most expensive phase of the total system cost. At the same time, the need to make a satellite body capable of surviving the stresses of the launch stages is a major element in their design phase. Satellites are also designed to be compatible with more than one model of launch vehicle and launching type.

In a more determined sense, there are multistage expendable and, manned or unmanned, reusable launchers. Owing to location and type of site, there are land-based and sea-based launch systems. Additional rocket motors, such as perigee and apogee kick propulsion systems, may also be required.

The process of launching a satellite is based mostly on launching into an equatorial circular orbit and after in GEO, but similar processes or phases are used for all types of orbits. The processes involved in the launching technique depend on the type of satellite launcher, the geographical position of the launching site, and constraints associated with the payload. In order to successfully put the satellite into the transfer and drift orbit, the launcher must operate with great precision with regard to the magnitude and orientation of the velocity vector. On the other hand, launching operations necessitate either TT&T facilities at the launching base or at the stations distributed along the trajectory.

2.2.1 Satellite Installation and Launching Operations

Satellites are usually designed to be compatible with more than one prototype of launchers. Launching, putting, and controlling satellites into orbit are very expensive operations, so the expenses of launcher and support services can exceed the cost of the satellites themselves. The basic principle of any launch vehicle is that the rocket is propelled by reaction to the momentum of hot gas ejected through exhaust nozzles.

Thus, for a spacecraft to achieve synchronous orbit, it must be accelerated to a velocity of 3070 m/s in a zero-inclination orbit and raised a distance of 42,242 km from the center of the Earth. Most rocket engines use the oxygen in the atmosphere to burn their fuel but solid or liquid propellant for a launcher in space must comprise both a fuel and an oxygen agent. There are two techniques for launching a satellite, namely by direct ascent and by Hohmann transfer ellipse.

2.2.1.1 Direct Ascent Launching

A satellite may be launched into a circular orbit by using the direct ascent method, shown in Fig. 2.7a. The thrust of the launch vehicle is used to place the satellite in a trajectory, the turning point of which is marginally above the altitude of the desired orbit. The initial sequence of the ascent trajectory is the boost phase, which is powered by the various stages of the launch vehicle. This is followed by a coasting phase along the ballistic trajectory, the spacecraft at this point consisting of the last launcher stage and the satellite. As the velocity required to sustain an orbit will not have been attained at this point, the spacecraft falls back from the highest point of the ballistic trajectory.

When the satellite and final stage have fallen to the desired injection altitude, having in the meantime converted some of their potential energy into kinetic energy, the final stage of the launcher, called the Apogee Kick Motor (AKM), is

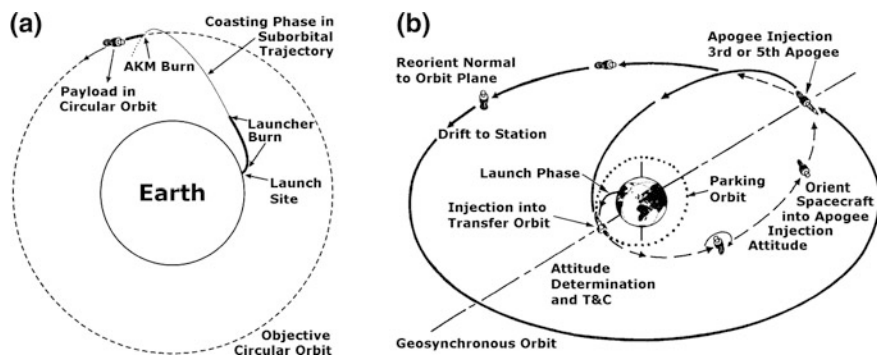


Fig. 2.7 Satellite installation in circular and synchronous orbit—Courtesy of Book: by Pascall

activated to provide the necessary velocity increase for injection into the chosen circular orbit. In effect, the AKM is often incorporated into the satellite itself, where other thrusters are also installed for adjusting the orbit or the altitude of the satellite throughout its operating lifetime in space. The typical launch vehicles for direct ascent satellite launching are US-based Titan IV, Russian-based Proton, and Ukrainian-based Zenit.

2.2.1.2 Indirect Ascent Launching

A satellite may be launched into an elliptical or synchronous orbit by using the successive or indirect ascent sequences, known as the Hohmann transfer ellipse method, illustrated in Fig. 2.7b. The Hohmann transfer ellipse method enables a satellite to be placed in an orbit at the desired altitude using the trajectory that requires the least energy. At the first sequence, the launch vehicle propels the satellite into a low parking orbit by the direct ascent method. The satellite is then injected into an elliptical transfer orbit, the apogee of which is the altitude of the desired circular synchronous orbit. At the apogee, additional thrust is applied by an AKM to provide the velocity increment necessary for the attainment of the required synchronous orbit. In practice, it is usual for the direct ascent method to be used to inject a satellite into a LEO and for the Hohmann transfer ellipse to be used for higher types of satellite orbits.

2.2.2 Satellite Launchers and Launching Systems

Two major types of launch vehicles can be used to put a satellite into LEO, HEO, and GEO constellation: Expendable and Reusable Vehicles. There are also two principal locations or site-based types of launching centers: Land-based and Sea-based launch systems.

2.2.2.1 Expendable Launching Vehicles

The great majority of communications satellites have been launched by expendable vehicles and this is likely to continue to be the case for many years to come. There are two types of these vehicles: expendable three-stage vehicles and expendable direct-injection vehicles.

1. Expendable Three-Stage Vehicles—Typical series of three-stage vehicles are Delta and Atlas (USA), Ariane (Europe), Long March (China), and H-II (Japan). In addition, a new generation of launchers has already been developed with two stages such as Delta III and Ariane 5. Both stages are propellant systems using cryogenic liquid fuel, while the first stage is assisted by nine strap-on solid-fuel motors.

The first and second stages of three-stage expendable launch vehicles are usually designed to lift it clear of the Earth's atmosphere, to accelerate horizontally to a velocity of about 8000 m/s and enter a parking orbit at a height of about 200 km. The plane of the parking orbit will be inclined to the equator at an angle not less than the latitude of the launch site. The most efficient way of getting from the parking orbit to a circular equatorial orbit is to convert the parking orbit into an elliptical orbit in the same plane, with the perigee at the height of the parking orbit and the apogee at about 36,000 km and then to convert the transfer ellipse to the GEO.

Thus, the third stage is fired as the satellite crosses the equator, which ensures that the apogee of the Geostationary Transfer Orbit (GTO) is in the equatorial plane. When the satellite is placed in the GTO, the third stage has completed its mission and is jettisoned. The final phase of the Hohmann transfer three-stage launch sequence is carried out by means of AKM built into the satellite. The propulsion of this motor is required to provide at the apogee of the GTO a velocity increment of such a magnitude and in such a direction as to reduce the orbit to zero and make the orbit circular. Once the satellite is in the GEO trajectory, the attitude is corrected, the antennas and solar panels are deployed and the satellite is drifted to the correct longitude (apogee position) for operation.

2. Expendable Direct-Injection Vehicles—Typical models of direct-injection launchers are the USA-based Titan IV and the Russian-based Proton, illustrated in Fig. 2.8 (a—Left) and (a—Right), respectively, and also Zenit (Ukraine). Otherwise, these types of vehicles do not need an AKM because direct-injection

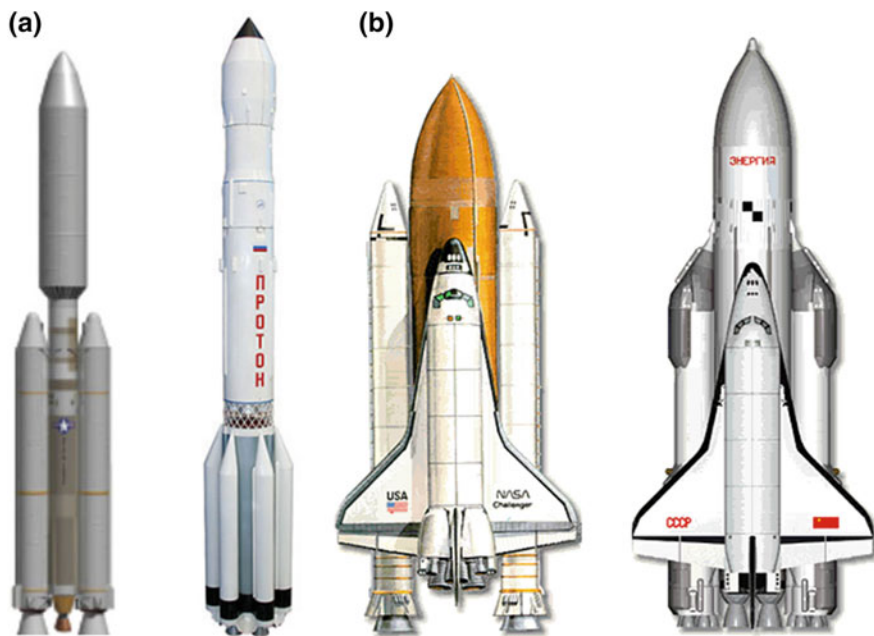


Fig. 2.8 Types of launch vehicles—Courtesy of Book: by Pascal

launchers have a fourth stage, which converts directly from GTO to GEO constellation. The Proton rocket is one of the most capable and reliable heavy lift launch vehicles in operation today. Proton D-1 and D-1-E launcher variants have three and four stages, respectively. At liftoff, the total weight of Proton is about 688 tons and this vehicle has the capability of placing a maximum of 4,500 and 2,600 kg into GTO and GEO, respectively.

2.2.2.2 Reusable Launch Vehicles

Reusable launch vehicles have already been developed in the USA (Space Shuttle) and former USSR (Energia/Buran), which have as their aim the development of vehicles that could journey into space and return, all or much of their structure being reusable, and thus, the satellite launching will cost less. Moreover, in using these launchers there will be less burnt-out upper stages than with expendable vehicles. What remains in space, the small pieces in transfer orbits for many years and much small debris, remains in LEO for a long time, adding to the growing space junk hazard for operational satellites and future space operations. There are other projects for the development of similar vehicles such as a small manned reusable Space Shuttle called Hermes (Europe) and Hope (Japan), unmanned space plane Hotol in UK is proposed, while in Germany and the USA two similar vehicles are projected: TAV (Trans Atmospheric Vehicle) and Sanger Space plane, respectively. Thus, in the development of these small vehicles, it is important to realize whether any of them could carry sufficient weight and be able to put communications satellites into the desired orbits.

1. **Space Shuttle**—The US-based NASA developed a fleet of manned reusable vehicles of Space Transportation System (STS) called Space Shuttle, which are capable of lifting a satellite of up to 29.5 tons into a parking orbit, inclined at 28.5° , with an altitude of up to 431 km, shown in Fig. 2.8 (b—Left). A Shuttle has three main elements: (1) the orbiter for carrying the satellite and crew; (2) a very large external tank containing propellant for the main engine of the orbiter; and (3) two solid-propellant boosters. The reusable Space Shuttle plane is 37.2 m long, the fuselage is 4.5 m in diameter, the wingspan is 23.8 m, and the mass is about 84.8 tons. This STS is designed to accommodate in total 7 crewmembers and passengers on board plane. The system came into service in 1981 and made over twenty successful operational flights until January 1986, when the Shuttle Challenger was destroyed by a fault in the solid-propellant booster and all the crew were killed in a tragic accident. Following this disaster, NASA redesigned the booster but decided to use STS only for regular launch program of government and scientific vehicles.

2. **Energia/Buran Space plane**—The launcher Energia is the most powerful operational reusable vehicle in the world, capable of carrying about 100 tons into space, whose four first-stage booster units are recoverable for reuse. In particular, it can launch the Buran space plane, enabling it to acquire a LEO and to land with the aid of its own rocket engine, shown in Fig. 2.8 (b—Right). The main purpose for which those very heavy lift vehicles were developed was to ferry personnel and

supplies for the Russian space station Mir and also to retrieve or repair satellites already in orbit.

The Energia vehicle can also carry into space a side-mounted canister containing an upper stage and a payload compartment suitable, for example, for a large heavy spacecraft or group of communications satellites to be placed in orbit. Energia flew for the first time on May 15, 1987, carrying a spacecraft mock-up and later on November 18, 1988, carrying an unmanned version of Buran space plane. The reusable Buran space plane is 36.3 m long, the fuselage is 5.6 m in diameter, the wingspan is 24 m, and the mass is about 100 tons. It can be flown in automatic configuration or under the control of a pilot to place satellites in LEO or to retrieve them and come back to base for the next use. Up to ten people, crew and passengers, can be accommodated and it can carry in the cargo bay up to 30 tons into an orbit of 200 km altitude and 51.6° inclination. In fact, this plane enables large satellites to be put into orbit and construction of space stations to be considered for both for telecommunication purposes and for scientific missions.

The Energia Launch Vehicle was also the successor to the N-1 Moon Rockets, except that Buran was also used to launch Polyus from Baikonur Cosmodrome in Kazakhstan (former Soviet Union). Energia was 60 m high and 18 m in diameter, consisting of a central core and four strap-on boosters, while the core was 58.1 m high and 7.7 m diameter. It used 4 RD-0120 rocket engines. The propellants were liquid hydrogen and oxygen. The strap-on boosters were then 38.3 m high and 3.9 m in diameter, with a single four-chamber RD-170 kerosene/liquid oxygen rocket engine.

In 1992, the Russian Space Agency (Roscosmos) decided to terminate the Energia/Buran Program due to Russia's economic difficulties after disintegration of former Soviet Union. At that stage, the second Orbiter had been assembled and assembly of the third Orbiter with improved performance was nearing completion.

Although the Energia project has been abandoned, it may return to service if a market is found, or adequate partners. Consideration is being given in Russia to the development of a more compact winged space plane designed to ferry personnel and their luggage into space, such as new developed space plane Kliper by NPO Energia, see Fig. 2.9. This compact shuttlecraft could be placed atop of a Proton, Soyuz, Angara or any other launchers.

2.2.2.3 Land-Based Launching Systems

Most satellite launches have taken place from the following launch facilities:

1. **US-Based Launch Centers**—The USA launches satellites from two main locations, in Florida Cape Canaveral, suitable for direct equatorial orbit, and the Vandenberg Air Force Base in California, suitable for polar orbit missions.
2. **Russian Launch Centers**—Russian satellites are launched from two main launch centers named Baikonur and Northern Cosmodrome. Baikonur lies north of Tyuratam in Kazakhstan, with the all launching support infrastructure for launching Proton and Energia heavy launchers. The Northern Cosmodrome is



Fig. 2.9 Russian space plane Kliper—Courtesy of Brochure: by Zak

located near Plesetsk, south of the town Archangelsk, suitable for launching satellites for all purposes in high-inclination orbits. This Cosmodrome is the world's busiest launch site.

3. **European Launch Centers**—The main European launch Cosmodrome is the Guiana Space Center in French Guiana, using Ariane vehicles. The position of this Cosmodrome enables the best advantage to be taken of the Earth's rotation for direct equatorial orbit.
4. **Chinese Launch Centers**—The principal launch sites in China are Jiuquan and Xi Chang, for launching Long March vehicles. In the meantime, the Xi Chang launch center has also become most used for launches into the GEO for the international market.
5. **Japanese Launch Centers**—The Japan's Tanegashima Space Center is situated in the prefecture of Kagashima. The facilities include the Takesaki Range for small rockets and the Osaki Range was used for the launch of H-I vehicles until the termination of program in 1992. After renovation, the Osaki Range will be used as the launching for the next generation of J-I Japanese vehicles. The new Yoshinobu launch complex has been constructed next to the Osaki center to satisfy the requirements of the new H-II launcher.

2.2.2.4 Sea-Based Launch Systems

The Sea Launch Multinational Organization was developed in March 1996 to overcome the cost of land-based launch infrastructure duplication around the world. The newly formed Sea Launch system is owned by the Sea Launch Partnership Limited in collaboration with international partners such as US Boeing Commercial

Space Company, Russian RSC Energia, Ukrainian KB Yuzhnoye/PO Yuzhmash, Shipping Anglo-Norwegian Kvaerner Group, and Sea Launch Company, LLC.

The Sea Launch Company, partner locations and operating centers, has US-based headquarters in Long Beach, California, and is manned by selected representatives of each of the partner companies.

The Sea Launch Partners have the following responsibilities and tasks:

1. Boeing responsibilities include designing and manufacturing the payload fairing and adapter, developing and operating the Home Port facility in Long Beach, integrating the spacecraft with the payload unit and the Sea Launch system, performing mission analysis and analytical integration, leading operations, securing launch licensing documents, and providing range services.
2. RSC Energia is responsible for developing and qualifying the Block DM-SL design modifications, manufacturing the Block DM-SL upper stage, developing and operating the automated ground support infrastructure and equipment, integrating the Block DM-SL with Zenit-2S and launch support equipment, planning and designing the CIS portion of launch operations, developing flight design documentation for the flight of the upper stage, and performing launch operations and range services.
3. KB Yuzhnoye/PO Yuzhmash are responsible for developing and qualifying Zenit-2S vehicle design modifications, integrating the launch vehicle flight hardware, developing flight design documentation for launch with respect to the first two stages, supporting Zenit processing and launch operations. Several significant configuration modifications have been made to allow the basic Zenit design to meet Sea Launch's unique requirements.
4. The Anglo-Norwegian Kvaerner Group is responsible for designing and modifying the Assembly and Command Vessel (ACV), designing and modifying the Launch Platform (LP), and integrating the marine elements. Furthermore, Barber Moss Marine Management is responsible for marine operations and maintenance of both vessels.

The partner team of contractors has developed an innovative approach to establishing Sea Launch as a reliable, cost-effective and flexible commercial launch

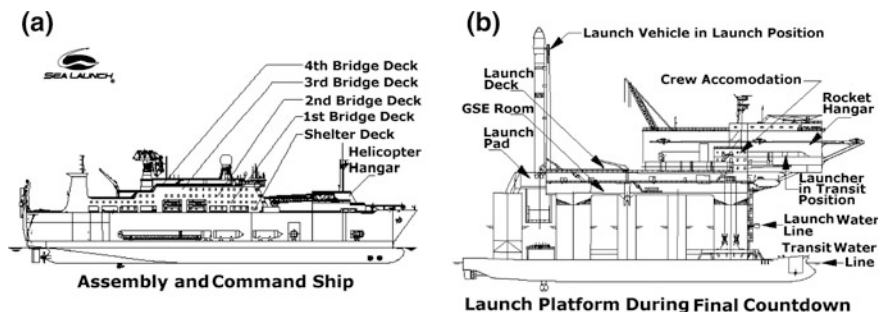


Fig. 2.10 Sea launch modules—Courtesy of Manual: by Sea Launch

system. Each partner is also a supplier to the venture, capitalizing on the strengths of these industry leaders. The system consists of two main modules: Assembly (Command and Control Ship) and Launch Platform, both illustrated in Fig. 2.10a, b, respectively. However, transit for the ACV and the LP from Home Port in Long Beach to the launch site on the equator takes 10–12 days, based on a speed of 10.1 knots.

The Sea Launch Home Port complex is located in Long Beach, California. The Home Port site provides the facilities, equipment, supplies, personnel, and other procedures necessary to receive, transport, process, test, and integrate the spacecraft and its associated support equipment with the Sea Launch system. The Home Port also serves as the marine base of operations for both of the Sea Launch vessels. The personnel providing the day-to-day support and service during prelaunch processing and launch conduct to Sea Launch and its customers are located at the Home Port. The ACV performs four important functions for Sea Launch operations: (1) It serves as the facility for assembly, processing, and checkout of the launch vehicle; (2) it houses the Launch Control Center (LCC), which monitors and controls all operations at the launch site; (3) it acts as the base for tracking the initial ascent of the launch vehicle; and (4) it provides accommodation for the marine and launch crews during transit to and from the launch site. Therefore, the ACV is designed and constructed specifically to suit the unique requirements of Sea Launch. The ship's overall dimensions are nearly 200 m in length, 32 m in beam, and a displacement of 34,000 tons.

Major features of the ACV include: a rocket assembly compartment; the LCC with viewing room; helicopter capability; customer work areas; spacecraft contractor; and customer accommodation. The rocket assembly compartment, which is located on the main deck of the ACV, hosts the final assembly and processing of the launch vehicle. This activity is conducted while the vessels are at the Home Port and typically in parallel with spacecraft processing. The bow of the main deck is dedicated to processing and fueling the Block DM-SL of the Zenit launch vehicle. After the completion of spacecraft processing and encapsulation, the encapsulated payload is transferred into the rocket assembly compartment, where it is integrated with the Zenit-2S and Block DM. The launchers and the satellite are assembled horizontally in the ACV before sailing from the port of Long Beach to the designated launch site. A launcher with a payload will then be transferred in the horizontal position to the launch pad on LP and raised to a vertical position for fueling and launching.

During the launch sequence, the crew of the LP will be transferred to the ACV, which will initiate and control the launch from a position about 3 miles away from the LP pad. The LP is an extremely stable sea platform from which to conduct the launch, control, and other operations. The LP rides catamaran-style on a pair of large pontoons and is self-propelled by a four-screw propulsion system (two in each lower hull, aft), which is powered by four direct-current double armature-type motors, each of which are rated at 3000 hp. The LP in navigation has normal draft at sea water level but once at the launch location, the pontoons are submerged to a depth of 22.5 m to achieve a very stable launch position, level to within

approximately 1°. The ballast tanks are located in the pontoons and in the lower part of the columns. Six ballast pumps, three in each pontoon, serve them. The LP has an overall length of approximately 133 m at the pontoons and the launch deck is 78 by 66.8 m. The Zenit-3SL launcher is a two-stage liquid propellant launch vehicle solution capable of transporting a spacecraft to a variety of orbits.

The original two-stage Zenit was designed by KB Yuzhnoye quickly to reconstitute former Soviet military satellite constellations. The design emphasizes robustness, ease of operation, and fast reaction times. The result is a highly automated launch capability using a minimum complement of launch personnel. The launcher as an integrated part of the Sea Launch system is designed to place spacecraft into a variety of orbits and is capable of putting 5250 kg of payload into GEO.

The Sea Launch mission provides a number of technical support systems that are available for the customer's use in support of the launch process, including most importantly the following:

1. Communications—Internal communication systems are distributed between the ACV and LP. This system includes CCTV, telephones, intercom, video teleconferencing, public address, and vessel-to-vessel radiocommunications, known as the line-of-sight (LOS) direct system.

This system links with the external communication system and provides a worldwide network that interconnects the various segments of the Sea Launch program. The external communication system includes Intelsat and two ground stations. The LES are located in Brewster, Washington, and Eik, Norway, and provide the primary distribution gateways to the other communication nodes. Customers can connect to the Sea Launch communication network through the convenient Brewster site. The Intelsat system ties in with the ACV and launch platform PABX systems to provide telephone connectivity. Additionally, critical Voice, Fax, Tlx, or data capability can be ensured by the Inmarsat satellite SES service.

2. Tracking and Data-Relay Satellite System (TDRSS)—The Sea Launch system uses a unique dual telemetry stream with the TDRSS. Telemetry is simultaneously received from the Zenit stages, the Block DM upper stage, and the payload unit during certain portions of the flight. The Block DM upper stage and payload unit data are combined but the Zenit data are sent to a separate TDRSS receiver. Zenit data are received shortly after liftoff at approximately 9 s and continue until Zenit Stage 2/Block DM separation, at around 9 min. These data are routed from the NASA White Sands LES to the Sea Launch Brewster LES and to the ACV. Otherwise, the data are also recorded at White Sands and Brewster for later playback to the KB Yuzhnoye design center.

When the payload fairing separates, the payload unit transmitter shifts from sending high-rate payload accommodation data by LOS to sending combined payload unit/Block DM by TDRSS. The combined data are again routed from White Sands to Brewster, where it is separated into Block DM and payload unit data and then sent on to the ACV. The data are received on board the ship through the Intelsat communications terminal and are routed to Room 15 for upper-stage data and Room 94 for PLU data. Simultaneously, Brewster routes Block DM data

to the Energia Moscow control center station. However, the TDRSS coverage continues until after playback of the recorded Block DM data.

3. Telemetry System—Sea Launch uses LOS telemetry systems for the initial flight phase, as well as the TDRSS for later phases. The LOS system, which includes the Proton antenna and the S-band system, is located on the ACV. Other telemetry assets include Russian ground tracking stations and the Energia Moscow control center. The following subsections apply to launch vehicle and payload unit telemetry reception and routing.

4. Weather (WX) Data System and Forecast—The ACV unit has a self-contained WX station, which includes a motion-stabilized C-band Doppler radar equipment, surface wind instruments, wave radar, upper-atmospheric balloon release station, ambient condition sensors, and access to satellite imagery and information from an on-site buoy.

2.3 Types of Orbits for Mobile and Other Satellite Systems

An orbit is the circular or elliptical path that the satellite traverses through space. This path appears in the chosen orbital plane in the same or different angle to the equatorial plane. All communications satellites always remain near the Earth and keep going around the same orbit, directed by centrifugal and centripetal forces. Each orbit has certain advantages in terms of launching (getting satellite into position), station keeping (keeping the satellite in place), roaming (providing adequate coverage), and maintaining necessary quality of communication services, such as continuous availability, reliability, power requirements, time delay, propagation loss, and network stability.

There is a large range of satellite orbits but not all of them are useful for fixed and Mobile Satellite Communications systems. In general, the one of most commonly used orbits for satellite communications is GEO constellations, after which HEO and latterly GIO, PEO, LEO, and MEO, shown in Fig. 2.11a.

Otherwise, it is essential to consider that satellites can serve all communication, navigation, meteorological, and observation systems for which they cannot have an attribute such as fixed or mobile satellites and the only common difference is which type of payload or transponder they carry on board. For example, its name can be satellite-specified for fixed communications but in effect it can carry major transponders for fixed communications and others for mobile or other purposes and vice versa.

After many years of research and experiments spent on finding the global standardization for spatial communications, satellites remained the only means of providing near-global coverage, even in those parts which other communications systems are not able to reach. There is always doubt about the best orbital

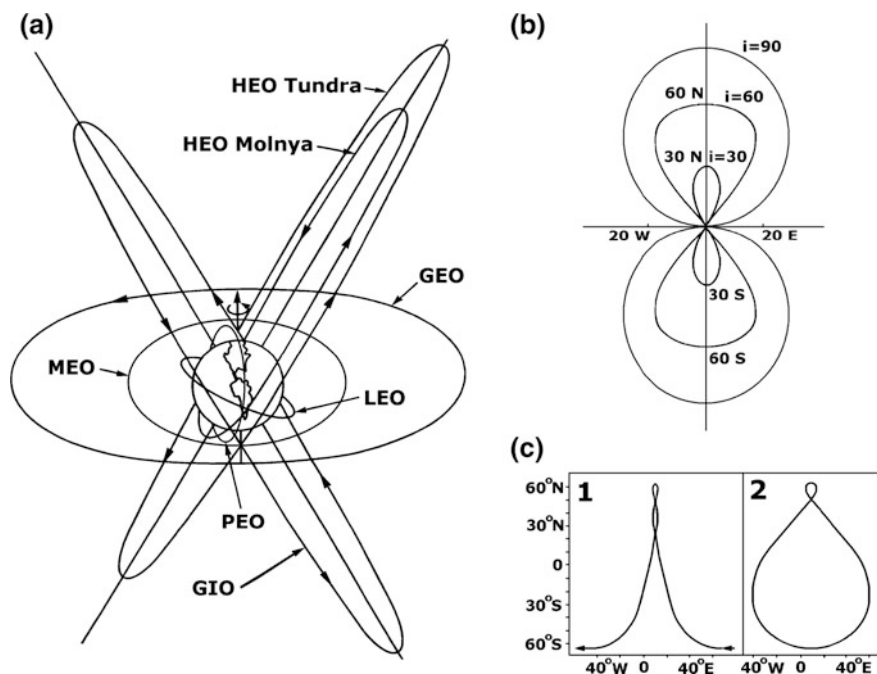


Fig. 2.11 Type of satellite orbits and tracks—Courtesy of Book: by Evans

constellation that can realize an appropriate global coverage and a reliable communications solution.

Unfortunately, there is no perfect system today; all systems have some advantages or disadvantages. The best conclusion is to abridge the story and to say briefly that today the GEO system is the best solution and has only congestion as a more serious problem. The extensive use of GEO shows that it provides something good. Inmarsat is the biggest GEO operator whose service and revenue confirm this point of view. The advantages of Inmarsat MSC solutions can be realized if someone uses them such as operators on board mobiles and finds out how powerful they are. Most of other regional GEO worldwide networks, such as ACeS, Optus and Thuraya, are also more successful than other non-GEO constellations.

In particular, Big LEO and ICO systems or hybrid constellations such as Ellipso have had several years of serious economical and concept difficulties. It is sufficient to see Table 2.3 to understand that the major reasons for LEO problems are more satellite cost, complex network, and short satellite visibility and lifetime. The LEO/PEO constellations are the same or similar and because of differences in inclination angle of orbital plane and type of coverage, they will be considered separately.

The track of the satellite varies from 0 to 360° , see Fig. 2.11b. The track of the GEO satellite is at a point in the center of the coordinate system; two tracks are

Table 2.3 The properties of four major orbits

Orbital properties	LEO	MEO	HEO	GEO
Development period	Long	Short	Medium	Long
Launch & satellite cost	Maximum	Maximum	Medium	Medium
Satellite life (Years)	3–7	10–15	2–4	10–15
Congestion	Low	Low	Low	High
Radiation damage	Zero	Small	Big	Small
Orbital period	<100 min	8–12 h	½ Sidereal Day	1 Sidereal Day
Inclination	90°	45°	63.4°	Zero
Coverage	Global	Global	Near global	Near global
Altitude range (km ⁻³)	0.5–1.5	8–20	40/A – 1/P	40 (i = 0)
Satellite visibility	Short	Medium	Medium	Continuous
Handover	Very Much	Medium	No	No
Elevation variations	Rapid	Slow	Zero	Zero
Eccentricity	0 to High	High	High	Zero
Handheld terminal	Possible	Possible	Possible	Possible
Network complexity	Complex	Medium	Simple	Simple
Tx Power/Antenna	Low	Low	Low/High	Low/High
Gain	Short	Medium	Large	Large
Propagation delay	Low	Medium	High	High
Propagation loss	High	Medium	Low	Zero

apparent movements of the GIO satellite with respect to the ascending node of both 30° and 60° inclination angles and the last is the track of the PEO satellite with an inclined orbit plane to the equator of 90°. The tracks of HEO Molniya (part of the track) and Tundra (complete track) orbits are shown in Fig. 2.11 (c-1/c-2), respectively. These two tracks pass over the African Continent and almost all of Europe.

This is very important for MSC systems that the orbit used can provide satellite view during 24 h with less handovers and network difficulties. However, for other types of broadcasting a communications satellite must be visible from the region concerned during the periods when it is desired to provide a communication service, which can vary from a few hours to 24 h a day. When the service is not continuous, it is desirable that the intervals during which the service is available repeat each day at the same time.

2.3.1 Low Earth Orbits (LEO)

The LEO systems are either elliptical or more usually circular satellite orbits between 500 and 2000 km above the surface of the Earth and below the Inner Van Allen Belt. The orbit period at these altitudes varies between ninety minutes and

two hours. The radius of the footprint of a communications satellite in LEO varies from 3000 to 4000 km.

Therefore, the maximum time during which a satellite in LEO orbit is above the local horizon for an observer on the Earth is up to 20 min. Earth is up to 20 minutes, which is shown in Fig. 2.11a. In this case, the traffic to a LEO satellite has to be handed over much more frequently than all other types of orbit. At this point, when a satellite, which is serving a particular user, moves below the local horizon, it needs to be able to quickly handover the service to a succeeding one in the same or adjacent orbit.

Due to the relatively large movement of a satellite in LEO constellation with respect to an observer on the Earth, satellite systems using this type of orbit need to be able to cope with large Doppler shifts. In fact, satellites in LEO are not affected at all by radiation damage but are affected by atmospheric drag, which causes the orbit to gradually deteriorate. Satellites in LEO and MEO constellations are subjected to orbital perturbation. For very LEO satellites, the aerodynamic drag is likely to be significant and in general, some of the other perturbations, such as precession of the argument of the perigee, resolve to zero in the orbit is circular or polar. On the other hand, a perturbation is unlikely to have a serious effect on the operation of a multisatellite constellation since it will usually affect all satellites of the configuration in equal measure.

The major advantages of LEO are as follows:

- (a) The LEO system may become important in the field of MSC using handheld terminals with global roaming and to be exceedingly useful in areas not covered by cellular systems. The LEO constellations cover almost the entire Earth's surface and some of them provide polar coverage and show promise in the fields of mobile data and Internet and FSS networks for broadband data transmission and communications.
- (b) High Doppler shift allows the LEO system to be used for satellite positioning, tracking, and determination.
- (c) The relatively small distance between LES and LEO results in much lower power and smaller user terminals. Furthermore, the one-way speed-of-light propagation delay of at least 0.25 s using GEO is obviated with LEO, in which effect can be annoying in two-way voice transmission. For example, for two-way voice via a satellite at an altitude of about 1000 km, the delay is only 13 ms in total for uplink and downlink.
- (d) Satellite path diversity eliminates signal interruption due to path obstruction. In Fig. 2.12, handover from satellite A to satellite B is demonstrated, as well as path diversity between satellites B and C. This figure illustrates the LEO MSS space and ground architecture with utilization of handheld personal terminals (PES). On the other hand, the Satellite Access Node (SAN) is the LES providing a link between PES terminals through satellites and ground telecommunications infrastructures.

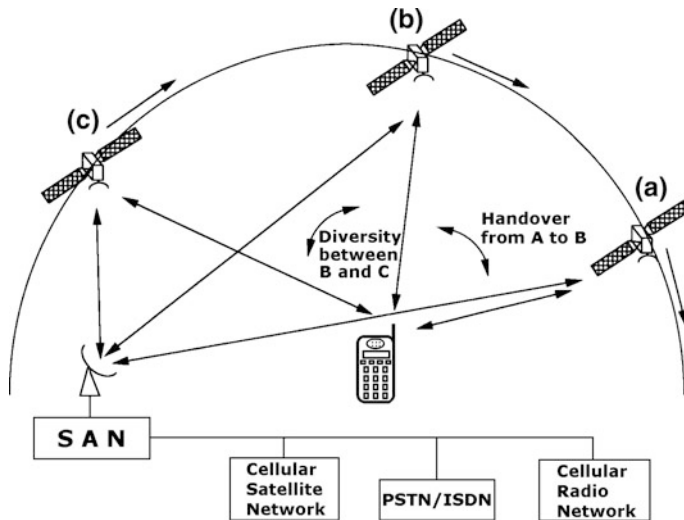


Fig. 2.12 LEO MSS diversity and handover—Courtesy of Book: by Huurdeman

The disadvantages of LEO are as follows:

- (a) The orbit period at about 1000 km altitude is in the order of approximately 100 min and the visibility at a point on the Earth is only some 10 min, requiring 40–80 satellites in six to seven planes for global coverage. Thus, in reality a GMSC system using this type of orbit requires a large number of satellites, in a number of different inclined orbits, which increases the total cost of the network.
- (b) Frequently handover is necessary for uninterrupted communications. Satellite visibility for MES could be improved by using more satellites. In such a way, the optimum number of satellites of about 48 inclined in the constellation in a carefully optimized pattern of orbit planes will provide continuous visibility of one or other of the satellites at any location on Earth surface.
- (c) During times of the year that the orbital plane is in the direction of the Sun, a satellite in LEO is eclipsed for almost one-third of the orbit period. Consequently, there is a significant demand on battery power, with up to 5000 charge/discharge cycles per year, which, with existing NiCd types of batteries, reduces satellite lifetimes to 3–7 years.
- (d) The launch cost is low, with direct injection into the orbit of several satellites, but the total cost is very high, with a minimum of 40 satellites being produced.

The first-generation LEO satellites were used for military communications because single GEO could be an easy target for an opponent. The large number of LEO satellites will reduce enormously the risk of vulnerability if someone wishes to destroy only one satellite.

Because this orbit configuration is in the initial phase of its exploitation, it is still free of congestion problems. There are two types of LEO constellations known as Little LEO, useful for messaging and satellite tracking systems and Big LEO, suitable for voice, video, and data communications.

2.3.1.1 Little LEO

The Little LEO mobile satellite systems are a category of LEO solutions that utilize birds of small size and low mass for low-bit-rate transmission under 1 Kb/s. Thus, the Little LEO systems are the constellations of very small non-GEO satellites, which operate in LEO orbits, providing mainly mobile data messaging and tracking services for vehicles and ships and other FSS and broadcasting services. The FCC has allocated a frequency band of 137–138 MHz for the downlink and 148–149.9 MHz for the uplink to Little LEO systems.

The Little LEO satellite frequency spectrum is a heavily utilized for different private and government services worldwide, such as Orbcomm, Falsat, VITASat, Starnet, and other systems. The mass of satellites in these solutions ranges from 40 kg in Orbcomm to 150 kg in the Starnet system. These systems prefer a spectrum below 1 GHz, because it enables the use of cost-effective equipment for non-voice two-way messaging and positioning satellite transceiver, which would be equipped with an alphanumeric display.

2.3.1.2 Big LEO

The Big LEO is a larger non-GEO satellite system, which operates in LEO constellations and provides mainly mobile telephony, Fax, data, and RDSS services. Compared to the Little LEO systems, satellites in Big LEO systems are expected to be bigger in body and to have more power and bandwidth to provide a different service to their subscribers. This system will use the underutilized spectrum available in the L-band because of the commercial failure of proposed RDSS service. Currently, the frequency spectrum of 1610–1626.5 MHz for uplinks and 2483.5–2500 MHz for downlinks is assigned to these MSC systems. It is interesting to note that although the names of these systems include LEO, their frequencies are the ones usually utilized in MEO and GEO satellite systems. For this reason, a new ICO system is systematized in the category of Big LEO constellations together with real Big LEO systems, such as Iridium and Globalstar, which are located at a lower altitude than ICO, at about 700–1500 km from the Earth's surface. Thus, all of the new proposed Big LEO systems would offer global handheld telephone service by means of satellites on lower altitudes moving very fast, instead of fixed GEO relays. The bigger size of the satellites enables them to carry a transponder on board with more complex data processing facilities than the simple store-and-forward feature of the Little LEO satellite configuration. Hence, an

important fact is that these systems are networking with cellular and spreading their roaming and billing capabilities in real global coverage.

2.3.2 Circular Orbits

The GEO satellite constellation has great advantages for MSS communication applications where polar coverage is not required but there are solutions for providing polar roaming. Satellite orbits in 63.4° inclined high-apogee HEO have some advantages from GEO also providing polar coverage. The most popular circular equatorial orbit with zero inclination is the GEO satellite constellation. The period of rotation is equal to that of the Earth and has the same direction. However, both of these satellite orbits exhibit high LOS loss and long transmission times and delays. Using new technology, these problems can be solved, or as an alternative to these orbits there are LEO and MEO constellations with their good and bad characteristics. The choice of orbit depends on the nature of the MSC mission, the acceptable interference in an adequate environment and the performance of the launchers.

2.3.2.1 Medium Earth Orbits (MEO)

The MEO satellite constellations, known also as Intermediate Circular Orbits (ICO), are circular orbits located at an altitude of around 10,000–20,000 km between the Van Allen Belts. The MEO satellites are operated in a similar way to Big LEO systems providing global coverage, which orbit is shown in Fig. 2.11a.

However, compared to a LEO system a MEO constellation can only be in circular orbit; Doppler effect and handover is less frequent; propagation delay is about 70 ms and free-space loss is greater; satellites are affected by radiation damages from the Inner Van Allen Belt only during the launching period; fewer eclipse cycles means that battery lifetime will be more than 7 years; cosmic radiation is lower, with subsequently longer life expectancy for the complete MEO configuration; higher average elevation angle from users to satellite minimizes probability of LOS blockage and higher RF output power required for both indoor and handheld terminals.

A LEO constellation for MSS global coverage requires around 10 satellites in two or three orbital planes, each plane inclining 45° to the equator. Their orbit period measures about 6–8 h, providing slightly over 1 h local visibility above the horizon for an observer on the Earth and handover from one to the next satellite is every 6 h minimum.

There is in exploitation a special model of MEO constellation known in practice as Highly Inclined Orbit. This particular orbit is of interest because it has been chosen for existing and proposed GNSS systems such as Navstar (GPS), Navsat, GLONASS, and the newly developed Galileo. In all, complete implementation of this orbit configuration would have 24 satellites in 3 orbital planes equidistant from

each other, at an altitude of 20,000 km and at an inclination of 55° . In comparison with existing GNSS, the new Galileo system will have 30 satellites in high MEO of about 28,000 km and at a similar inclination of 56° . At this point, its interest to polar MSC would be the eventual prospect of satellite sharing with navigation services, in a similar fashion to a high PEO with minimum of 3 satellites in the same orbital plane.

2.3.2.2 Geostationary Earth Orbits (GEO)

A GEO has a circular orbit in the equatorial plane, with an orbital period equal to the rotation of the Earth of 1 sidereal day, which is achieved with an orbital radius of 66,107 (Equatorial) Earth Radii, or an orbital height of 35,786 km. Otherwise, a satellite in a GEO will appear fixed above the surface of the Earth and remain in a stationary position relative to the Earth itself. Theoretically, this orbit is with zero inclination and track as a point but in practice, the orbit has small nonzero values for inclination and eccentricity, causing the satellite to trace out a small figure eight in the sky, which is shown in Fig. 2.11a.

The footprint or service area of a GEO satellite covers almost one-third of the Earth's surface or 120° in longitude direction and up to 75° – 78° latitude North and South of the Equator but cannot cover the polar regions. In this way, near-global coverage can be achieved with a minimum of three satellites in orbit moved apart by 120° , although the best solution is to employ four GEO satellites for better overlapping. This type of orbit is essentially used for commercial communication services for both FSS and MSS with the following advantages:

- (a) The satellite remains stationary with respect to one point on the Earth's surface and so the LES antennas can be beamed exactly toward the focus of the GEO satellite without periodical tracking. Only mobile high-gain antennas need auto-tracking systems, while low-gain omnidirectional antennas are free of tracking systems.
- (b) The new Inmarsat GEO space constellation consisting of four satellites can cover all three-ocean regions with four overlapping longitudes, except for the polar regions beyond latitudes of 75° North and South. Otherwise, the polar regions can be covered for maritime and aeronautical MSS applications with current HF radio systems or in combination with PEO or HEO satellite constellations.
- (c) The Doppler shift, affecting synchronous digital systems caused by satellites to drift in orbit (affected by the gravitation of the Moon and to a lesser extent of the Sun), is small for all LES and MES within satellite coverage.

The disadvantages of GEO compared with LEO and MEO operation are as follows:

- (a) The long signal delay is due to the large distance of about 35,800 km if the satellite is in zenith for MES and about 41,000 km at the minimum elevation

angle of about 5° . For the EM waves traveling at the speed of light, this causes a round-trip signal delay of 240–270 ms and full duplex delay of 480–540 ms. Thus, the voice used via satellite can experience some disturbance but echo cancellation devices developed in the 1980s can reduce the problem. Besides, for data transmission equipment, especially when using error-correction protocols that require retransmission of blocks with detected errors, complex circuitry with special high-capacity buffer devices is required to overcome delay problems. In addition, practical experience has shown that given good control of the echo, a telephone connection which includes one hop in each direction via a GEO satellite is acceptable to public users.

- (b) The required higher RF output power and the use of directional antennas aggravate GEO operation slightly for use with handheld terminals, although it is not critical, because some GEO operators provide this service, such as Thuraya and ACeS.
- (c) The launch procedure to put a satellite in GEO is expensive but the total cost of 4 satellites is less than the cost of a minimum of 12 or 40 for MEO and LEO, respectively.

As stated earlier, the major disadvantage of a GEO satellite in a voice transmission is the round-trip delay between satellite and LES of approximately 2.5 s, which can be successfully solved with current and newly advertised echo cancellation circuits. Because of the enormous use of the GEO constellation for many space applications, some parts of the GEO are becoming congested, owing to only one radius and latitude. This orbit is geostationary and so its track is one point called the subsatellite point and obviously, handover and Doppler effect does not apply to GEO. A GEO satellite is at essentially fixed latitude and longitude, so even a narrow-beam Earth antenna can remain fixed. Satellites in GEO can use high- and recently low-gain antennas, which helps to overcome the great distances in achieving the required Effective Radiated Isotropic Power (EIRP) at ground level. On the other hand, using satellite spot beam antennas, coverage can be confined to smaller spot areas, bigger power and higher speed of transmission, such as new generation of Inmarsat-3 spacecraft. Furthermore, a variety of perturbing forces causes the GEO satellite to drift out of its path and assigned position toward so-called inclined orbit (GIO). By far the most important perturbations are the lunar and at a lesser degree the solar gravitational forces, which cause the satellite to drift in latitude or North–South direction. The longitudinal drift in East–West direction is caused by fluctuations in the gravitational forces from the Earth, due to its non-spherical shape and by fluctuations in solar radiation pressure. To counteract these perturbations, the spacecraft needs station-keeping devices. The GEO satellites pass through both Van Allen Belts only on launch, so their effect is insignificant. After reaching the end of operational life, a satellite has to be removed from its orbital slots into a graveyard orbit some 200 km above the GEO plane.

Thus, the GEO satellite constellation seems likely to continue to dominate in the satellite communications world, especially in MSS, providing near-global coverage with low- and high-power transmission. In particular attractive is the reliable and

economical use of the Inmarsat standard-C low-power transceiver and low-gain omnidirectional antenna for maritime, land vehicles and aeronautical two-way data/messaging and telex and one-way e-mail service.

The major existing GEO mobile systems in the world are Inmarsat and GEOSAT of the COSPAS–SARSAT system as global solutions and ACeS, AMSC, MSAT, Artemis, Emsat, Optus, N-Star, Solidaridad, and Thuraya as regional networks. Some of these systems, such as ACeS and Thuraya, also provide a service for handheld and mini indoor terminals, which makes it obvious that some authors made the mistake of assuming that for GEO it is very difficult to provide a handheld service and that Inmarsat mini-M is the smallest terminal for GEO, as is mentioned on page 13 of “Low Earth Orbital Satellites for Personal Communication Networks” written by A Jamalipour.

Early in 1995, Pasifik Satelit Nusantara of Indonesia along with Philippine Long Distance Telephone and Jasmine International Public Co Ltd of Thailand came together and formed a joint venture for MSS today known as Asia Cellular System (ACeS). The ACeS handheld dual mode (GSM/ACeS) terminal is manufactured by Ericsson.

Thuraya is Private Joint Stock Company registered on April 26, 1997, in UAE under Federal Law No. 8 of 1984 as a Regional GEO Mobile Satellite Communication System Operator providing voice, low-bit-rate data and facsimile services. The two prototypes of Thuraya handheld terminals are being manufactured by Hughes Network Systems (USA) and ASCOM (Switzerland).

2.3.2.3 Geosynchronous Inclined Orbits (GIO)

This system would consist in four satellites at six-hour intervals around the Earth orbit at an inclination of 45° to the equatorial plane, which is shown in Fig. 2.11a. The satellites provide polar coverage for six hours either side of their most northerly and southerly movement. Special LES with full-tracking antennas are needed; therefore, this system in general must be considered complex and expensive for a polar communication system.

Otherwise, a GIO satellite has a period of orbit equal to or very little different from a sidereal day (23 h 56 min and 4.1 s), which is time for one complete revolution of the Earth. The satellite movement speed has only very little difference from the angular velocity of the Earth, so this movement also has constant angular velocity. Otherwise, the projection of this movement on the equatorial plane is not at a constant velocity. There is an apparent movement of the satellite with respect to the reference meridian on the surface of the Earth and that of the satellite on passing through the nodes. The orbit may be inclined at any angle, which produces a repeating ground track. Figure 2.11b presents tracks of 30° and 60° inclined orbits. The coupled N–S and E–W motion of GIO satellites is shown as a figure eight pattern, while the patterns could also be distorted circles. Depending on the inclination angle, the GIO satellite shows points on the equator at various longitudes.

A satellite may operate in this orbit for several reasons. First, it is often desirable to save the inclination control fuel required for GEO circle. Sometimes there is no need to control inclination because tracking LES antennas are required for other reasons, while mobiles such as ships and aircraft require tracking antennas. Some GEO satellites may last beyond their planned lifetime if run low on fuel and cease inclination control. In effect, the GIO constellation with nonzero inclination can be chosen because of easy launching and placing of the satellite into orbit. This satellite must move with an angular velocity equal to the Earth and be in a prograde orbit, that is, revolving eastward in the same direction as the Earth rotates. Otherwise, the only requirements for a GIO constellation are the right period and direction of rotation.

2.3.3 Highly Elliptical Orbits (HEO)

Using inclined HEO configuration, both polar areas can be effectively covered with four satellites; two in each polar orbit. The elevation angle to the HEO satellites remains high for most of the 12-h period of visibility, which is especially required for continuous Euro-Asian regional coverage providing land MSC service. At this point, blocking of the beam due to occlusion of the satellite by buildings, mountains, hills, and trees is minimized. Beside, multiple trajectories caused by successive reflection of various obstacles are also reduced in comparison with systems operating with low-elevation angles, such as GEO.

The apogee altitude combines polar coverages with nearly synchronous advantages. Thus, minimum two special LES in both northern and southern polar regions are required to serve MES terminals. The LES tracking can be reached by a fairly directive fixed antenna while the satellite is in its slow apogee sector, the HEO space constellation is namely designed to cover the area under the apogee. Tracking of the satellite is facilitated on account of the small apparent movement and the long visibility duration. Otherwise, it is even possible to use antennas whose 3 dB bandwidth is a few tens of degrees, with fixed pointing toward the zenith, which permits the complexity and cost of the terminal to be reduced while retaining a high gain. A satellite in HEO constellation near the apogee can also use a high-gain antenna to overcome the great distances in achieving the required EIRP values. The noise captured by the LES antenna, from the ground or due to interference from other radio systems and atmosphere, is also minimized due to the high-elevation angles. At any rate, these advantages have led the former USSR to use these orbits for a long time in order to provide coverage of high-latitude territories for mobile systems.

The HEO satellite two-way voice transmission has a similar delay as a GEO at the apogee of about 0.25 s. Therefore, free-space loss and propagation delay for HEO is comparable to that of the GEO constellation. Compared with GEO, the launch and satellite cost of the HEO constellation is reasonably low; this constellation is free of congestion because of only a few current and projected new HEO

systems and provides high-elevation angles for LES, which reduces atmospheric losses. Due to the relatively large movement of a satellite in HEO with respect to an observer on Earth, satellite systems using this model of orbit need to be able to cope with large Doppler shifts, 14 kHz for Molniya and 6 kHz for Tundra orbits in L-band 1.6 GHz. However, as the former USSR's experience has shown, satellites in this orbit tend to have rather a short lifetime due to the repetitive crossing of both Van Allen Belts. The rest of the disadvantages are the necessity of constant satellite tracking at the MES, compensation of signal loss variation, long eclipse periods, and complex control system of MES and spacecraft.

The HEO satellite typically has a perigee at about 500 km above the Earth's surface and an apogee as high as 50,000 km. The orbits are inclined at 63.4° in order to provide services to locations at high northern latitudes. The particular inclination value is selected in order to avoid rotation of the apsides, i.e., the intersection of a line from the Earth's center to the apogee, and the Earth's surface will always occur at latitude of 63.4°N . Orbit period varies from eight to 24 h. Owing to the high eccentricity of the orbit, a satellite will spend about two-thirds of the orbital period near apogee and during that time it appears to be almost stationary for an observer on Earth (this is referred to as apogee dwell). After this period, a switchover needs to occur to another satellite in the same orbit in order to avoid loss of communications. There have to be at least three HEO satellites in orbit, with traffic being handed over from one to the next every eight hours at a minimum.

When there is an orbit in HEO plane of nonzero inclination, the satellite passes over the region situated on each side of the equator and will possibly cover the polar regions if the inclination of the orbit is close to 90° . By orienting the apsidal line, namely the line between perigee and apogee, in the vicinity of the perpendicular to the line of nodes (when ω is close to 90° or 270°), the HEO satellite at the apogee systematically returns above the regions of a given hemisphere. In this way, it is possible to establish satellite links with LES or MES located at high latitudes. Although the satellite remains for several hours in the vicinity of the apogee, it does move with respect to the Earth and after a time dependent on the position of the MES, the satellite disappears over the horizon as seen from the mobiles. However, to establish permanent links it is necessary to provide several suitably phased satellites in similar orbits, which are spaced around the Earth (with different right ascensions of the ascending node and regularly distributed between 0 and 2π) in such a way that the satellite moving away from the apogee is replaced (handover) by another satellite in the same area of the sky as seen from the MES. However, the problems of satellite acquisition and tracking by the MES are simplified. Finally, there only remains the problem of handover and switching the links from one satellite to other, so the RF link frequencies of the various satellites can be different in order to avoid interference.

Examples of HEO systems are Molniya, Tundra, Loopus, Borealis of Ellipso system and Archimedes, which orbits are shown in Fig. 2.11a. The ESA proposed Archimedes system employs a so-called "M-HEO" 8-hour orbit. This produces three apogees spaced at 120° . Each apogee corresponds to a service area, which

could cover a major population centre, for example the whole European continent, the Far East and North America.

2.3.3.1 Molniya Orbit

The first prototype HEO Molniya satellite was launched in 1964 and to date more than 150 have been deployed, primarily produced by the Applied Mechanics NPO in Krasnoyarsk, former USSR. The HEO Molniya satellites weigh approximately 1.6 metric tons at launch and stand 4.4 m tall, with a base diameter of 1.4 m. Electrical energy is provided by 6 windmill-type solar panels, producing up to 1 kW of power. A liquid propellant attitude control and orbital correction configuration maintains satellite stability and performs orbital maneuvers, although the latter usage is rarely needed. Sun and Earth sensors are used to determine proper spacecraft attitude and antenna pointing. The first Molniya 3 spacecraft appeared in 1974, primarily to support civil communications (domestic and international), with a slightly enhanced electrical power system and a communications payload of three 6/4 GHz transponders with power outputs of 40/80 W.

The second stratum of the Russian space-based communications system consists of 16 HEO Molniya-class spacecraft in highly inclined 63° semi-synchronous orbit planes, which initial perigees between 450 and 600 km fixed deep in the Southern Hemisphere and apogees near 40,000 km in the Northern Hemisphere. In fact, Molniya satellites are synchronized with the Earth's rotation, making two complete revolutions each day with orbital period of 718 min. The laws of orbital mechanics dictate that the spacecraft orbital velocity is greatly reduced near apogee, allowing broad visibility of the Northern Hemisphere for periods up to eight hours at a time. Thus, by carefully spacing 3 or 4 Molniya spacecraft, continuous communications can be maintained. This type of orbit was pioneered by the USSR and is particularly suited to high-latitude regions, which are difficult or impossible to service with GEO satellites.

The 16 operational Molniya satellites are divided into two types and four distinct groups. Namely, eight Molniya 1 satellites were divided into two constellations of four vehicles each. Both constellations consist of four orbital planes spaced 90° apart, but the ascending node of one constellation is shifted 90° from the other; that is, the Eastern Hemisphere ascending nodes are approximately 65° and 155° E, respectively. Although the system was designed to support the Russian Orbita TV network, a principal function was to service government and military communications traffic via a single 40 W 1.0/0.8 GHz satellite transponder.

The hypothetical Russian Molniya network can employ minimum 3 HEO satellites in three 12-h orbits separated by 120° around the Earth, with apogee distance at 39,354 km and perigee at 1000 km. This orbit takes the name from the communication system installed by the former USSR, whose territories are situated in the Northern Hemisphere at high latitudes. The orbital period (t) is equal to $(t_E / 2)$, or about 12 h. The characteristics of an example Molniya orbit are given in Table 2.4.

Table 2.4 Molniya and Tundra orbit parameters

Characteristics	Molniya orbit	Tundra orbit
Orbital period (t)	12 h	24 h
Sidereal period	11 h 58 min 2 s (half day)	23 h 56 m 4 s (full day)
Semi-major axis (a)	26,556 km	42,164 km
Inclination (i)	63.4°	63.4°
Eccentricity (e)	0.6–0.75	0.25–0.4
Perigee altitude (h_p) (e.g.,: $e = 0.71$)	$a(1 - e) - R$ 1250 km	$a(1 - e) - R$ 25,231 km
Apogee altitude (h_a) (e.g.,: $e = 0.71$)	$a(1 + e) - R$ 39,105 km	$a(1 + e) - R$ 46,340 km

The only one-track cycles of a total of two satellite tracks on the surface of the Earth is shown in Fig. 2.11 (c-1) for a perigee argument equal to 270° . The shape of this track is cycles of one orbit only near Greenwich Meridian, so the center of the next identical track is around 180° westward. Therefore, the satellite at apogee passes successively on each orbit above two points separated by 180° in longitude. The apogee is situated above regions of 63° latitude (the altitude of the vertex is equal to the value of the inclination and the apogee coincides with the vertex of the track when the argument of the perigee is equal to 270°). The large ellipticity of the orbit results in a transit time for the period of the orbit situated in the Northern Hemisphere greater than that in the Southern Hemisphere. The value of inclination, which makes the drift of the argument of the perigee equal to zero, is 63.45° . A value different from this leads to a drift, which is nonzero but remains small for value of inclination, which does not deviate too greatly from the nominal value. By the way of example, for an inclination $i = 65^\circ$, that is variation of 1.55° , the drift of argument of the perigee has a value of around 6.5° per annum.

It is evident that the Molniya HEO satellite has the advantage of high-elevation-angle coverage of the Northern Hemisphere because of a need to completely cover a great part of the Russian territory. Three satellites in this orbit and phasing are chosen so that at least one satellite is available at any time over the horizon. Thus, with three satellites, each satellite is used (or handover is) 8 h per day, while with four satellites handover is every 6 h. The LES must use tracking antenna systems, so a terminal with only one antenna will have an outage during handover (switching) from one satellite to another.

The disadvantages of the Molniya orbit include the need for multiple satellites (which the system does not need), the poor, virtually useless coverage of the Southern Hemisphere and the need for tracking antennas at each LES. Since the distance from terminal to satellite is continually changing, the received power and frequency vary (Doppler effect). The former may require automatic uplink power control and scheduling is needed to allow LES to switch satellites simultaneously. As the satellite altitude varies, the beam also coverage changes, so the satellite carries a tracking antenna that must be kept continuously pointed at operating LES.

2.3.3.2 Tundra Orbit

The Russian Tundra HEO system employs 2 satellites in two 24-h orbits separated by 180° around the Earth, with apogee distance at 53,622 km and perigee at 17,951 km, which provides visibility duration of more than 12 h with high-elevation angles. The Tundra orbit can be useful for regional coverage for both FSS and MSS applications. Similar to the Molniya orbit, this orbit is particularly useful for LMSS, where the masking effects caused by surrounding obstacles and multiple path are pronounced at low-elevation angles, ($>30^\circ$).

The period (t) of the orbit is equal to t_E , which is around 24 h. The characteristics of an example orbit of this type are given in Table 2.4. This orbit has only one track on the Earth's surface, as shown in Fig. 2.11 (c-2), for a perigee argument equal to 270° , inclination $i = 63.4^\circ$ and eccentricity $e = 0.35$. The latter parameter can have three values of eccentricity $e = 15$, $e = 25$, and $e = 45$.

According to the value of orbital eccentricity, the loop above the Northern Hemisphere is accentuated to a greater or lesser extent. For eccentricity equal to zero, the track has a form of Fig. 8, with loops of the same size and symmetrical with respect to the equator. When the eccentricity increases, the upper loop decreases, while the lower loop increases and the crossover point of the track is displaced toward the North. This loop disappears for a value of eccentricity of the order of $e = 0.37$ and the lower loop becomes its maximum size. The transit time of the loop represents a substantial part of the orbital period and varies with the eccentricity. The position of the loop can be displaced toward the East or West, with respect to the point of maximum latitude, by changing the value of argument of the perigee (ω) and the eccentricity.

2.3.3.3 Loopus Orbit

The proposed Loopus system, which employs 3 satellites in three 8-h orbits separated by 120° around the Earth, has an apogee distance at 39,117 km and perigee at 1238 km. This orbit has similar advantages and disadvantages as for the Molniya orbit. One of the problems encountered by the LES is that of repointing the antenna during the handover (changeover) from one satellite to another. With orbits whose track contains a loop, it is possible to use only the loop as the useful part of the track in the trajectory. Handover between two satellites is performed at the crossover point of the track. At this instant, the two satellites are seen from the LES in exactly the same direction and it is not necessary to repoint the antenna. To achieve continuous coverage of the region situated under the loop, the transmit time of the loop must be a submultiple of the orbit period and the number of satellites. Hence, the coverage can be extended to one part of the hemisphere by increasing the number of satellites in orbit regularly spaced about the globe.

2.3.4 Polar Earth Orbits (PEO)

The PEO constellation is today a synonym for providing coverage of both polar regions for different types of meteorological observation and satellite determination services. Namely, a satellite in this orbit travels its course over the geographical North and South Poles and will effectively follow a line of longitude. Certainly, this orbit may be virtually circular or elliptical depending upon requirements of the program and is inclined at about 90° to the equatorial plane, covering both poles. The orbit is fixed in space while the Earth rotates underneath and consequently, the satellite, over a number of orbits determined by its specific orbit line, will pass over any given point on the Earth's surface. Therefore, a single satellite in a PEO provides in principle coverage to the entire globe, although there are long periods during which the satellite is out of view of a particular ground station. Accessibility can of course be improved by deploying more than one satellite in different orbital planes. If two PEO satellite orbits are spaced at 90° to each other, the time between satellites passes over any given point will be halved, which orbit is shown in Fig. 2.11 (A).

The PEO system is rarely used for communication purposes because the satellite is in view of a specific point on the Earth's surface for only a short period of time. Any complex steerable antenna systems would also need to follow the satellite as it passes overhead. At any rate, this satellite orbit may well be acceptable for a processing store-and-forward type of communications system and for satellite determination and navigation.

There are four primary requirements for PEO systems as follows:

- (1) To provide total global satellite visibility for worldwide LEOSAR COSPAS-SARSAT distress and safety satellite beacons EPIRB, PLB, and ELT applications;
- (2) To provide global continuous coverage for current or newly developed and forthcoming satellite navigation systems;
- (3) To provide at L-band or any convenient spectrum the communication requirements of ships and aircraft in the polar regions not covered by the Inmarsat system; and
- (4) To provide global coverage for meteorological and synoptic observation stations.

The Inmarsat team has studied two broad ranges of orbit altitude of PEO for both distress and communication purposes: first, low altitudes up to 1400 km and second, high altitudes above 11,000 km. In reality, these two orbit ranges are separated by the Inner Van Allen radiation belt. In the regions of the radiation belt, the radiation level increases roughly exponentially with height at around 1000 km, reaching a peak at about 5000 km altitude. Therefore, a critical requirement to reduce high-energy proton damage to the solar cell arrays of the satellite system constrains the PEO to low and high altitudes. As is evident, another Outer Van Allen Belt has no negative influence on these two PEO constellations because it lies far away between MEO and GEO satellite planes.

These two specific systems studied by Inmarsat are COSPAS–SARSAT low PEO at 1000 km altitude and high PEO at 12,000 km altitude, similar to that studied by ERNO, named SERES (Search and Rescue Satellite) system. Thus, it is considered that these two systems demonstrate clearly the solution tradeoff and constraints on a joint PEO distress, SAR, and communication mission. Other possible orbits for polar coverage can be an inclined HEO Molniya constellation of four satellites; GIO 45° inclined orbit of four satellites and 55° inclined circular MEO at 20,000 km altitude for GPS and GLONASS satellite navigation systems. In the meantime, the COSPAS–SARSAT system has developed a special GEOSAR system using three GEO satellites for global distress communications satellite beacons in combination with already-developed LEOSAR systems using four PEO satellites.

For both the low and high PEO systems, the number of operational satellites required to provide adequate Earth coverage needs to be minimized in order to achieve minimum system costs. An IMO and ICAO requirement for the GMDSS/COSPAS–SARSAT mission is that there should be no time delay in distress alerting anywhere in the globe.

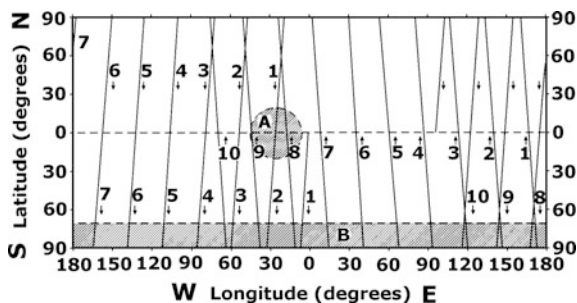
2.3.4.1 Low PEO

The low PEO satellite constellation similar to the LEO satellite constellation mostly employs both polar and near-polar orbits for communications and navigation utilities. Thus, a particular example of a system that uses this type of orbit is the COSPAS–SARSAT SAR system for maritime, land, and aeronautical applications. This system uses 8 satellites in 4 near-polar orbits: four US-based SARSAT satellite constellations at 860 km orbits, inclined at 99°, which makes them Sun-synchronous and four COSPAS satellite configurations at 1000 km orbits, inclined at 82°. However, this orbit was also suitable for the first satellite navigation systems Transit and Cicada, developed by the USA and the former USSR, respectively.

Otherwise, with a limited number of low-altitude PEO satellites, it is impossible to provide continuous coverage to polar region, because the view of individual spacecraft is relatively small and their transit time is short. However, because the time for a single orbit is low, less than two hours, and a different section of the polar region is covered at each orbit due to Earth rotation, this drawback is somewhat offset. For a given number of satellites, preferably about eight, it is possible to optimize the constellation that maximizes total system coverage, to improve hand-over and minimize waiting time between transits. This problem can be solved with additional GES terminals over pole area.

Figure 2.13 illustrates the Earth track of ten successive orbits of satellite in low PEO with an altitude of 1000 km. The MES in shaded area A (4200 km in diameter) would see the satellite, in the absence of environmental screening, at an angle of elevation not less than 10°, while the satellite was passing through the equatorial plane. The coverage area has the same size and shape wherever the

Fig. 2.13 Type of satellite orbits and tracks—Courtesy of Book: by Pascall



satellite is in the orbit but its apparent size and shape would change with latitude, being distorted by the map projection used in the figure. Thus, the South Pole coverage area at a single pass of the satellite is shown in figure by shaded area B.

The same figure shows that a single PEO satellite in a polar orbit will have a brief sighting of every part of the Earth's surface everyday. There will be 2 or 3 of these glimpses per day near the equator, the number increasing as the poles approach.

The period of visibility was seen from the MES range from about 10 min, the satellite passing overhead, down to a few seconds when the satellite appears briefly above the horizon. If the orbital plane of the satellite is given an angle of inclination differing from 90° of the PEO, a similar Earth track is obtained but the geographical distribution of the satellite visibility changes. One LEO satellite with an orbital inclination of 50° would have better visibility between 60° N and 60° S latitude than a PEO satellite but it would have no visibility at all of the polar regions.

The low PEO configuration is attractive for mobile distress communications for two reasons. First, the transmission path loss is relatively low, allowing reliable communication with a low-powered satellite beacon and PEO spacecraft. An altitude of about 1000 km is the upper limit for good reception of signals at 243/406 MHz sent from emergency distress beacons. Secondly, the Doppler shift is high, approximately 30 kHz at 1.6 GHz, allowing accurate location of the distress transmitter. On the other hand, there are several significant disadvantages. However, as mentioned earlier, PEO coverage is not continuous unless there is simultaneous communication between a distress buoy and a ground terminal because of the small footprint of each individual satellite. Accordingly, storage and retransmission of distress messages onboard processing would be necessarily adding to the distress alert delay time and also to satellite mass and complexity.

The short visibility period during a transit and the uneconomic need for large numbers of satellites for continuous coverage makes a low PEO unattractive for communications considerations. If this orbit configured well as an economic solution for distress coverage in polar regions to be used for communications purposes, users would have to operate with the following restrictions: (1) only burst mode, non-simultaneous data communication would be possible; (2) transmission time and/or bit rate would be limited by satellite message storage capability;

(3) replies to the message would require an interrogation or polling system from the MES expecting a reply; and (4) depending on the PEO constellation and MES position, a reply could take some hours.

However, many of these PEO communication limitations would be removed if a system of intersatellite links, possibly in addition to inter-GEO infrastructure, were used to provide a near-continuous, simultaneous two-way communication system. The complexity and likely cost of such system would almost certainly not be justified by the expected low level of polar communication traffic. Thus, in considering the possible integration of PEO and GEO for communication purposes, it is necessary to determine the additional requirements and constraints arising from polar operation. In this context, for reliable communications the number of additional LES required for operation to PEO is a significant element of the overall system. For example, a constellation of eight low PEO would require about six LES worldwide for polar coverage assuming message storing and forwarding techniques, where a high PEO would require a minimum of two LES located in North and South polar latitudes for continuous polar coverage with simultaneous two-way communications. In addition, it would be necessary to obtain reliable terrestrial links between the LES of each system, as well as intersatellite links between the PEO and the GEO satellites.

In any case, by using the store and transmit method, a low PEO system could effectively be served for the relay of mobile distress, safety, and urgency messages, for maritime, land, and aeronautical applications via satellite beacons to receive-only terminals onshore.

2.3.4.2 High PEO

The high PEO constellation would consist of three satellites separated by 120° in the same circular orbit of 12,000 km altitude, geometrically similar to the GEO and as orbit similar to MEO configuration. This orbit can provide continuous coverage to both polar regions above 59° latitude. In such a way, six satellites (in two orbital planes of three satellites each) would provide continuous and real global coverage if that was required, which GEO constellation alone cannot obtain.

By comparison with low PEO systems transmission path losses are higher at an altitude of 12,000 km but not to the extent that a distress beacon need be especially high powered to transmit successfully to a high PEO satellite. Reception of the COSPAS-SARSAT existing two very low-powered distress frequencies will be interfered, but not impossible. The Doppler shift is lower (about 10 kHz at 1.6 GHz), not allowing very accurate area location of the distress transmitters. Single high-latitude LES in both Arctic and Antarctic polar regions allows reception with no delay of all distress messages transmitted from above 59° latitude. Furthermore, using these two LES positioned at high latitude with continuous visibility of at least one of the three satellites and collocated or linked with an Inmarsat LES, can offer a full range of near continuous communication services to the polar regions.

2.3.5 Hybrid Satellite Orbits (HSO)

The hybrid satellite constellation can be configured by several types of combinations between existing orbital solutions today. Namely, any of these combinations can provide better global coverage for both hemispheres, including both polar regions. In this context will be introduced shortly five hybrid constellation systems, which are currently using or developing MSC and navigation systems as follows:

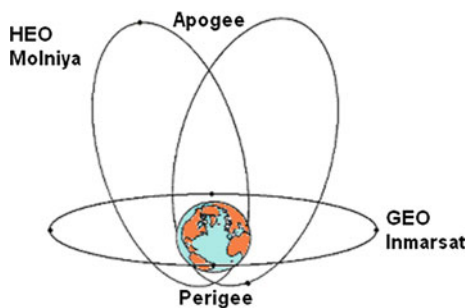
1. Combination of GEO and HEO Constellations—The development of a MSS that would provide reliable communications with MES terminals, such as oceangoing vessels, land mobiles (road vehicles and trains) and aircraft, rural areas and remote terminals, illustrated in Fig. 2.14. This MSC system, called Marathon, includes five GEO Arcos-type satellites and four Mayak-type satellites in a HEO, as well as a ground segment that is composed of base stations and terminals installed at fixed or mobile users premises.

This MSC system, called Marathon, includes five GEO Arcos-type satellites and four Mayak-type satellites in a HEO, as well as a ground segment that is composed of base stations and terminals installed at fixed or mobile users premises. Therefore, the combination of GEO and non-GEO satellite constellations makes it possible to render GMSC services, including those at high latitudes and in the both polar areas; this is especially important for Russia, with its vast northern Eurasian territories and to provide the most reliable satellite communication between the territories of the Western and Eastern Hemispheres.

This hybrid constellation can be useful for the Alaska, Greenland, and northern territory of Canada as well. The similar hybrid constellation with 2 HEO satellites can be configured for complete coverage of Southern Hemisphere with apogees on opposite side as shown in the same figure. This HSO configuration is the best solution for providing complete global coverage for aeronautical applications.

2. Combination of GEO and PEO Constellations—This current combination of orbits has been developed by the efforts of the COSPAS–SARSAT organization, with the assistance of IMO, Inmarsat, and other international and regional contributors. At the other words, the COSPAS–SARSAT space segment is a combination of three GEO operational satellites of the subsystem called GEOSAR and four PEO operational satellites of the subsystem called LEOSAR, with spare

Fig. 2.14 Combination of GEO–HEO constellations—
Courtesy of Paper: by Ilcev



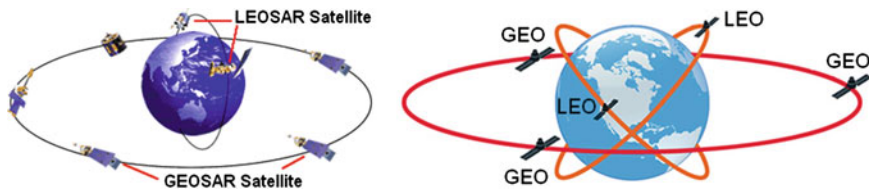


Fig. 2.15 Combination of GEO-PEO and GEO-LEO constellations—Courtesy of Paper: by Ilcev

spacecraft for all participants. The GEOSAR employs one satellite type of INSAT-2A and two GOES-type GOES-E and GOES-W, while the LEOSAR configuration provides two COSPAS and two SARSAT spacecrafts. Otherwise, the GEOSAR project in the future has to include the European MSC and two Russian Luch-M spacecraft. This system is responsible for providing distress alert via special radio beacons (EPIRB, PLB and ELT) and to help SAR forces on-scene determination for maritime, land, and aeronautical applications, in which constellation is shown in Fig. 2.15 (Left).

3. Combination of GEO and LEO Constellations—Celestri is the Motorola trademark name for a proposed GEO and LEO satellite hybrid communication network, shown in Fig. 2.15 (Right). The network will combine 9 GEO and 63 LEO satellites in 7 planes with Earth-based control equipment and provide interfaces to existing telecommunication infrastructures, the Internet, and corporate and personal networks. The system will offer a 64 Kb/s voice circuit from anywhere in the world. The architecture is not limited to fixed-sized channels but permits dynamic bandwidth assignment based on application demand. Business users will benefit by Celestri's to provide remote access to LAN infrastructures.

4. Combination of MEO and HEO Constellations—The newly proposed MSS Ellipso is developing in combination with an initial complement of seven Concordia satellites deployed in a circular equatorial MEO at an altitude of 8050 km and ten Borealis satellites in two HEO planes inclined at 116.6° . They have apogees of 7605 km and perigees of 633 km and a three-hour orbital period.

This combination of two constellations, shown in Fig. 2.16 (Left), would provide coverage of the entire Northern Hemisphere including North Pole areas and part of the Southern Hemisphere up to 50° latitude South. The HEO satellites can spend a greater proportion of their orbital periods over the northern latitudes and, together with the MEO constellation, the Ellipso hybrid system will provide voice, data, and Fax communication and navigation RDSS services to areas with large landmasses, enormous populations with a large density of users and potentially widespread markets. This system is also planned to cooperate with the terrestrial PSTN and other services.

5. Combination of MEO and LEO Constellations—The Kompomash consortium for space systems in Russia have prepared the Gostelesat satellite system for MSS, shown in Fig. 2.16 (Right), using 24 satellites in MEO and 91 in LEO

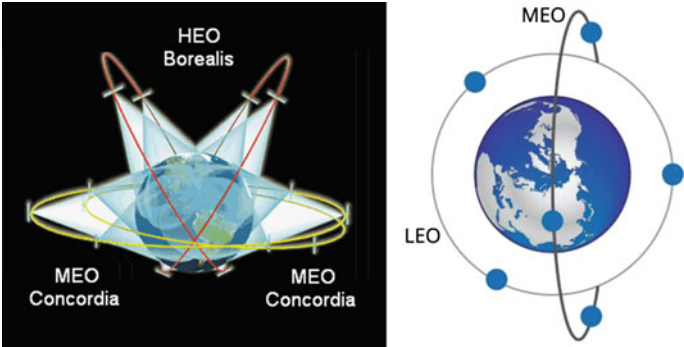


Fig. 2.16 Combination of MEO and HEO and MEO and LEO constellations—Courtesy of Paper: by Ilcev

satellite constellation. Thus, this satellite project is provided for future global MSC and navigation applications with possibility to cover both polar regions.

2.4 Spacecraft Subsystems

A communications satellite essentially consists of two major functional units: payload and bus. The primary function of the payload is to provide communication between LES and MES, while the bus provides all the necessary electrical and mechanical support to the payload and all satellite missions illustrated in Fig. 2.17

The payload is made up of the multipurpose repeaters and antenna systems. The repeater performs the required processing of the signal, and the antenna system is used to receive signals from LES and to transmit signals to MES in the coverage area and vice versa.

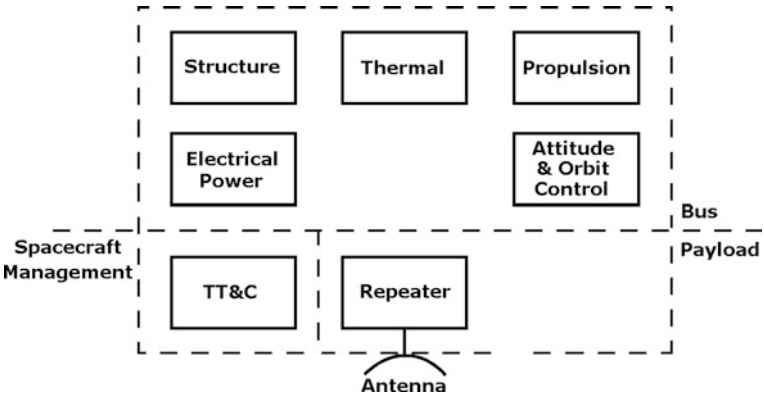


Fig. 2.17 Spacecraft subsystem—Courtesy of Book: by Richharia

2.4.1 Satellite Repeaters for Mobile Satellite Communications

The function of a satellite repeater is to receive the uplink RF signals from either ground segment service or feeder links, then to convert these signals to the appropriate downlink frequency and power, and to retransmit them toward the service or feeder links ground segment. Two main types of repeaters are possible for onboard utilization: Transparent and regenerative transponders, however, are developed many other types for different satellite applications.

2.4.1.1 Transparent or Bent-Pipe Transponder

The basic function of the satellite transponder is to isolate individual carriers or groups of carriers of signals and to boost their power level before they are retransmitted to the ground stations. The carrier frequencies are also altered as the carriers pass through the satellite. Satellite repeaters that process the carrier in this way are typically referred to as transparent or bent-pipe satellite transponders, illustrated in Fig. 2.18. Only the basic RF characteristics of the carrier (amplitude and frequency) are altered by the satellite. The detailed signal carrier format, such as the modulation characteristics and the spectral shape, remains completely unchanged.

Transmission via a transparent satellite transponder is often likened to a bent-pipe because the satellite simply channels the information back to the ground stations. A bent-pipe is a commonly used satellite link when the satellite transponder simply converts the uplink RF into a downlink RF, with its power

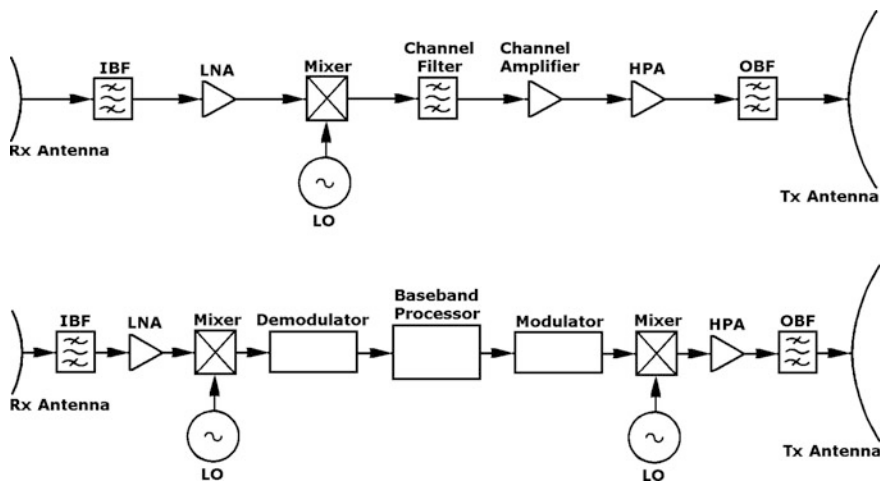


Fig. 2.18 Configuration of spacecraft Transponders—Courtesy of Handbook: by ITU

amplification. Initially, the received uplink signals from LES or MES by Rx antenna are filtered in an Input Bandpass Filter (IBF) prior to amplification in a low-noise amplifier (LNA).

In addition, the output of the LNA is then fed into a local oscillator (LO), which performs the required frequency shift from uplink to downlink RF and the bandpass channel filter after the mixer removes unwanted image frequencies resulting in the downconversion, prior to undergoing two amplification stages of signals in the channel and high-power amplifier (HPA). Finally, the output signal of the HPA is then filtered in the output bandpass filter (OBF) prior to transmission through Tx antenna to the ground. The IFB is a bandpass filter which blocks out all other RF used in satellite communications. After that, the receiver converts the incoming signal into a lower frequency, using an LO which is controlled to provide a very stable frequency source. This is needed to reduce all noises to facilitate processing of the incoming signal and to enable the downlink frequencies to be established.

The channel filter isolates the various communications channels contained in the waveband allowed through by the input filter. Filtering often leads to large power losses, creating a need for extra amplification, usually followed by a main amplifier. In order to attain the required gain of HPA, this segment may employ either a solid-state power amplifier (SSPA) or a traveling wave tube amplifier (TWTA). In a more complex transponder design, in order to achieve higher RF power, it may be possible to combine the output of several amplifiers.

Therefore, to do this the incoming signal must be divided in such a way so as to provide separate identical input to each amplifier, see 6 TWTA presented in Fig. 2.19. A power combiner then recombines the RF signals from the amplifiers to produce a single RF output. The output filter removes all unwanted signals from the transmitted downlink returning to the Earth stations. High reliability throughout the lifetime of the satellite is achieved by duplicating critical units in the receiver, such as TWTA.

2.4.1.2 Regenerative Transponder

Other satellite system designs go through a more complex onboard process to manipulate the carrier's formats, by using onboard processing architecture. This payload architecture offers advantages over the transparent alternative, including improved transmission quality and the prospect of compact and inexpensive MES and handheld user terminals. A typical onboard processing system will implement some or all of the functions that are performed by the ground-based transmitter and/or receiver in a transparent satellite system.

Therefore, these functions may include recovery of the original information on board the satellite and the processing of this information into a different carrier format for transmission to the ground stations. In fact, any satellite transponder that recreates the signals carrier in this way is usually referred to as a regenerative transponder, illustrated in Fig. 2.18. This type of satellite transponder provides demodulation and modulation capacity completely on board the satellite.

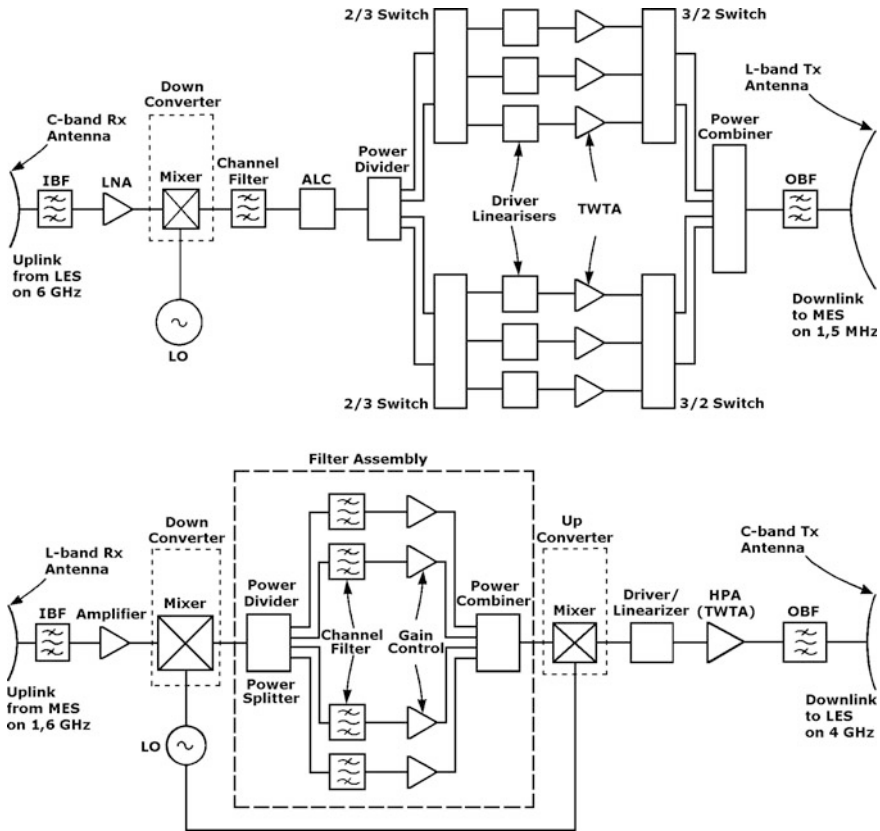


Fig. 2.19 Diagram of spacecraft C/L and L/C-band Transponders—Courtesy of Book: by Gallagher

The received uplink signal goes along the downconverter segment prior to coming into the onboard demodulator, where it is demodulated and processed in the base band processor.

This technology provides flexible functions, such as switching and routings. The downlink signal generated by an onboard modulator passes along the upconverter segment and is transmitted via the antenna. For this type of system link design can be separately conducted for the uplink and downlink because link degradation factors are decoupled between the uplink and the downlink by the onboard demodulator and modulator, supported by the base band processor.

A regenerative transponder with base band processing permits reformatting of data without limitation to MES Rx, while the bent-pipe system requires a satellite link design for the entire link, involving both uplink and downlink, but the forward link burst rate is limited by the MES G/T and demodulation performance. Advanced MSS include intersatellite links to establish a direct connection between

satellite transponders and in this way to enlarge system coverage and help the reception of signals from other GEO satellite coverage not visible for particular LES and MES terminals. Moreover, an intersatellite link can help to solve the problems associated with some TTN infrastructures or to reduce landline charges.

In the same way, intersatellite links can also provide a connection from a satellite to neighboring satellites in a constellation of non-GEO space segment and so, they are beneficial in reducing the number of satellite hops when Earth coverage of each satellite is limited. Intersatellite links are usually implemented with regenerative satellite transponder systems, mainly due to the flexible connection to intersatellite links.

2.4.1.3 Satellite Transponders for Inmarsat-2 MSC

A transponder as the nucleus of the Inmarsat network receives information from LES in a directed beam of energy at 6 GHz in the C-band and converted information broadcasts at 1.5 GHz in the L-band. The link from MES follows the reciprocal path. The Inmarsat-2 payload consists of two transparent transponders: the C/L-band, illustrated in Fig. 2.19, and the L/C-band, shown in Fig. 2.19.

1. Inmarsat-2 C/L-band Transponder—This transponder receives uplink signals in the C-band of 6.4 GHz from LES and retransmits downlink signals in the L-band of 1.5 GHz to MES, after frequency conversion and signal amplification by a HPA. The signals received by a C-band antenna are fed via IBF and LNA to a downconverter section. A signal channel is followed by an automatic level control (ALC) device, which limits the level of the signal to the amplifier. The HPA consists of six TWTA and their associated power supplies. In front of each TWTA is a driver/linearizer, predisposed to compensate the nonlinear RF properties of the TWTA. The signal driver supplies an equal drive signal to each of the four TWTAs that are active at any given time and the other two can be activated for backup if the operating TWTA malfunctions. For this reason, the signal driver is preceded by an amplitude equalizer. However, the active TWTA are selected by 2/3 and 3/2 switches and their output powers are combined by a power combiner. The total power is fed to an L-band transmission antenna via OBF.

2. Inmarsat-2 L/C-band Transponder—This transponder receives uplink signals in the L-band of 1.6 GHz from MES and retransmits downlink signals in the C-band of 3.6 GHz to LES, after frequency conversion and signal amplification by the HPA. The signals received by an L-band antenna are fed to a downconverter via IBF and LNA. At the down-converter, signals are converted into 60 MHz IF by LO. A filter assembly then provides the required characteristics divided into four channels. Following upconversion, the signal passes to an ALC unit and the power for four channels is combined and signals are upconverted from 60 to 3.8 GHz by activated TWTA. The amplified signal in HPA then goes through bandpass and harmonic filters in OBF before being distributed among the 7 cup-dipole elements of the C-band transmit antenna for radiation to the Earth's surface.

2.4.2 Satellite Repeaters for COSPAS–SARSAT System

The COSPAS–SARSAT organization initially developed the first generation of LEO Search and Rescue (LEOSAR) systems, such as COSPAS and SARSAT LEO spacecraft. COSPAS–SARSAT is also employing four GEO satellites for its new GEO Search and Rescue (GEOSAR) 406.05 MHz mission, such as the Russian Luch-M, the US Geostationary Operational Environmental Satellite (GOES), Indian National Satellite System (INSAT), and the European Meteosat Second Generation (MSG). In the following context will be introduced Luch-M and GOES transponders of the GEOSAR subsystem including COSPAS transponder and SARSAT Rx/Tx configurations of the LEOSAR subsystem.

2.4.2.1 COSPAS GEOSAR Luch-M Transponder

The Luch-M spacecraft is developed by Russia and is serving for COSPAS–SARSAT and also for communication applications. The COSPAS GEO Luch-M East bird is at 95°E covering Indian Ocean and Luch-M West at 16°W is covering Pacific Ocean.

A functional diagram of the Luch-M GEOSAR transponder is shown in Fig. 2.20. The repeater is redundantly configured and is comprised of the following subsystems:

- The UHF 406 MHz receive antenna, LNA, and IF1 Amplifier as main components of SAR 406 MHz Receiver (Rx);
- The IF1 Amplifier, Power Amplifier (PA), and SHF 11,381 MHz transmit antenna as main parts of SAR 11,381 MHz Transmitter (Tx); and
- The intermediate components of Rx and Tx are local oscillator and synthesizer.

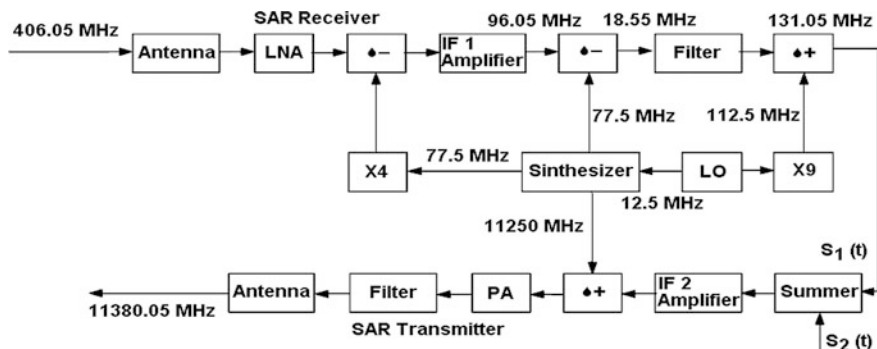


Fig. 2.20 Diagram of COSPAS GEOSAR Luch-M Transponder—Courtesy of Manual: by COSPAS–SARSAT

The 406.05 MHz signals from COSPAS–SARSAT distress beacons are received by the Luch-M UHF antenna. The signal is downconverted twice to an Intermediate Frequency (IF) of 18.55 MHz after which it is filtered. The 3 dB beamwidth of this filter is 600 kHz. This IF signal ($S1(t)$) is upconverted and then combined with signals from other instruments onboard the satellite ($S2(t)$). This composite signal is amplified, upconverted to 11381.05 MHz and amplified to a power of 3.75 W. The composite amplified signal is then filtered before being transmitted via the satellite .6 M parabolic antenna.

2.4.2.2 SARSAT GEOSAR GOES Transponder

The GOES spacecraft is developed by the USA and is serving for COSPAS–SARSAT and also for meteorological applications. The SARSAT GEO GOES East bird is at 75°W covering Atlantic Ocean and GOES West at 135°W is covering almost entire Pacific Ocean.

A functional diagram of the GOES SAR repeater is illustrated in Fig. 2.21. The repeater is redundantly configured and consists of the following units:

- One UHF 406 MHz receive antenna, Switch (SW) dividing two 406 MHz LNA shared with another satellite subsystem and two dual-conversion 406 MHz receivers;
- Two 3 W (watt) phase-modulated L-band transmitters synchronously switched and one SHF 1544.5 MHz transmit antenna; and
- Telemetry and Command (T&C) points interfaced with the spacecraft Command and Telemetry subsystem.

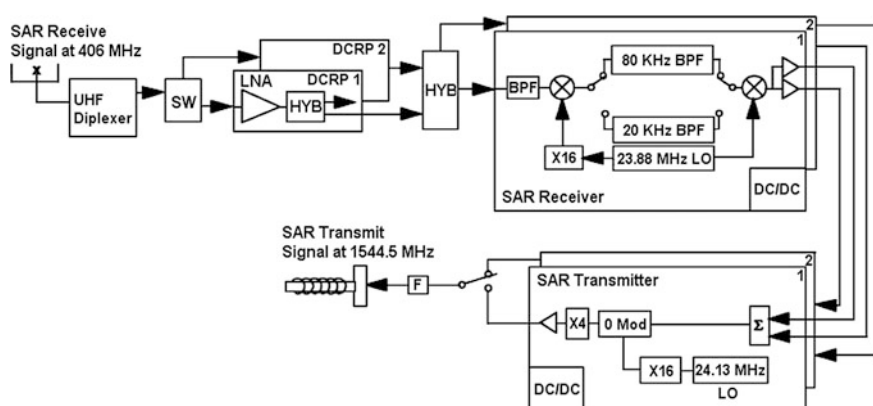


Fig. 2.21 Diagram of SARSAT GEOSAR GOES Transponders—Courtesy of Manual: by COSPAS–SARSAT

The 406 MHz SAR signals from COSPAS–SARSAT distress beacons are received on the UHF antenna and fed through the antenna diplexer and switch to an LNA module in one of the redundant pairs of Data Collection Platform Repeaters (DCPR). The DCPR LNA modules are used as a part of the SAR implementation to accommodate circuit efficiency on the spacecraft.

The LNA module outputs are connected to the redundant pair of SAR receivers. The signal applied to the selected receiver is downconverted for bandpass filtering in accordance with one of two commandable bandwidth modes; a narrow band mode of 20 kHz or a wide band mode of 80 kHz. The filtered output signal is further downconverted to near baseband and fed through amplifiers to the SAR transmitter. However, the overall gain of the SAR receiver can be command-selected into a fixed gain or ALC mode.

The outputs of the receivers are provided to the redundant pair of SAR transmitters. The selected SAR transmitter phase modulates the signal, multiplies the signal to 1544.5 MHz, and amplifies the modulated carrier to 3 W. The phase-modulated signal has the nominal modulation index set such that the carrier suppression is 3 dB with the receiver in the ALC mode or with the receiver in the fixed gain mode operating with two nominal beacon signals plus the noise. A baseband limiter restricts the modulation index from exceeding 2 radians. The transmitter output is applied through a 4 MHz bandwidth filter to the helical antenna and radiated with an EIRP of +15.0 dBW.

The GOES repeater has redundant LNA modules, receivers, and transmitters that can be selected to define a complete repeater configuration. At this point, a specific repeater configuration can be operated in the modes described in Table 2.5.

ALC mode is concerning two equal test tones each at 7 dB above the receiver noise applied to the receiver input will not produce intermodulation products within the transponder bandwidth greater than 30 dB below the test tone output level.

The GOES SAR Receiver and Transmitter parameters are shown in Table 2.6.

Note 1 in Table of Rx Parameters is nominal input level at antenna from 5 W beacons located at $45\pm$ Elevation angle to the satellite, which includes 4.1 dB polarization losses.

Note 2 in Table of Tx Parameters is concerning Fixed Gain Mode, which is presenting two equal test tones each at 2 dB above the Rx noise applied to the repeater receiver input will not produce intermodulation products within the transponder bandwidth greater than 20 dB below the test tone output level.

Table 2.5 GOES repeater operating mode

Mode	Band center frequency (MHz)	Receiver 3 dB bandwidth (kHz)
Narrow band with ALC	406.025	20
Narrow band fixed gain	406.025	20
Wide band with ALC	406.050	80
Wide band fixed gain	406.050	80

Table 2.6 GOES SAR receiver and transmitter parameters

Receiver (Rx) parameters	Unit	Values	Transmitter (Tx) parameters	Unit	Values
Nominal input level antenna ¹	dBW	−173.1	Center frequency	MHz	1544.5
System noise temperature	K	359	Output power transmitter	W	3.0
G/T	dB/K	−18.5	Repeater EIRP	dBW	+15.0
Rx bandpass characteristics: Narrow band mode (to 406.025 MHz)	KHz	±6.0(1 dB BW) ±10.0(3 dB BW) ±20.0(20 dB BW)	Phase Jitter (50 Hz Bandwidth)	deg. (rms)	≤10
Narrow band mode (to 406.05 MHz)	KHz	±30.0(1 dB BW) ±40.0(3 dB BW) ±50.0(20 dB BW)			
Dynamic range	dB	≤15	Modulation type	Type	Linear phase
Group delay (over 4 kHz) in KHz	μs	≤13	Tx Nominal Modulation Index Modulation Index Limit	Radians peak	1.0 2.0
Image rejection	dB	60	Downlink frequency stability	N/A	$\pm 2.5 \times 10^{-6}$
AGC time constant	ms	≤40	Amplitude ripple (over 24 h)	dB	±1
Frequency (RF) stability (<0.5 s) RF conversion oscillator	N/A	$\pm 1 \times 10^{-9}$	Linearity	N/A	See Note 2

2.4.2.3 COSPAS LEOSAR Transponder

The COSPAS LEOSAR payload system functional diagram is illustrated in Fig. 2.22.

The uplink signal from SARP and SARR receive antenna (SPA) is entering via filter and RF switch in one of two receivers. From each Rx, the 75 kHz signal by speed of 2.4 Kb/s is passing formatters, T&C, and entering in one of two Tx units. The downlink signal from the SAR L-band transmit antenna (SLA) can be detected by any COSPAS–SARSAT ground Local User Terminal (LUT) in the LEOSAR satellite system known as LEOLUT.

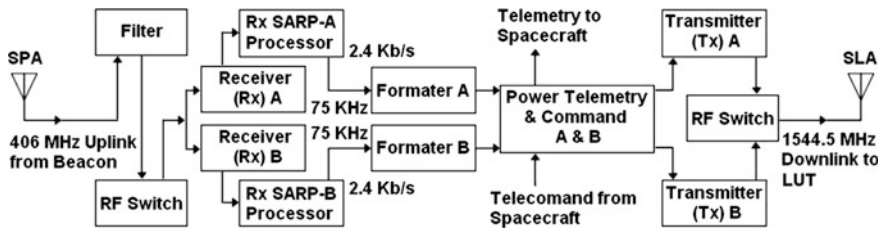


Fig. 2.22 Diagram of COSPAS LEOSAR Transponders—Courtesy of Manual: by COSPAS–SARSAT

The COSPAS LEOSAR payload of the COSPAS–SARSAT system is composed of:

- SAR Repeater (SARR) is getting SAR distress signals from ground beacons and without processing is retransmitting them to the LUT land infrastructures;
- SAR processor (SARP) is processing receiving SAR signals and retransmit them to the first-available LUT station; and
- Uplink UHF 406.05 MHz and downlink SHF 1544.5 MHz antennas.

The SARR provides local mode coverage for the 406 MHz band without processing distress and SAR signals. The SARP provides both local mode and global mode coverage for the UHF 406 MHz band via COSPAS satellites, which have an improved SARP with memory known as SARP-2, shown in Fig. 2.20 as Rx SARP A and B processors. Processed data are transmitted to the ground stations via the downlink transmitter. In addition, this transponder is also providing routine Telecommand (TC) from spacecraft and Telemetry (TM) to spacecraft for better control and management of spacecraft.

2.4.2.4 SARSAT LEOSAR Spacecraft Receiver and Transmitter

The LEOSAR receiver is an integrated component of SARR without processing elements and can be also integrated in SARP configuration with processing and formatting modes. As shown in Fig. 2.23, the two 406 MHz receivers are getting RF signal from beacons via antenna and via mixers is transforming it in IF signal. Each receiver contains automatic gain control (AGC) and provides two outputs to drive the two transmitters, while via mixers they share the same oscillator and are coherent in a ratio of 1:2.

As illustrated in Fig. 2.23, each one of the transmitters has four inputs in Baseband Summer: one for each of the two 406 MHz receivers, one for the Processed Data Stream (PDS) channel, and one spare port. The signals are then entering in Phase Modulator where the receiver input is illuminated by a sinusoidal

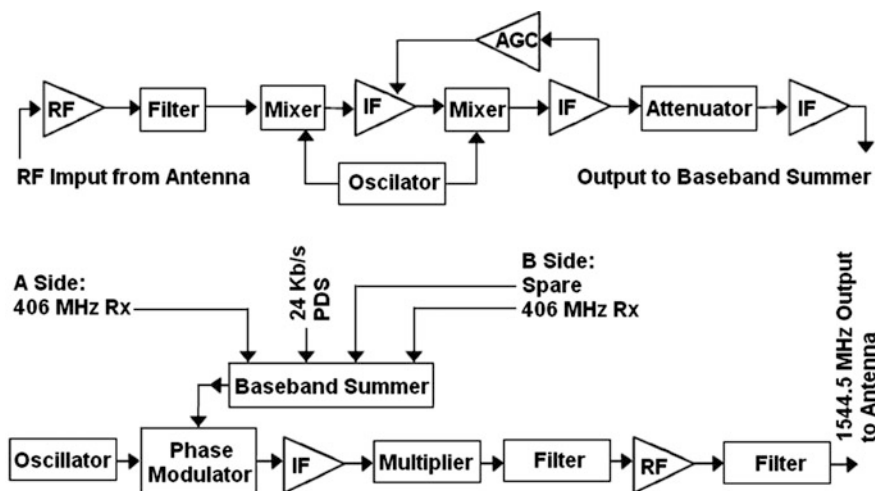


Fig. 2.23 Diagram of Inmarsat-3 transponders—Courtesy of Book: by Martin

signal from oscillator at the maximum frequency and level of IF. The IF signal is passing multiplier and filtering process and as RF is transmitted at 1554.5 MHz via antenna to Local User Terminal (LUT) on the ground.

2.4.3 Satellite Repeaters for New Generation of GEO and non-GEO MSC

The new generations of GEO and non-GEO satellites for MSC are developed by Inmarsat GEO satellite operator, while non-GEO satellites, such as Big LEO satellite constellations, are developed by Iridium and Globalstar, and Little LEO satellite constellation is developed by Orbcomm.

2.4.3.1 Inmarsat-3 GEO Satellite Communication Repeater

The Inmarsat-3 development was not paced by a lifetime limit of Inmarsat-2, but by its capacity limitation. The number of ships using Inmarsat increases every year, both by choice and gradually because of requirements that ships be equipped for MSC. In addition, since 1990, Inmarsat has provided service to airplanes as well as ships. Besides requiring increased satellite capacity, the airplanes also require more of the satellite power because of their smaller antennas.

Planning of the Inmarsat-3 series began about 1988. Proposals were received early in 1990; that summer, negotiations with a contractor began in parallel with a

5-month technology validation program. Thus, the technology program concentrated on the L-band multibeam antennas and the L-band transmitter power amplifier.

The final contract was signed early in 1991; it included four satellites and options for more. In March 1994, one option was changed to an order for a fifth satellite. The capacity increase relative to the foregoing Inmarsat satellites is achieved by the use of spot beams at the L-band transmit-and-receive frequencies used between the satellites and the mobile terminals. The main requirements on the Inmarsat-3 payload illustrated in Fig. 2.24 are as follows:

1. Reconfigurable spot beams so that each satellite can provide the desired coverage from any of the five operating locations;
2. Flexible allocation of the total L-band power among the spot beams and the global beam to adapt to changing traffic loading; and
3. Channelization matching the international L-band allocations and switchable to adapt to Inmarsat needs and to ensure compatibility with other L-band satellites.

The Inmarsat-3 payload consists of five sections. The C-band-to-L-band forward channel is for MSC from fixed terminals to mobiles and is matched to the L-band-to-C-band return channel for communications in the opposite direction. These two channels are the main part of the payload. There is also a C-band-to-C-band channel for administrative traffic between mobile terminals in limited circumstances such as during search and rescue efforts. The final section of the payload is a navigation channel that augments the US GPS and the Russian GLONASS system for new communication, navigation, and surveillance (CNS) service known as Global Satellite Augmentation System (GSAS). When Inmarsat-3 was developed, the L-band-to-L-band and navigation capabilities were new features in MSS.

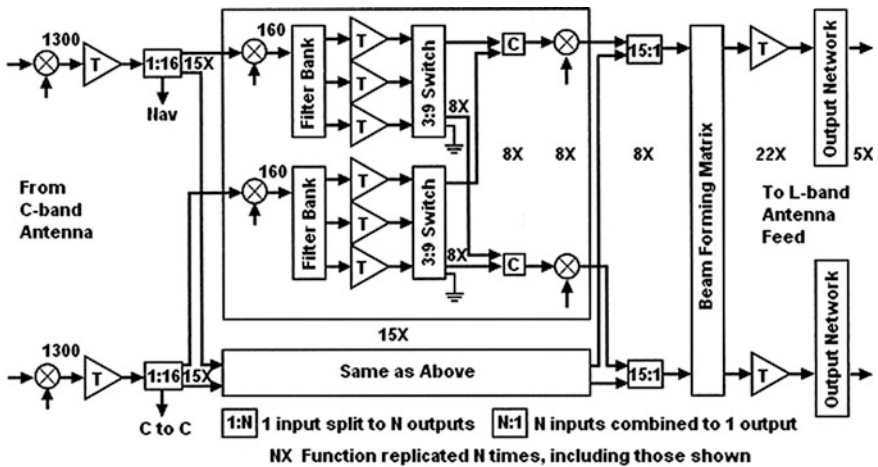


Fig. 2.24 Diagram of Inmarsat-3 Transponders—Courtesy of Book: by Martin

Inmarsat-3 spot beams in coverage area are quite small to allow frequency reuse and large enough to moderate the payload complex, which grows with the number of coverage areas. Many of these coverage areas have the same geometry at more than one location, or differ only by a rotation of the coverage areas about the subsatellite point. As a result, the Inmarsat-3 design has seven spot beams for the L-band transmission, each generated by four- or six-element subarrays of the 22-element feed array, while global coverage beams are used at C-band. The feed array can be rotated 21° about its axis to change the coverage geometry at the time of satellite relocation. Adjacent beams overlap, but some separated beam combinations have sufficient isolation to allow frequency reuse, namely to use of the same frequencies in two different beams.

The limited L-band spectrum available for use with mobile terminals is reused twice in the Inmarsat-3 design. When Inmarsat-3 was developed, the L-band allocation was subdivided for Maritime, Land, Aeronautical, and general mobile applications. To match this, the Inmarsat-3 frequency plan has 10 bands with 1–4.5 MHz bandwidths. All except the narrowest band have switchable Surface Acoustic Wave (SAW) filters that can divide the band into two or three sub-bands separated by 200 kHz guard bands for use in different antenna beams, or combine the sub-bands with the intervening guard bands for use in the same beam. The frequencies of the guard bands are different in the two reuses of the spectrum in order to maximize the ability to adjust spectrum usage. More recently, most of the allocations have been generalized to allow any application. Nevertheless, the Inmarsat-3 channelization is still good for efficient spectrum utilization.

The combination of many frequency bands and multiple satellite antenna beams would typically imply a very complex switching network on the satellite to provide flexible interconnections; this has been avoided by using a multiport amplifier. This amplifier is composed of a beam-forming matrix followed by an output filtering and routing network. The beam-forming matrix has eight inputs, corresponding to seven spot beams plus one global beam. The output network has 22 output ports, corresponding to the 22 antenna feeds. The combination of the matrix and the network allows the power of all 22 amplifiers to be routed to any one beam or to be distributed among many beams, both spot and global. The distribution of power is controlled by the signal phasing set in the beam-forming matrix. All the amplifiers operate at high efficiency regardless of the distribution of signals among the spot and global beams. Even if the total power is routed to one beam, it is spread over the four or six feeds that generate that beam, thereby avoiding excessive power at any one feed. Two global channels, and the L-band-to-L-band channel (when used), are always transmitted through global beams; all other channels in the L-band frequency plan may be routed to either spot or global beams.

2.4.3.2 Inmarsat-4 GEO Satellite Communication Repeater

The next question Inmarsat faced was whether to use a bent-pipe (transparent) transponder as on Inmarsat-3 before or to design a new regenerative payload on the

last generation of the Inmarsat-4 payloads. In the end, notwithstanding the better characteristics offered by a regenerative solution, Inmarsat decided to also implement its fourth-generation navigation payload as a bent-pipe repeater.

The company has negotiated European spacecraft manufacturer Astrium a 700 million US \$ contract to build three Inmarsat I-4 satellites. The contract was awarded in May 2000 with a planned service start sometimes in 2004. However, satellite development difficulties and slower than expected demand delayed the start of service to 2005. Astrium includes the former Matra Marconi Space, which built the Inmarsat-2 satellites and the payload for the Inmarsat-3s. Two Inmarsat-4 satellites were placed in two of Inmarsat's existing IOR and AOR-W orbits in 2005. The third Inmarsat-4 satellite was launched later to extend the coverage area so as to make it global, based on business and operational considerations. The satellite locations are subjected to review and may change.

The focus for the Inmarsat 4 satellite transponders is what Inmarsat calls the Broadband Global Area Network (BGAN) with communication speeds to 432 Kb/s for Internet access and multimedia transmissions, as well as traditional voice and data services. As a prelude, Inmarsat started Regional BGAN at the end of 2002 with transponder capacity leased from Thuraya (described below). This service has data rates to 144 Kb/s and is available in the coverage area of Thuraya spacecraft. The communication payload requires a powerful satellite platform, capable of providing the required DC power (10–12 kW) and other resources. Inmarsat 4, like prior Inmarsat generations, uses L-band for links between the satellite and large, fixed gateway stations. The satellite has a 9-m diameter L-band reflector used for both transmission and reception. This antenna consists of 120-element feed on the adjacent face of the satellite body, to form up to 228 spot beams that cover Earth. In addition, for compatibility with Inmarsat 3 satellites there is a global coverage beam and a 19-beam array of wide spots that cover Earth, both in L-band. The satellite also has a dual-polarization Earth coverage C-band antenna beam. Each satellite has the capacity of 630 channels each with a data rate up to 432 Kb/s. Onboard processing can dynamically reassign channels among the beams to satisfy user requests.

Inmarsat 4 has a two-channel navigation payload, each with wider bandwidth than the single channel on Inmarsat 3. Navigation data are transmitted to the satellite at C-band and transmitted from it on two frequencies. These signals augment GPS and other navigation satellite systems by transmitting integrity and correction data that can improve the accuracy of users' position determination. Other information about Inmarsat 4 is as follows.

2.4.3.3 Projected Inmarsat-5 GEO Satellite Communication Repeater

Inmarsat's first wholly owned satellites, the Inmarsat-2, were launched in the early 1990 s, and the Inmarsat-3 as the first generation to use spot beam technology followed later in the decade, both of these spacecrafts are spare or still operational. The new constellation of the Inmarsat-4 set a new benchmark for MSC in terms of

their power, capacity, and flexibility. In fact, one Inmarsat-4 satellite is 60 times more powerful than an Inmarsat-3 and its constellation series is expected to continue in commercial operation until about 2020.

In the meantime, Inmarsat has entered into agreement with ESA to become the commercial operator of a new satellite called Alphasat. The satellite is part of an ESA initiative to develop a new spacecraft platform capable of carrying onboard a large communications payload. Accordingly, Alphasat is scheduled for completion in 2012 and to supplement the existing I-4 satellites. It will provide service over Europe, the Middle East, and Africa.

On the other side, in 2010 Inmarsat satellite organization has contracted the US aerospace manufacturer Boeing to build a new constellation of Inmarsat-5 satellites as a part of a new 1.2 billion US \$ worldwide wireless broadband network called Inmarsat Global Xpress™. Boeing will construct three Inmarsat-5 (I-5) satellites based on its 702HP spacecraft platform. The first is scheduled for completion in 2013, with full global coverage expected by the end of 2014.

The spacecraft will break new ground by transmitting in a portion of the radio spectrum never before utilized by the commercial operator of a global satellite system and with the extremely high-frequency Ka-band transponder for the first time. Accordingly, each fifth generation of Inmarsat-5 will carry a payload of 89 Ka-band beams capable of flexing capacity across the globe and enabling Inmarsat to adapt to shifting subscriber usage patterns over their projected lifetime of 15 years.

2.4.3.4 Globalstar Big LEO Satellite Communication Repeater

The Globalstar satellite uses a simple transparent repeater rather than a digital onboard processor such as that being flown on Iridium. Figure 2.25 shows a block diagram of the Globalstar communication payload and a phased array antenna with 16 spot beams, which provides gain and a degree of frequency reuse. This spacecraft is a 3-axis stabilized platform with the Earth-facing panel always parallel to the orbit tangent. All transmissions are relayed at C-band (7/5 GHz) through GES terminals that track the satellites as they pass near by in LOS. However, this condition places a constraint that a given LEO satellite must simultaneously see both the mobile user at L/S-bands and the GES at C-band.

Through a relationship with Qualcomm, Inc., Globalstar provides service using of the CDMA cellular standard. A GPS onboard receiver is used to accurately determine the orbit parameters and also to supply accurate time and frequency to the satellite systems. The attitude control system uses small, one Newton, thrusters for onboard attitude control. At this point, yaw steering is employed to provide sufficient solar array power during all phases of mission. Solar panel and a large nickel-hydrogen battery provide power for all phases of the mission. However battery recharge takes place where satellite is moving over the oceans, there is less traffic with mobile users.

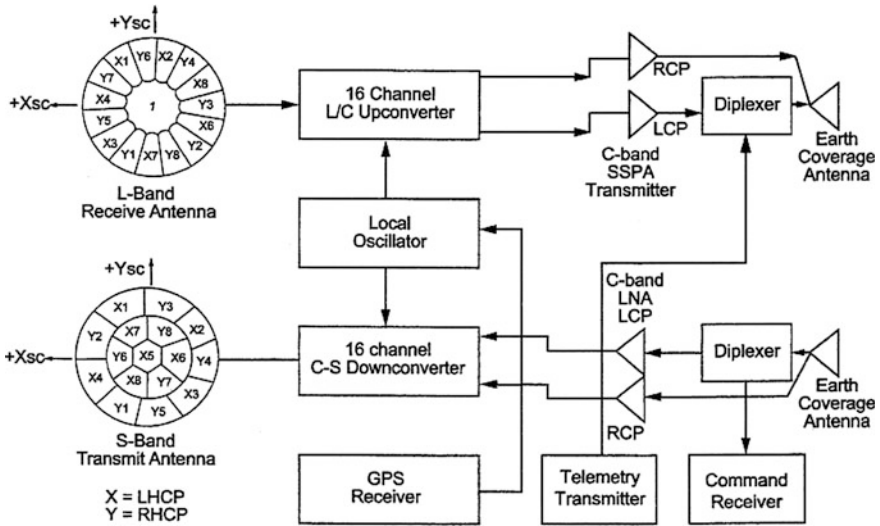


Fig. 2.25 Diagram of Globalstar spacecraft transponders—Courtesy of Paper: by Dietrich

2.4.3.5 Iridium Big LEO Satellite Communication Repeater

Iridium, such as Globalstar, as Big LEO satellite system, provides too much useless coverage of the ocean regions, resulting in less effective capacity over land areas than can be obtained with either a MEO or a GEO strategy. Without intersatellite links, LEO satellite cannot serve users out on the oceans since an operating GES must simultaneously be in view. Globalstar has not this facility, but this is in contrast to Iridium’s ability to serve users no matter where on or above the planet they may be located by employing intersatellite links.

The Iridium satellite bus presented in Fig. 2.26 is adapted on the satellite’s Earth face, which integrates communication and other types of antennas, crosslink arrays, secondary payload, and other parts. The secondary payload envelope reserved is $300 \times 400 \times 700$ mm, the two larger dimensions being aligned with the Earth face, the single largest dimension being aligned with the y-axis. While traveling on its orbit, the satellite keeps its z-axis pointed toward the center of the Earth and its velocity vector is aligned with the x-axis. The field of view of the secondary payload is a 75° half-cone opening at its top and center, and pointing toward the nadir (+Z). It is clear of any obstruction.

2.4.3.6 Diagram of VSAT GEO Satellite Communication Repeater

As stated earlier, a transparent payload makes no distinction between uplink carrier and uplink noise, and both signals are forwarded to the downlink. Therefore, at the

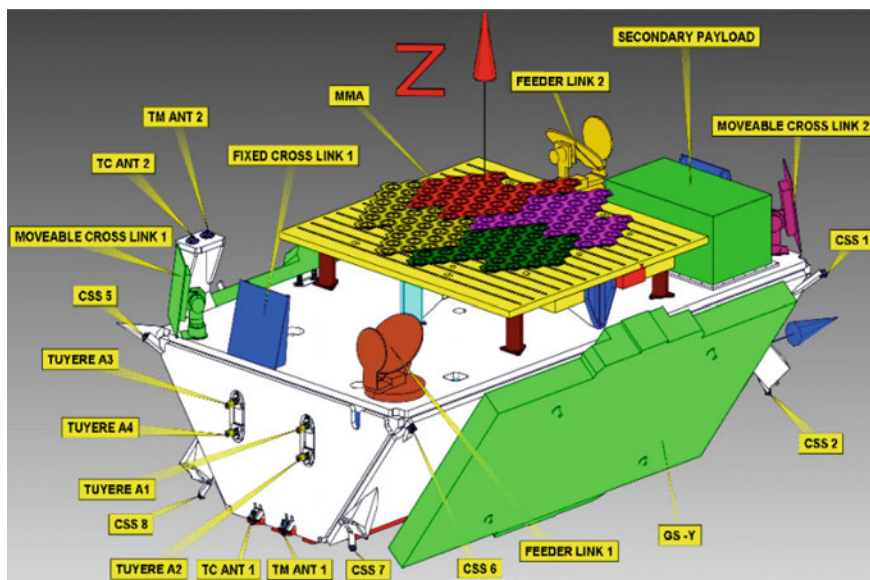


Fig. 2.26 Components of Iridium transponders—Courtesy of Brochure by Thales Alenia

Earth station receiver, one gets the downlink noise together with the uplink-retransmitted noise.

However, a regenerative payload entails onboard demodulation to process of the uplink carriers. Onboard regeneration is most conveniently performed on digital carriers. The bit stream obtained from demodulation of a given uplink carrier is then used to modulate a new carrier at downlink frequency. This carrier is noise-free; hence, a regenerative payload does not retransmit the uplink noise on the downlink. The overall link quality is therefore improved.

Moreover, intermodulation noise can be avoided, as the satellite Channel Amplifier (CA) is no longer requested to operate in a multicarrier mode. Indeed, several bit streams at the output of various demodulators can be combined into a Time Division Multiplex (TDM), which modulates a single high-rate downlink carrier. However, this carrier is amplified by the CA section, which can be operated at saturation without generating intermodulation noise, as the carrier it amplifies is unique. This concept of VSAT Satellite transponder is illustrated in Fig. 2.27.

It should be emphasized that today's commercial satellites that can be used for VSAT services are not equipped with regenerative payloads but only with transparent ones. Only a few experimental satellites such as NASA's Advanced Communications Technology Satellite (ACTS) and the Italian ITALSAT satellite have incorporated a regenerative payload, but they are no longer in operation. Some satellites of the EUTELSAT fleet are equipped with a regenerative payload (Skyplex) but can be used only by Earth stations operating according to the DVB-S standard.

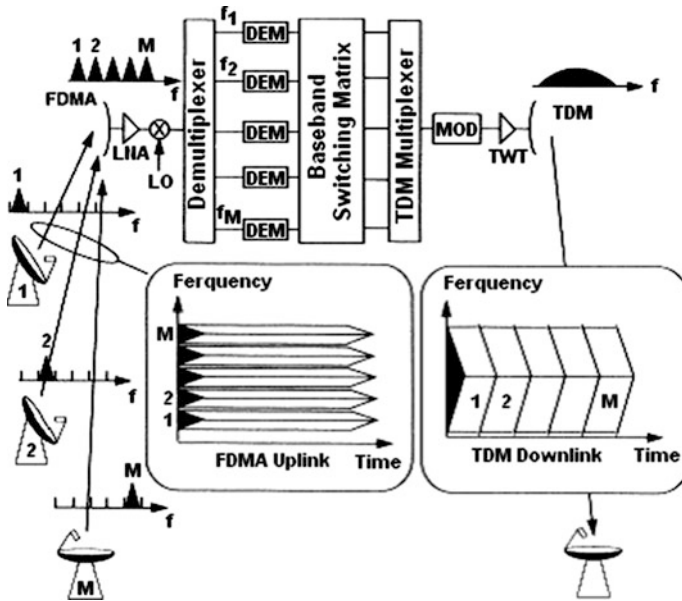


Fig. 2.27 Diagram of VSAT Spacecraft transponders—Courtesy of Book: by Maral

2.4.4 Satellite Navigation Repeaters for GNSS

The new generations of modern GEO satellites can augment the performance of the US GPS and Russia GLONASS by providing a separate ranging channel to transmit integrity and correction data. In fact, this concept dates back to the late eighties and has evolved to its current form known as Regional Satellite Augmentation System (RSAS).

In this book, the use of the ICAO known nomination Satellite-based Augmentation System (SBAS), which appear in the world classification of the acronyms, will be replaced by the Regional Satellite Augmentation System (RSAS) as more convenient nomenclature.

The RSAS complementary information processed in any Master station is diffused by GEO satellite by means of a pseudo-GPS or GLONASS signals and covers a wide regional or geographical area integrated in Global Satellite Augmentation System (GSAS). The RSAS data will allow satellite navigation to meet the stringent reliability and other requirements set by authorities for Air Traffic Control (ATC) and Maritime Traffic Control (MTC). Land users, such as road and railways, will also be able to take advantage of the improvements in positioning accuracy. However, augmented GNSS-1 solutions of RSAS were recently developed to improve the mentioned deficiencies of current GPS and GLONASS military

systems and to meet the present transportation civilian requirements for high-operating integrity, continuity, accuracy, and availability (ICAA).

The three current RSAS networks, the US Wide Area Augmentation System (WAAS), the European Geostationary Navigation Overlay Service (EGNOS), and the Japanese MTSAT Satellite-based Augmentation System (MSAS) are using Inmarsat, Artemis, and MTSAT GEO constellations, respectively. The GSAS network worldwide will require additional GEO spacecraft to provide the necessary coverage, availability, and improved continuity of service. In response to this need, Inmarsat has already embarked the new navigation transponder to support RSAS functions on its current generation of GEO Inmarsat-4 satellite shown in Fig. 2.28, which is a modern development to provide new broadband services.

In additional, WAAS is leasing transponders on two GEO spacecraft, such as the Canadian Telsat Anik F1R and PanAmSat Galaxy XV, EGNOS is using Inmarsat-4 and Artemis constellation and MSAT system is employing own GEO spacecraft of MTSAT series.

2.4.4.1 Inmarsat-4 Navigation Payload

The first GNSS transponder was installed in the payload of Inmarsat-3 spacecraft providing RSAS service until 2005.

The current EGNOS space constellation is composed of two leased transponder hosts of the Inmarsat-4 AORE and IOR, and one ESA Artemis GEO spacecraft. The Inmarsat-4 hosts a navigation payload capable of transmitting RSAS information on both the GPS L1 and the L5 frequencies via GES to the mobile users. Therefore, this payload will transmit satellite navigation signals and allow the real-time relay from a single-ground-monitoring network of integrity and accuracy augmentation data for orbiting GNSS. A functional diagram of the Inmarsat-4 satellite is shown in

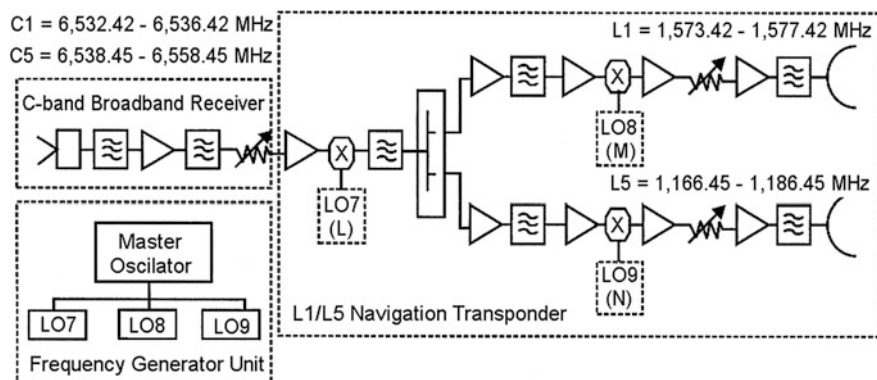


Fig. 2.28 Diagram of Inmarsat-4 Navigation (GNSS) transponder—Courtesy of Article: by Razumovsky

Fig. 2.26. This payload is a dual-channel bent-pipe transponder that converts two C-band (C1 and C5) uplink signals from one GES to two downlink signals in two separate bands with the following characteristics:

1. Bandwidth Limitations—A 4.0 MHz-wide C1 band uplink channel will be relayed in the L1 downlink channel and allow the transmission of the RSAS L1 signal. A 20.0 MHz-wide C5 band uplink channel will be relayed in the L5 downlink channel and allow the transmission of the newly defined L5 RSAS signal. The system has not allocated 24 MHz of bandwidth at both frequencies, but the design included extra spectrum necessary to uplink the newly defined signal at L5 and that left only 4 MHz available at L1, still more than what was provided by Inmarsat-3.

2. Transponders—The Inmarsat-4 communication and navigation transponders share both the C-band broadband receiver and the Frequency Generator Unit, while the two channels (C1-L1 and C5-L5) share hardware until after filtering at IF. The uplink channels are frequency-translated, using coherent translation oscillators (derived from a common master oscillator), to the corresponding L1 and L5 downlink channels. The navigation uplink signals are downconverted with the same local oscillator (LO7) to a suitable IF and then split into two signal paths where the appropriate channel bandwidths are applied. The two signals are then upconverted into the final output frequency (L1 and L5) and fed to the L-band HPA unit before transmission to the dedicated navigation antenna providing global coverage. Both channels have independent telecommandable gain adjustment. All signal paths through the navigation transponder are cold redundant.

3. Broadcast Power—The payload broadcast power will be 28.1 dBW at L1 and 26.2 dBW at L5 in a global coverage antenna beam. The output power at the L1 frequency has been designed to assure the payload would be capable of delivering the necessary power as required by the ICAO Standards and Recommended Practices (SARP) for an RSAS signal. The EIRP output of 26.2 dBW at L5 will permit the user and receiver to receive the same level of radiated power it receives at L1, taking into consideration the smaller free-space loss at L5 frequency. The transponder is expected to accommodate only one signal per channel, which means that the specified broadcast powers are saturated values.

4. Navigation Performance—The overall ranging and navigation performance of the RSAS signal through a bent-pipe repeater is influenced not only by the satellite repeater characteristics, but also by the supporting ground signal generator and timing system. Unlike GPS satellites, Inmarsat navigation payloads do not generate the signal. Rather, their “clock” is on the ground and the signal is generated there. Hence, a closed-loop control system is necessary in order to make the ranging signal appear as if it is originated on board the GEO satellite by compensating for the uplink delay contribution due to the ionosphere, Doppler shift and the frequency offset induced by the GEO transponder itself.

In other words, the final aim was to have the signal broadcast by the transponder resemble a GPS signal. In particular, the closed-loop control system must be capable of maintaining signal code/carrier coherence; that is, the code-chipping rate (1.023 MHz of GPS C/A code) and carrier frequency (L1) must be kept in the

constant ratio of 1:1540 as it happens in the GPS signals. This requirement allows GPS receivers to improve the accuracy of their ranging measurements and ICAO has included it in the SBAS signal specification.

5. Transponder Control—In the past, Inmarsat sponsored the development of signal generators and receivers to demonstrate achievable performance and to perform validation testing of the Inmarsat-3 payloads, both prior to and after launch. Inmarsat-4 navigation signals, like those for Inmarsat-3 satellite, will require a proper control of the signal code and carrier: This time the challenge will be to maintain a proper code/carrier coherence on both L1 and L5 signals. The navigation satellite transponder control will be realized through the Ground Uplink System (GUS) at a GNSS GES terminal. Inmarsat plans to design and develop prototype onboard equipment for proper navigation signal generation and control that will be used for the ground and in-orbit test campaign in order to conduct end-to-end system tests.

2.4.4.2 Artemis Navigation Payload

The aim of the Artemis payload illustrated in Fig. 2.29 was the same as GNSS payload of Artemis spacecraft, namely to provide an overlay function with supplementary services to the current operating GPS and GLONASS constellation. This kind of services uses existing GEO communications satellite to transmit overlay signals to provide improvement of accuracy, system's integrity data, and wide area differential corrections.

The Artemis Navigation (GNSS) satellite payload is caring transparent transponder capable to receive a signal from a European coverage and to translate it onto two separate downlink carriers, the first operating at L-band (L1) and the second one at Ku-band (channel F2), which is illustrated in Fig. 2.30.

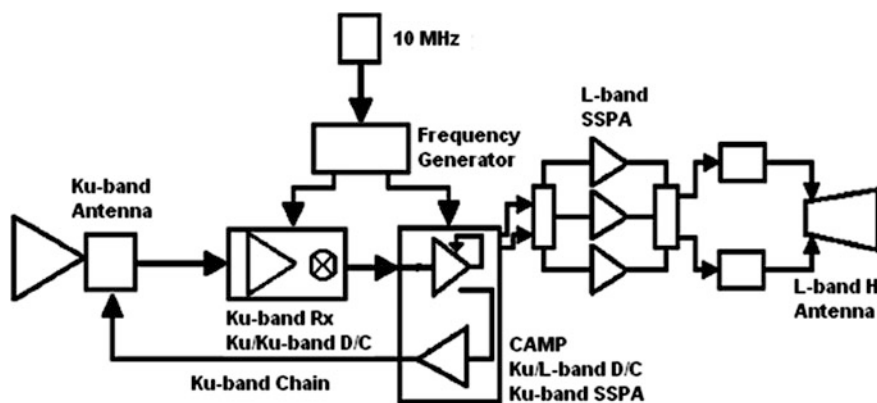


Fig. 2.29 Diagram of Artemis Navigation Transponder—Courtesy of Paper: by Comparini

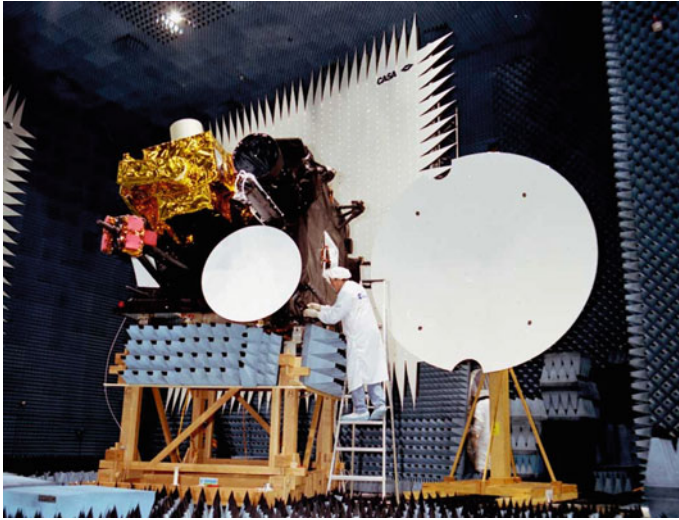


Fig. 2.30 Artemis Communication and GNSS Payload—Courtesy of Brochure: by ESA

The signal transmitted from the ground is a single 500 b/s carrier modulated with timing, integrity, ranging, and other information and spread with a PN code, a type compatible to those used for GPS. The code orthogonality properties assure the possibility to transmit the GNSS signal on the same nominal carrier frequency of the GPS and GLONASS.

The uplink signal is received at 13.875 GHz by a Ku-band antenna and filtered by the input multiplexer, which separates the incoming communication signal from the NOS signal and to reduce interference. This filter is mounted directly on the spacecraft top floor. The input signal is divided into two paths and translated; therefore, the heart of the payload performing this function is the CA module. At this point, the low-noise amplification mode is mainly performed by the Ku-band receiver (KRx), which provides a noise figure of 2.3 dB and a gain of about 53 dB. The signal is then fed into the Ku-band channel filter. Realized using high Q waveguide dielectric resonator, it gives only 10 nSec of group delay variation into the useful bandwidth of 4 MHz. The signal feeds the CA element at the output of the channel filter. The unit splits the signal into two paths: One feed the Ku-band transmit section without RF conversion while the other is downconverted to the GPS frequency. The two CA output signals are amplified by two SSPA components providing 16 watts end of life. All the RF units use latest technologies (hybrids, MMIC, MCM) both for the RF portion and for the control and interface functions.

1. Ku-band Rx—The KRx unit provides low-noise amplification and down-conversion of signals at RF of 13.855-13.895 GHz down to 12.728-12.768 GHz, by means of a net subtraction of an externally generated LO frequency. The equipment is provided with an internal power supply and a command and telemetry interface. The signal is then amplified in a linear amplifier. Thus, group delay,

phase, AM/PM conversion, and gain are specified as required by GNSS payload performances analysis. The unit is internally redundant and includes a redundant DC/DC converter to derive the required secondary voltages from the satellite main bus. The input interface is WR75 standard, and the LO input and the RF output interfaces are SMA 50 Ω. The key technologies characterizing the units are: MMIC to achieve reduction of dimensions and mass and a high level of reproducibility; RF hybrid technique to house both MW and low frequency analog/digital integrated circuits; and application-specific integrated circuit (ASIC) for control and TM/TC interface functions.

A IF bandpass filter provides suppression of out-of-band mixing products. The rest of the IF section is composed of a cascade of different MMIC circuits which provide the required gain and gain control dynamic. The first gain stage is an MMIC LNA as the one above described: The large passband of the MMIC allows use on the IF section as well, in order for it to have negligible contribution to the overall noise figure. The MESFET process has been used for the MMIC units following in the IF chain. The Flatness Corrector (FC) circuit gives the possibility to control the gain slope variation consequent to the complete system assembling and possible slope variations over temperature. The gain control block is the variable gain amplifier (VGA), which is composed of three self-biased stages of amplification with an embedded analog attenuator, allowing compensation of gain drift over temperature of the complete assembly. The analog attenuator design was based on a broadband-extended T configuration using cold FET units. This approach was chosen because of its minimal VSWR variation and flat gain response over the attenuation range. A medium power amplifier (MPA) is used as the output stage in order to have higher output level and better linearity. The circuit design is single-ended and consists of a two-stage amplifier. Output FET gate periphery was selected to achieve a 32 dBm third-order intercept point while keeping the channel temperature below 110 °C when in maximum environmental temperature.

2. Channel Amplifier (CA)—This unit is used to amplify the signal to a level suitable to drive the power amplifier. It can be set into two operational modes: **a)** Nominal operation of amplifier level control (ALC) mode is settable by means of ground command to different levels. In fact, the ALC loop maintains the output power level within the specified limits recovering uplink variations and input signal dynamic; and fixed gain mode, when ALC is off, is the amplifier provides a fixed gain set by ground command.

The CA unit is in general redundant and equipped with internal DC/DC converter and provides two different outputs: **L1** at a frequency of **1575.42 MHz**, generated by means an external local oscillator, and **F2** at a frequency of **12748 MHz** (no conversion is required). As in the Ku-band RX, also in CA hybrid circuits are used to realize the RF channels and the digital control circuits. The following modules compose the unit: two complete Ku-band CA modules (main and redundant); two Ku/L-band RF downconverter including CA module functionality (main and redundant); redundant DC/DC converter; and couplers for redundancy and channels splitting.

3. **Solid-State Amplifier (SSA)**—The SSA module is used for high-power amplification of L-band downlink signal. This unit, manufactured by the UK MMS company, now Astrium UK), consists of: an amplifier module in which high-power RF amplification and power generation occur; an electronic power conditioner (EPC) which accepts primary bus from the spacecraft and generates the required secondary stabilized voltages.

4. **RF Generator (RFG)**—This unit generates two frequencies at L- and X-bands using a 10.0 MHz reference signal coming from the external USO. In the Artemis implementation, the onboard USO has been used. The unit consisting of the navigation RFG plus the 10 MHz reference has been specified as a whole, in terms of performance; however, the RFG and the 10 MHz USO are mechanically separated in order to maintain the qualification of the assembly achieved on Artemis for the RFG, on SICRAL, and on UHF spacecraft update programs for the 10 MHz ultra-stable oscillator (USO). The RFG output frequencies are 1127 MHz and 11172.58 MHz. Direct synthesis approach is used: Multiplication is obtained using active stages and step recovery diodes. At this point, quartz and helical filters are used to reduce spurs and phase noise to levels within the specification.

2.4.4.3 MTSAT Navigation Payload

The Japanese GNSS payload is integrated together with meteorological payload onboard series of GEO Multifunctional Transport Satellite (MTSAT). The MTSAT GNSS payload is the forerunner of a family of GNSS payloads able to fulfill augmentation system for Japan and surrounded areas allowing interoperability among the different systems.

The MTSAT-1R spacecraft launched in February 2005 replaced unsuccessful MTSAT-1, which was destroyed in a launch failure in 1999. Air traffic communications will use the L-band and the AMSS protocol, which is specific to mobile satellite communications. Links between the Earth segment and satellite segment will use the Ku- and Ka-band.

This advanced GEO satellite has rideshare payloads that play a critical role in the next generation of the Japanese satellite-enhanced air traffic management systems and weather forecasting. The aeronautical payload is a natural improvement on the previously designed MTSAT-1 payload. A new state-of-the-art digital processor facilitates communication link frequency adjustments that may be required by international coordination agreements after MTSAT-1R is put into service. It also facilitates efficient use of the allocated frequency spectrum and improves transponder performance.

2.4.5 Repeaters for Stratospheric Platform Systems (SPS)

The SPS or high-altitude platform (HPA) transponder shown in Fig. 2.31 illustrates a block diagram of a typical onboard architecture, which together with its associated

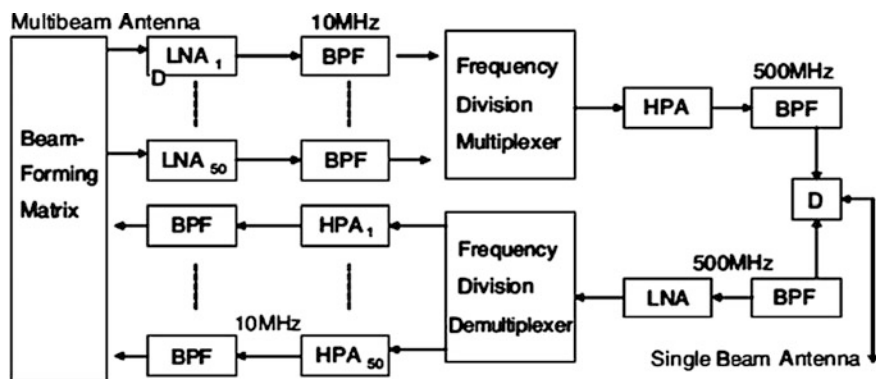


Fig. 2.31 SPS transponder—Courtesy of Book: by Aragon-Zavala

antenna subsystem would make a complete communications payload. In this case, SPS transponder has a bandwidth of 500 MHz using Code Division Multiple Access (CDMA) scheme. This transponder can accommodate up to 50 antenna beams with eight wideband carriers (assuming a carrier bandwidth of 1.25 MHz each). For the uplink, the carriers are received by the platform antenna subsystem and amplified by a LNA.

The bandwidth of these carriers is limited to only 10 MHz using passband filters, and multiplexed using Frequency Division Multiplexing (FDM). Before these multiplexed signals are transmitted to the ground station, they are amplified further by a HPA stage, filtered and multiplexed. For the downlink, the process is similar but in the reverse direction, and instead of a multiplexer, a demultiplexer is used [ITU-F1500, 00].

Other transponder components play a very important role in the system performance, and their design and selection should not be underestimated. Thus, the LNA element having low-noise temperature and sufficient gain, ensure that noise contributions from the succeeding stages are kept small. Frequency converters serve as frequency-change devices (also known as mixers), which either upconvert or downconvert the signals in the SCP to differentiate between uplink and downlink. Noise performance is also important for these frequency converters, although not as critical as the noise figure for the LNA, since this is the first element in the frontend and therefore, is the one that contributes most to the overall noise. IF processors have the main functions of providing most of transponder gain, defining a frequency response of each channel and, whenever required, performing beam-to-beam routing functions. Filters are in charge of limiting adjacent spurious signals as well as noise that can be generated by the payload itself. Their design is crucial for the performance of the communications payload. Transmitters are in charge of amplifying the signals to the level required for downlink transmissions. Examples of well-known transmitters used for satellite communications are the TWT and power transistors, the latter used for specific applications in lower

frequency bands. Payload system performance or performance parameters of individual equipments need to be specified so that the required SCP payload performance is achieved by the following important parameters:

1. **Antenna Coverage Area**—Since antenna gain and SCP coverage are related to antenna dimensions, antenna electrical efficiency and antenna feeder losses;
2. **Figure of Merit G/T**—Since this parameter as ratio of the receive antenna gain to system noise temperature (gain-to-noise temperature) affects link performance;
3. **EIRP**—Is determined by the power capability of the SCP;
4. **Power per Backhaul Carrier**—Is the user link;
5. **Isolation Between Channels**—It reduces potential adjacent channel interference issues;
6. **Spurious Outputs**—Reduce interference levels to other contiguous wireless systems;
7. **Amplifier Linearity**—Since nonlinearities produce intermodulation products that can leak into the passband of other equipment, causing interference; and
8. **Group Delay Variation**—Is the time delay experienced by the modulation waveform in passing through the equipment, causing signal dispersion and performance degradation.

2.4.6 Satellite Antenna System for MSC

The antenna array system of Inmarsat-2 satellite for MSC is shown in Fig. 2.32 (Left). The antenna system mounted on the spacecraft structure, similar to the transponders, is composed of two main integrated elements: the C/L-band and the L/C-band antenna.

1. **Inmarsat-2 C/L-band Arrays**—This uplink is actually the feeder link, which operates in the 6 GHz RF range. The signals sent by LES are detected by a C-band receiving array, comprising seven cup-dipole elements in the smallest circle. On the other hand, the L-band transmit antenna is the biggest segment of the whole system, consisting of 43 individual dipole elements, arranged in three rings around a single central element. Thus, this antenna is providing near-global coverage service downlink for MES in the 1.5 GHz RF spectrums.

2. **Inmarsat-2 L/C-band Arrays**—These arrays are actually the service uplink and operates in the 1.6 GHz RF range. The signals sent by MES in adjacent global coverage region are detected by L-band receiving array, comprising nine cup-dipole elements arranged in a circle. Finally, the C-band transmit antenna consists of seven cup-dipoles for radiation of the feeder downlink to LES in the 3.6 GHz RF spectrum.

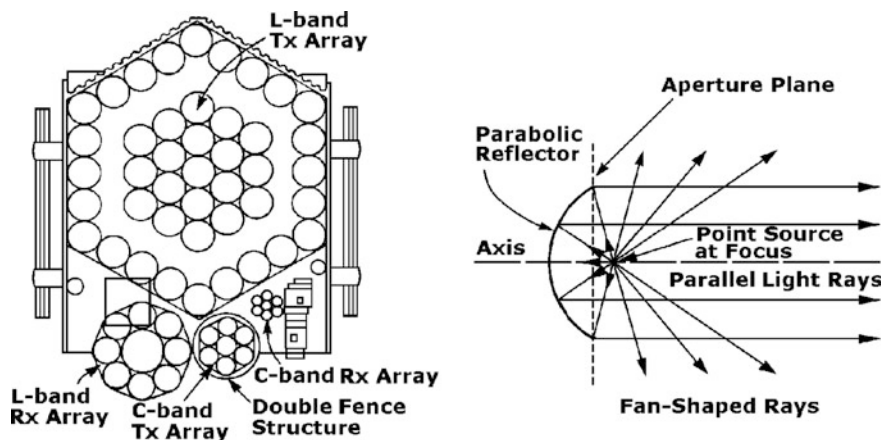


Fig. 2.32 Spacecraft antenna systems—Courtesy of Book: by Gallagher

2.4.6.1 Characteristics of Satellite Antennas

Both transmit antenna array systems are providing a global (wide) footprint on the Earth's surface. However, narrow circular beams from GEO or non-GEO can be used to provide spot beam coverage. For instance, from GEO the Earth subtends an angle of 17.4° . Antenna beams 5.8° wide can reuse three frequency bands twice in providing Earth disk coverage. The directional properties of antenna arrays can be exploited to permit RF reuse in space communications, which is similar to several radio stations using the same RF being geographically far apart.

Earth coverage by seven spot beams (six spots are set out around one spot in the center) can be arranged by three pairs of beams: 1 and 4, 2 and 5, and 3 and 6, operating on frequencies f_2 , f_3 , and f_4 , respectively. Mutual interference within pairs is avoided by pointing one beam as far away from the other as possible. Satellite antenna coverage of the center of the disk is provided by a single-beam operating on frequency f_1 . Thus, the main advantage with this satellite spot footprint that is specific Earth areas can be covered more accurately than with wide beams.

Furthermore, a greater power density per unit area for a given input power can be achieved very well, when compared with that produced by a global circular beam, leading to the use of much smaller receiving MES antennas. The equation that determines received power (P_R) is proportional to the power transmitted (P_T) separated by a distance (R), with gain of transmit antenna (G_T) and effective area of receiving antenna (A_R) and inversely proportional to 4π and square of distance. The relations for P_R and G_T are represented as follows:

$$\begin{aligned} P_R &= P_T G_T A_R / 4\pi R^2 \\ G_T &= 4\pi A_T / \lambda^2 \end{aligned} \quad (2.46)$$

where G_T = effective area of transmit antenna and λ = wavelength. The product of P_T and G_T is gain, generally as an increase in signal power, known as an EIRP. Signal or carrier power received in a link is proportional to the gain of the transmit and receive antennas (G_R) represented as:

$$P_R = P_T G_T G_R \lambda^2 / (4\pi R)^2 \text{ or } P_R = P_T G_T G_R / L_P L_K [W] \quad (2.47)$$

The last relation can be derived with the density of noise power giving:

$$P_R/N = P_T G_T (G_R/T_R) (1/K L_P L_K) \quad (2.48)$$

where L_P = coefficient of energy loss in free space, L_K = coefficient of EMW energy absorption in satellite channels, T_R = temperature noise of receiver, G_R/T_R is the figure of merit and K = Boltzmann's Constant (1.38×10^{-23} J/K or its alternatively value is -228.6 dBW/K/Hz).

At any rate, P_R has a minimum allowable value compared with system noise power (N); that is, the Carrier and Noise (C/N) or Signal and Noise (S/N) ratio must exceed a certain value. This may be achieved by a trade-off between EIRP ($P_T G_T$) and received antenna gain (G_R). If the receive antenna on the satellite is very efficient, the demands on the LES/MES are minimized. Similarly, on the satellite-to-Earth link, the higher the gain of the satellite transmit antenna, the greater the EIRP for a given transmitter power. Satellites often have parabolic dish antennas, though there are also other types, such as phased arrays. The principal property of a parabolic reflector is its ability to turn light from a point source placed at its focus into a parallel beam, mostly as illustrated in Fig. 2.32 (Right). In practice the beam can never be truly parallel, because rays can also be fan-shaped; i.e., a car headlamp is a typical example. In a microwave antenna, the light source is replaced by the antenna feed, which directs waves toward the reflector. The length of all paths from feed to aperture plane via the reflector is constant, irrespective of their angle of parabolic axis. The phase of the wave in the aperture plane is constant, resulting in maximum efficiency and gain. The gain of an aperture (G_a) and parabolic (G_p) type of antennas are as follows:

$$\begin{aligned} G_a &= \eta (4\pi A/\lambda^2) = 4\pi A_E/\lambda^2 \\ G_{yps} &= \eta (\pi D)^2/\lambda^2 \end{aligned} \quad (2.49)$$

where η = efficiency factor, A = projected aperture area of antenna, $A_E = \eta^A$ is the effective collecting area, and D = parabolic antenna diameter. Thus, owing to correlation between frequency and wavelength, $f = c/\lambda$ is given in the following relations:

$$G_p = \eta (\pi D f/c)^2 = 60,7 (D f)^2 \quad (2.50)$$

where the second relation comes from considering that $\eta \approx 0.55$ of numerical value. If this value is presented in decibels, then the gain of antenna will be calculated as follows:

$$G_T = 10 \log G_p \quad (2.51)$$

For example, a parabolic antenna of 2 m in diameter has a gain of 36 dB for a frequency at 4 GHz and a gain of 38 dB for a frequency at 6 GHz. Parabolic antennas can have aperture planes that are circular, elliptical, or rectangular in shape. Thus, antenna with circular shape and homogeneous illumination of aperture with a gain of -3 dB has about 47.5 % of effective radiation, and the rest of the power is lost. To find out the ideal characteristics, it is necessary to determine the function diagram of radiation in the following way:

$$F(\delta_o) = s(\delta_o)/s(\delta_o = 0) \quad (2.52)$$

where parameter $s(\delta_o)$ = flow density of radiation in the hypothetical satellite angle (δ_o) and $s(\delta_o = 0)$ = flow density in the middle of the coverage area. Looking the geometrical relations in Fig. 2.2a follows the relation:

$$F(\delta_o) = d_o/h = \cos \delta \sqrt{(k^2 - \sin^2 \delta_o)} / 1 - k \quad (2.53)$$

where, as mentioned, $k = R/(R + h) = \sin \delta$ and if $\delta_o = \delta$, the relation is defined by the following equation:

$$F(\delta) = k \cos \delta \quad (2.54)$$

For GEO satellite, the value of ΔL is given as a function of angle δ , which is the distance from the center of the coverage area, where the function diagram of the radiation is as follows:

$$F(\delta) = \Delta L = 20 \log R/(R + h) \cos \delta = 10 \log R/(1 + 2R/h) [\text{dB}] \quad (2.55)$$

Therefore, in the case of GEO satellites the losses of antenna propagation are greater around the periphery than in the center of the coverage area for about 1.32 dB. The free-space propagation loss (L_P) and the input level of received signals (L_K) are given by the equations:

$$\begin{aligned} L_P &= (4\pi d/\lambda)^2 \\ P_R/S &= P_T G_T / 4\pi d^2 L_K \end{aligned} \quad (2.56)$$

The free-space propagation loss is caused by geometrical attenuation during propagation from the transmitter to the receiver.

2.4.6.2 Link Performance with Monobeam and Multibeam Antenna Coverage

As stated earlier, the most important parameter of spacecraft transponder and the overall RF link quality depends on the gain of the satellite antenna. From Eq. (2.47), it can be seen that the satellite antenna gain is constrained by its beamwidth, whatever the frequency at which the link is operated. So the antenna gain is imposed by the angular width of the antenna beam covering the zone to be served. If the service zone is covered using a single-antenna beam, this is referred to as single or monobeam beam coverage, which displays one of these characteristics:

- The satellite may provide coverage of the whole region of the Earth, which is visible from the satellite as a global coverage and thus permits long-distance links to be established, for example from one continent to another with 20 dB bandwidth. In this case, the gain of the satellite antenna is limited by its beamwidth as imposed by the coverage.
- The satellite may provide coverage of only part of the Earth (a region or country) by means of a narrow beam (a zone or spot beam), with 3 dB beamwidth of the order of 1° to a few degrees.

With a single-beam satellite antenna coverage, it will be therefore necessary to choose between either extended coverage providing service with reduced quality of service to geographically dispersed GES terminals, or with reduced coverage providing service with improved quality of transmission and reception to selected geographically concentrated GES terminals.

Multibeam satellite antenna coverage allows these two alternatives to be reconciled. Thus, satellite extended coverage may be achieved by means of the juxtaposition of several narrow-beam antenna coverages, so in such a way, each beam provides an antenna gain that increases as the satellite antenna beamwidth decreases (reduced coverage per beam). The satellite link performance improves as the number of beams increases, which limit can be determined by the antenna technology, whose complexity increases with the number of beams and the mass.

The complexity originates in the more elaborate satellite antenna technology and the requirement to provide onboard interconnection of the coverage areas, so as to ensure within the satellite payload routing of the various carriers that are unlinked in different beams to any wanted destination beam.

In Fig. 2.33 (Left) is illustrated satellite that provides global coverage with a single beam (monobeam) of beamwidth and in Fig. 2.33 (Right) is illustrated that satellite supports spot beams with beamwidth of a consequently reduced coverage, known as multibeam satellite coverage. In both cases, all GES terminals in the satellite network are within the correspondent satellite coverage or in LOS with satellite. Multibeam coverage is providing the following advantages:

1. Impact on Earth Segment—The satellite communication link performance is evaluated as the ratio of the received carrier power C to the noise power spectral

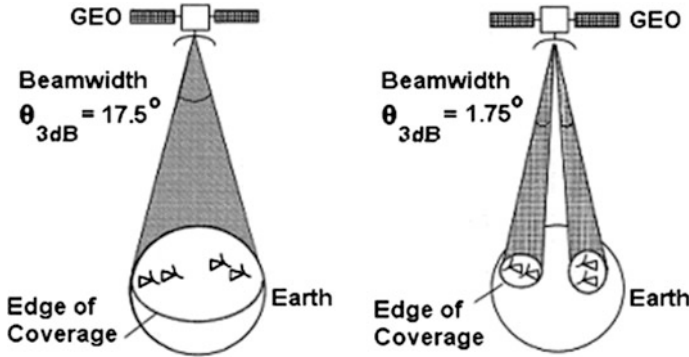


Fig. 2.33 Global monobeam and multibeam antenna coverage—Courtesy of Book: by Maral

density N_0 and is quoted as the C/N_0 ratio, expressed in Hz. The expression for $(C/N_0)_U$ for the uplink (U) is given by the following equation:

$$(C/N_0)_U = (\text{EIRP})_{\text{station}} (1/L_U) (G/T)_{\text{satellite}} (1/k) [\text{Hz}] \quad (2.57)$$

Assuming that the noise temperature at the satellite receiver input is $T_{\text{satellite}} = 800 \text{ K} = 29 \text{ dBK}$ and is independent of the beam coverage (this is not rigorously true but satisfies a first approximation), let $L_U = 200 \text{ dB}$ and neglect the implementation losses. At this point, this equation becomes (all terms in dB):

$$\begin{aligned} (C/N_0)_U &= (\text{EIRP})_{\text{station}} - 200 + (G_R)_{\text{satellite}} - 29 + 228.6 \\ &= (\text{EIRP})_{\text{station}} + (G_R)_{\text{satellite}} - 0.4 [\text{dBHz}] \end{aligned} \quad (2.58)$$

where $(G_R)_{\text{satellite}}$ is the gain of the receiving satellite antenna in the direction of the GES transmitting terminals. This relation is represented by the two cases considered receiver:

- Global coverage ($\theta_{3 \text{ dB}} = 17.5^\circ$), which implies $(G_R)_{\text{satellite}} = 29\,000/(\theta_{3 \text{ dB}})^2 \approx 20 \text{ dBi}$.
- Spot beam coverage ($\theta_{3 \text{ dB}} = 1.75^\circ$), which implies $(G_R)_{\text{satellite}} = 29\,000/(\theta_{3 \text{ dB}})^2 \approx 40 \text{ dBi}$.

The expression for $(C/N_0)_D$ for the downlink (D) is given by:

$$(C/N_0)_D = (\text{EIRP})_{\text{satellite}} (1/L_D) (G/T)_{\text{station}} (1/k) [\text{Hz}] \quad (2.59)$$

Assume that the power of the carrier transmitted by the satellite is $P_T = 10 \text{ W} = 10 \text{ dBW}$. Let $L_U = 200 \text{ dB}$ and neglect the implementation losses. Thus, this equation becomes (all terms in dB):

$$\begin{aligned}
 (C/N_0)_D &= 10-200 + (G_T)_{\text{satellite}} + (G/T)_{\text{station}} + 228.6 \\
 &= (G_T)_{\text{satellite}} + (G/T)_{\text{station}} + 38.6 \text{ [dBHz]}
 \end{aligned}
 \tag{2.60}$$

This relation is represented for the two cases considered transmitter:

- Global coverage ($\theta_3 \text{ dB} = 17.5^\circ$), which implies $(G_T)_{\text{satellite}} = 29 \text{ 000}/(\theta_3 \text{ dB})^2 \approx 20 \text{ dBi}$.
- Spot beam coverage ($\theta_3 \text{ dB} = 1.75^\circ$), which implies $(G_T)_{\text{satellite}} = 29 \text{ 000}/(\theta_3 \text{ dB})^2 \approx 40 \text{ dBi}$.

In case that values indicate the reduction in $(EIRP)_{\text{station}}$ and $(G/T)_{\text{station}}$, the transmission system is changing from a satellite with global coverage to a multi-beam satellite with coverage by several spot beams. In this case, the multibeam satellite permits an economy of size and, hence, cost of the Earth segment. For instance, a 20 dB reduction of $(EIRP)_{\text{station}}$ and $(G/T)_{\text{station}}$ may result in a tenfold reduction of the antenna size (perhaps from 30 m to 3 m) with a cost reduction for the GES terminal for more than 100 times. If an identical GES is retained (a vertical displacement toward the top), an increase of C/N_0 is achieved which can be transferred to an increase of capacity, if sufficient bandwidth is available, at constant signal quality in terms of Bit Error Rate (BER).

2. Frequency Reuse—Frequency reuse consists of using the same frequency band several times in such a way as to increase the total capacity of the network without increasing the allocated bandwidth (B). In the case of a multibeam satellite, the isolation resulting from antenna directivity can be exploited to reuse the same frequency band in separate beam coverages. The frequency reuse factor is defined as the number of times that the bandwidth is used. In theory, a multibeam satellite with M single-polarization antenna beams, each being allocated the bandwidth, combines reuse by angular separation and reuse by orthogonal polarization may have a frequency reuse factor equal to $2M$. This signifies that it can claim the capacity which would be offered by a single-beam satellite with single polarization using a bandwidth of $M \times B$. In practice, the frequency reuse factor depends on the configuration of the service area which determines the coverage before it is provided by the satellite. If the service area consists of several widely separated regions (for example, urban areas separated by extensive rural areas), it is possible to reuse the same band in all beams. The frequency reuse factor can then attain the theoretical value of M . Figure 2.31 shows an example of multibeam coverage.

On the other hand, multibeam coverage is providing the following disadvantages:

1. Interference Between Beams—In practical reality, the interference generation within a multibeam satellite system is called self-interference. Thus, the effect of self-interference appears as an increase in thermal noise under the same conditions as interference noise between systems. At this point, it must be included the term $(C/N_0)_I$, which expresses the signal power in relation to the spectral density interference. Taking account of the multiplicity of sources of interference, which become more numerous as the number of beams increases, relatively low values of

$(C/N_0)_1$ may be achieved and the contribution of this term impairs the performance in terms of $(C/N_0)_T$ of the total link. As modern satellite systems tend to re-use frequency as much as possible to increase capacity, self-interference noise in a multibeam satellite link may contribute up to 50 % of the total noise.

2. Interference Between Coverage Areas—A satellite payload using multibeam coverage must be in a position to interconnect all network Earth stations and consequently must provide interconnection of coverage areas. The complexity of the payload is added to that of the multibeam satellite antenna subsystem, which is already much more complex than that of a single-beam satellite. Different techniques, depending on the onboard processing capability (no processing, transparent processing, regenerative processing, etc.) and on the network layer, are considered for interconnection of coverage:

- Interconnection by transponder hopping (no onboard processing);
- Interconnection by onboard switching (transparent and regenerative processing); and
- Interconnection by beam scanning.

Multibeam satellite systems make it possible to reduce the size of GES terminal and hence the cost of the Earth segment infrastructure. Frequency reuse from one satellite beams to another permits an increase in capacity without increasing the bandwidth allocated to the system. However, interference between adjacent satellite channels, which occurs between beams using the same frequencies, limits the potential capacity increase, particularly as interference is greater with Earth stations equipped with small antennas.

The simplest form of a payload with multibeam antenna radiation is shown in Fig. 2.34. A three-transponder payload uses one transponder per coverage circle. At this point, there is not connectivity between coverage areas in this simple transponder. However, this payload could be designed so each transponder antenna illuminates three coverage circles and provide connectivity, but would cover three times area with just one-third the gain.

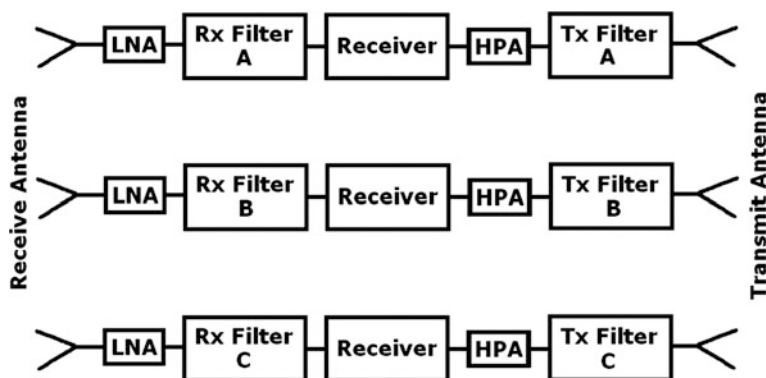


Fig. 2.34 Multibeam antenna coverage transponder—Courtesy of Book: by Swan

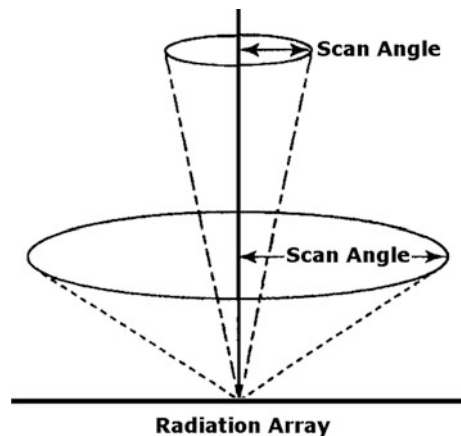
The relatively simple changes to multiple small satellite antenna beams have the following significant consequences as follows:

- (a) The area covered by each beam is much smaller, increasing the antenna gain and allowing smaller, less expensive ground terminals;
- (b) The same total RF power can be used to carry more traffic, and/or reduce the RF power;
- (c) The same transponder bandwidth can be used multiple beam antenna, greatly increasing the available bandwidth of the satellite:
 1. This allow the same bandwidth to be reused, increasing the amount that can be accommodated within the bandwidth; and
 2. A terminal has to be tuned to the correct RF to function and retuned if it is moved.
- (d) Connectivity between beams, if required for the mission, must be provided by additional hardware on the satellite since a single uplink does not encompass the entire coverage area.

The beams may be formed by individual feeds (circles) and by mechanical or electronic beam former. Mechanical beam formers use fixed wave-guide components to control the RF phase and amplitude. Usually there is one amplifier for each transponder in each composite beam. Electronic beam formers use electronic RF phase shifters, and sometimes electronic amplitude control, to produce the multi-beam. There are many radiating elements and each usually has its own amplifier.

Antenna radiators fall into two categories, reflector and direct radiating antennas. Reflector antenna uses a feed that indirectly radiates the energy toward the illuminated area of users, while direct radiating antenna radiates the energy direct to the coverage area. The spot beam antenna can be pointed in various directions within a cone characterized by the scan angle, shown in Fig. 2.35. In this way, a direct radiating array can radiate at large scan as illustrated in Fig. 2.35. Such arrays are

Fig. 2.35 Antenna scan angles—Courtesy of Book: by Swan



attractive for LEO communications satellites because they operate over large scan angle than reflectors antenna, so they require a scan angle of 63° to cover its field of view. In contrary, a GEO communications satellite requires a scan angle of 7° to cover the entire visible circle on the Earth.

In addition to the fact that satellite antenna gain decreases as the scan angle increases, and polarization purity decreases as well. Thus, the LEO satellites require a direct radiating antenna, while commercial GEO satellite do not use this type of antenna, but directional reflector antennas. At MEO, the situation is less clear and both direct radiating and reflector antennas have been proposed for this orbit.

2.4.6.3 Basic Types of Antenna Onboard Spacecraft

Four main types of antennas are used on satellites: 1. Wire antennas: monopoles and dipoles; 2. Horn antennas; 3. Reflector antennas; and 4. Array antennas.

Wire antennas are used primarily at VHF and UHF to provide communications for the Telemetry, Tracking, and Command (TT&C) systems. They are positioned with great care on the body of the satellite in an attempt to provide omnidirectional coverage. Most satellites measure only a few wavelengths at VHF frequencies, which make it difficult to get the required antenna patterns, and there tends to be some orientations of the satellite in which the sensitivity of the TT&C system is reduced by nulls in the antenna pattern.

An antenna pattern is a plot of the field strength in the far field of the antenna when a transmitter drives the antenna. The gain of an antenna is a measure in dB of the antenna's capability to direct energy in one direction, rather than all around. A useful principle in antenna theory is reciprocity, which means that an antenna has the same gain and pattern at any given frequency whether it transmits or receives. An antenna pattern measured when receiving is identical to the pattern when transmitting.

As stated earlier, the antenna is providing global, spot, and multiple beam coverages, but it can provide scanning and orthogonally polarized beams or coverage zones as well. The pattern is frequently specified by its 3-dB beamwidth, the angle between the directions in which the radiated (or received) field falls to half the power in the direction of maximum field strength. However, a satellite antenna is used to provide coverage of a certain area or zone on the Earth's surface, and it is more useful to have contours of antenna gain with maximum strengths of the signal in the middle of the coverage area and with decreasing of signals to the peripheries. When computing the signal power received by a GES from the satellite, it is important to know where the station lies relative to the satellite transmit antenna contour pattern so that the exact EIRP can be calculated. If the pattern is not known, it may be possible to estimate the antenna gain in a given direction if the antenna boresight or beam axis direction and its beamwidth are known.

Horn antenna is used at MW when for global coverage relatively wide beams are required. A horn is a flared section of waveguide that provides an aperture several wavelengths wide and a good match between the waveguide impedance and the

free space. It is also used as feeds for reflectors, either singly or in clusters. Horns and reflectors are examples of aperture antennas that launch a wave into free space from a waveguide. It is difficult to obtain gains much greater than 23 dB or beamwidths narrower than about 10° with horn antennas. For higher gains or narrow beamwidths, a reflector antenna or array must be used.

Reflector antennas are usually illuminated by one or more horns and provide a larger aperture than can be achieved with a horn alone. For maximum gain, it is necessary to generate a plane wave in the aperture of the reflector. This is achieved by choosing a reflector profile that has equal path lengths from the feed to the aperture so that all the energy radiated by the feed and reflected by the reflector reaches the aperture with the same phase angle and creates a uniform phase front. One reflector shape that achieves this with a point source of radiation is the paraboloid, with a feed placed at its focus. The paraboloid is the basic shape for most reflector antennas and is commonly used for Earth station antennas. Satellite antennas often use modified paraboloidal reflector profiles to tailor the beam pattern to a particular coverage zone. Phased array antennas are also used on satellites to create multiple beams from a single aperture and have been used by Iridium and Globalstar to generate up to 16 beams from a single aperture for their LEO system.

2.4.7 Satellite Bus

The satellite bus is usually called a platform and consists of several sections, shown in Fig. 2.17. The function of the satellite platform is to support the payload operation reliably throughout the mission of primary construction section, such as Structure Platform (SP), Electric Power (EP), Thermal Control (TC), Attitude and Orbit Control (AOC), Telemetry, Tracking and Command (TT&C) and Propulsion Engine.

2.4.7.1 Structure Platform (SP)

The structure has to house and keep together all components of bus and communications modules, enable protection from the environment, and facilitate connection of the satellite to the launcher. It comprises a skeleton on which the equipment modules are mounted and a panel, which covers and provides protection for sensitive parts during the operational phase from micrometers and helps to shield the equipment from extremes of heat, coldness, vacuum, and weightlessness, including the relatively small dynamic forces produced by the station-keeping, attitude control engines, and inertial momentum devices.

The spacecraft is protected during the launch phase with an enclosure, or nose cone. At the end, the nose cone is jettisoned, at which time the spacecraft must survive the inertial and thermal stress of an additional propulsion stage until it is inserted into orbit. In this sense, a spacecraft is virtually free of gravitational stress

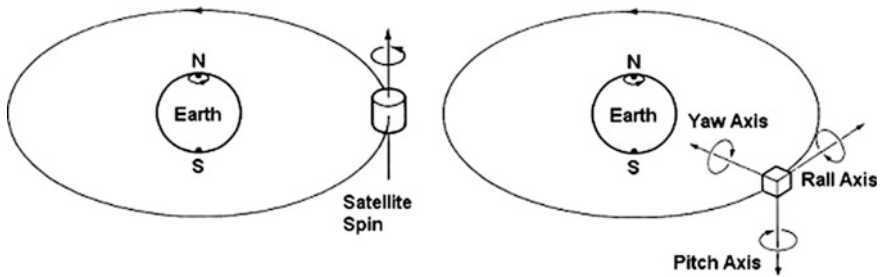


Fig. 2.36 Satellite spin RPY 3-axes stabilization—Courtesy of Book: by Roddy

when in orbit, which allows the use of very large deployable arrays, which would collapse under their own weight on the Earth's surface without problems.

Thus, large stresses are developed during launch as a result of massive acceleration and intense vibration, so the SP body must be sufficiently strong to withstand all external forces. On the other hand, all large structures such as antenna and solar arrays have to be folded and protected during a launch sequence and must have a deployable mechanism. The deployment of structures requires a special technique in the vacuum of space because of the lack of a damping medium, such as air.

Most satellites are either cylindrically shaped and are stabilized by spinning the whole of main body shown in Fig. 2.36 (Left) or box-shaped three-axis body stabilized illustrated in Fig. 2.36 (Right). Spin stabilized structures have a cylindrical part, which rotates at a speed of 50–100 rpm and the despun stabilized part has mounted antennas always facing to the Earth. The spin axis lies along the pitch axis parallel to the Earth's N–S axis. Thus, spin is initiated during the launch phase by means of small gas jet.

The spinning part of the cylinder is covered with solar cells and its spin axis is oriented perpendicularly toward the Sun. Body-stabilized structures rotate once for every rotation of the Earth so that the side with mounted antenna will always face the Earth. This platform utilizes a deployed set of solar panels with solar cells mounted on one side of the panel surface relative to the Sun.

The box-shaped body-stabilized satellite has three-axis known as Roll, Pitch, and Yaw (RPY), which define the attitude of satellite. Thus, all three-axes pass through the center of satellite gravity. The Yaw axis is directed toward the Earth's center, the Pitch axis is normal to the orbital plane, and Roll axis is perpendicular to the other two. In fact, the Roll axis is tangential to the orbit and lies along the satellite velocity vector.

Materials in space are not subjected to gravitational stress or atmospheric corrosion and the effects of the space environment are not all benign by any means. The high vacuum causes some materials to sublime or evaporate and some to weld together on contact. The latter behavior means that special attention has to be given to the materials used for bearings. The basic materials for the main frame are aluminum or magnesium alloys and special plastic or fiber materials and for other components carbon fiber, epoxy resins, and carbon nanotube filaments are used.

2.4.7.2 Electric Power (EP)

The primary source of power for a communications satellite is the Sun. Hence, solar cells are used to convert energy received from the Sun into an electrical source. The principal components of the power supply system include: (1) The power electric generator, usually solar cell arrays, located on the spinning body of a spin-stabilized satellite or on the paddles for a three-axis stabilized satellite; (2) reliable electrical storage devices, such as batteries, for operating during periods of solar eclipses; (3) the electrical harness for conducting electricity to all of the devices demanding power; (4) the special converters and regulators delivering regulated voltage and currents to the devices on board the spacecraft; and (5) the electrical control and protection section is associated with the remote monitoring TT&C satellite system.

The solar arrays are the motor during entire life of the satellite, providing sufficient power to all active components. Occasionally, if the Earth or Moon blocks the Sun, then researchable batteries provide secondary electrical power.

A typical satellite power subsystem is shown in Fig. 2.37. Most components are duplicated for redundancy. On spinning satellites, the rings are between the power subsystem, on the rotating part, and the satellite bus.

Each cell delivers about 150 mA at a few hundred millivolts and an array of cells must be connected in series or in parallel together to give the required voltage and current for operating the equipment until the end of its life and to recharge the batteries when the satellite moves out of an eclipse. Charge is applied via the main electric power bus or a small section of the solar cell. In the course of exploitation, batteries are sometimes reconditioned by intentionally discharging them to a low charge level and recharging again, which usually prolongs their life.

The operational status of batteries including recharge, in-service, or reconditioning is remotely controlled by a special ground segment. The mass of a battery constitutes a significant portion of the total satellite mass. Therefore, a useful figure of merit to evaluate the performance of a battery is capacity in W/h per unit weight taken at the end of its life. Until recently virtually all satellites used Ni-Cd (Nickel-Cadmium) batteries because of their high reliability and long lifetime. These batteries provide a low specific energy of about 30–40 Wh/kg. The latest type of Ni-H (Nickel-Hydrogen) batteries can store at least 50 % more energy per kilogram.

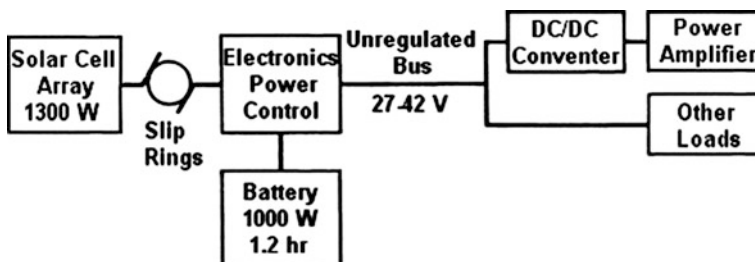


Fig. 2.37 Satellite electric power subsystem—Courtesy of Book: by Gordon

When a satellite passes through the Earth's shadow, the solar arrays stop producing power and the satellite structures use the energy from batteries. The GEO satellite undergoes around 84 eclipses in a year, with a maximum duration of 70 min. Thus, the eclipse occurs twice a year for 42 consecutive days each time. The percentage of eclipses' duration for GEO and HEO is much less than for lower satellite orbits. The LEO satellites can undergo several thousand eclipses in a year. For example, a LEO satellite in equatorial orbit at an altitude of 780 km can remain in the Earth's shadow for 35 % of the orbital period. For a MEO under similar conditions, the maximum eclipse duration would be about 12.5 % of the orbital period and a total duration of about 3 h a day, with about 4 eclipses per day. Otherwise, the Sun can also be sometimes eclipsed by the Moon's shadow, which is less predictable.

2.4.7.3 Thermal Control (TC)

There is not air at satellite orbit surrounded by very harsh space environment. The average satellite temperature is determined by the absorbed solar energy, different thermal radiation into space, and internal electric dissipation, what depends on the satellite shape and surface.

Thermal control of a communications satellite is very important factor during entire satellite lifetime, which is necessary to achieve normal temperature balance and proper performance of all subsystem. Thermal stress results in high-temperature effects from the Sun and low temperatures occurring during eclipse period. The obvious objective of the TC is to assure that the spacecraft structure and all equipment is maintained within temperatures that will provide successful operations. A satellite undergoes different thermal and other conditions during the launch and operational phase. The vacuum in space limits all heat transfer mechanisms to and from a spacecraft and its external environment to that of radiation. However, some main parts are usually in direct sunlight with a flux density of over 1 kW/m^2 , while other parts are facing the shadow side at a temperature of about -270°C . In addition, an eclipse causes temperature variation from around -180° to $+60^\circ\text{C}$, when the ambient temperature falls well below 0°C and rises rapidly from the moment the satellite emerges from the eclipse. All these extremes have to be eliminated or moderated for normal satellite operations, especially because all electronic devices need optimum temperatures between -5° and $+45^\circ\text{C}$.

These problems can be solved by remote TC techniques, using both passive and active means of controlling and regulating the temperature inside spacecraft. The passive means are simple and reliable, using surface finishes, filters, and insulation blankets. The active means are necessary to supplement the passive systems, which include louvers and blinds operated by bimetallic strips, heat pipes, thermal louvers, and different electrical heaters. Heat pipes are used to transfer heat from internal hot spots or devices to remote radiator surfaces or must be transported to the outside surface where it can be dissipated. On the other hand, special electric heaters are used to maintain minimum component or structure temperatures during cold

conditions. Accordingly, the TC subsystem ensures temperature regulation for optimum efficiency and satellite performance.

2.4.7.4 Attitude and Orbit Control (AOC)

The attitude and orbital control subsystem checks that a spacecraft is placed in its precise orbital position, and maintains, thereafter, the required attitude throughout its mission. Control is achieved by employing momentum wheels, which produce gyroscopic torques, combined with an auxiliary reaction control gas thruster system. Various sensors are employed to detect attitude errors, including Sun's initial orientation purposes.

The satellite antennas require AOC system that will keep them pointed always at the Earth, frequently within 0.1° and 0.01° . Figure 2.38 shows a block diagram of AOC system. The sensors direct any pointing errors and correct them by changing the speed or direction of a rotating wheel. The main performance specification of an AOC system is determined by the disturbance torques and the required pointing accuracy.

The AOC system performs satellite orientation and accurate orbital positioning throughout its lifetime, because loss of attitude renders a spacecraft useless. There are in use two common AOS, such as attitude control and orbit or station-keeping control systems. The objective of attitude control was to keep the antenna RF beam pointing at the intended areas on the Earth, which procedure involves as follows: (1) measuring the attitude of the satellite by sensors; (2) comparing the results of measurements with the required values; (3) calculating the corrections to reduce eventual errors; and (4) introducing these corrections by operating the appropriate torque units.

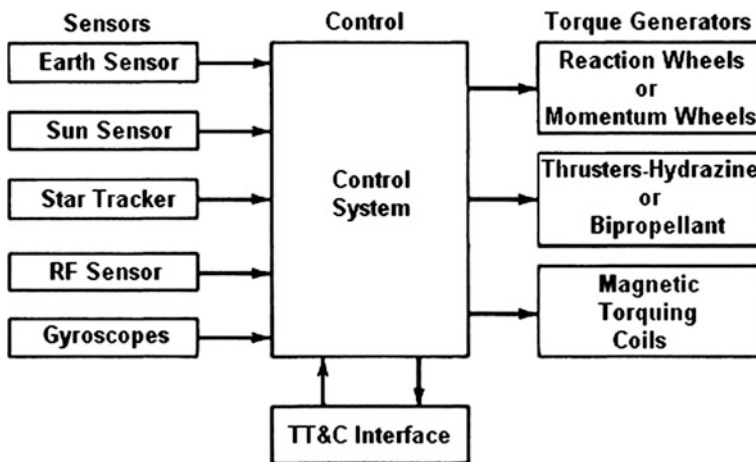


Fig. 2.38 Satellite AOC subsystem—Courtesy of Book: by Gordon

1. Attitude Control—Currently, all types of attitude stabilization systems have relied on the conservation of angular momentum in a spinning element, which can be classified into the two categories already mentioned, such as spin-stabilized and three-axis stabilization. The satellite is rapidly spun around one of its principal axes of inertia. Thus, in the absence of any perturbing torque, the satellite attains an angular momentum in a fixed direction in an absolute frame of reference. For the GEO satellite, the spin (pitch direction) axis must be parallel to the axis of the Earth's rotation. The perturbation torques reduce the spin of the satellite and they affect the orientation of the spin axis. The second system of attitude control is a body-stabilized design in a three-axis stabilized satellite, whose body remains fixed in space. This solution is the simplest method of attitude control using a momentum wheel, which simultaneously acts as a gyroscope, in combination with spin and drive stabilization. Certain perturbing torques can be resisted by changing its spin speed and the resulting angular momentum of the satellite.

2. Orbit or Station-keeping Control—Onboard propulsion requirements for both GEO and non-GEO are important to keep a satellite in the correct orbital attitude and position. For this reason, several types of propulsion systems are used, such as arc jet thrusters, ion and solar electrical propulsion, pulsed plasma thrusters, and iridium-coated rhenium chambers for chemical propellants. In order that the appropriate station-keeping corrections can be applied, it is essential that the orbit and position of a satellite are accurately determined. This may be done by making measurements of the angular direction and distance of the satellite from the Earth station, or a number of LES. When the orbit and position of the satellite have been determined, it is possible to calculate the velocity increments required to keep the N–S and E–W excursion of the satellite within the tolerated limits. The frequency with which N–S correction must be made depend on the maximum allowable value of the orbital inclination but the total increment required each year to cancel out the attraction of the Sun and Moon is 40–50 m/s. Otherwise, E–W station-keeping is usually achieved by allowing the satellite to drift toward the nearest point of equilibrium until it reaches the maximum tolerable error in longitude, then the process is repeated on the other side of the nominal longitude, and finally, the satellite drifts back once more toward the point of equilibrium, and the process is repeated. The frequency and magnitude of the velocity increments required depend on the angular distance between the satellite and the points of equilibrium and on the tolerable error, which is a maximum of about 2 m/s.

2.4.7.5 Telemetry, Tracking, and Command (TT&C)

The telemetry, tracking, command, and communication equipment enables data to be send continuously to the Earth stations, received from these stations and allows ground control stations to track the spacecraft and to monitor the health of the spacecraft and also to send commands to carry out various tasks such as switching the transponders in and out of service, and switching between redundant units.

The TT&C subsystem monitors and controls the satellite functions right from the liftoff stage to the end of its operational life in space. The TT&C subsystem is therefore very important, not only during orbital injection and the positioning phase, but also throughout the operational life of the satellite. During the orbital injection and positioning phase, the telemetry link is primarily used by the tracking system to establish the satellite-to-Earth station communication channel. After the satellite is put into the desired slot in its intended orbit, its mission is to monitor the health of various subsystems onboard satellite.

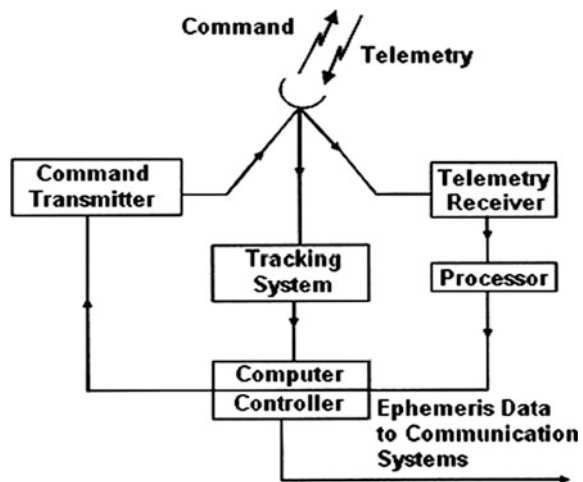
In Fig. 2.39 is shown the block schematic arrangement of the basic TT&C subsystem.

The TT&C system supports the function of spacecraft management for successful operation of payload and bus subsystems. The main functions of a TT&C are as follows:

1. **Telemetry Subsystem**—The function of telemetry is to monitor various spacecraft parameters and performances such as voltage, current, temperature, output from attitude sensors, reaction wheel speed, pressure of propulsion tanks and equipment status and to transmit the monitored data to the Satellite Control Center (SCC) on the Earth. At this point, the telemetered data are analyzed at the SCC and used for routine operational and failure diagnosis purposes, to provide data about the amount of fuel remaining, to support determination of orbital parameters, etc.

2. **Tracking Subsystem**—The function of tracking is to provide necessary sources to Earth stations for the tracking and determination of orbital parameters. In such a way, to maintain a satellite in its assigned orbital slot and provide look angle information to LES in the network, it is necessary to estimate the orbital parameters regularly. These parameters can be obtained by tracking the communications satellite from the ground and measuring its angular position and range. Most SCC employ angular and range or range-rate tracking to control satellite orbits.

Fig. 2.39 Satellite TT&C—
Courtesy of Book: by Maini



3. **Command Subsystem**—This subsystem receives commands transmitted from the ground SCC, verifies reception, and executes commands to perform various functions of the satellite during its operational mission, such as Satellite transponder and beacon switching, Antenna pointing control, Switch matrix reconfiguration, Controlling direction and speed of solar arrays drive, Battery reconditioning, Thruster firing, and Switching heaters of the various systems.

2.4.7.6 Propulsion Engine (PE)

The functions of the propulsion motors are to generate the thrust required for the attitude and orbital control of errors caused by solar and lunar gravity and other influences, or possibly the adequate assistance of the satellite into its final orbit.

Hence, these errors are normally corrected at set intervals in response to commands from SCC. The necessary impulse is provided by thrusters, which operate by ejecting hot or cold gas under pressure. The thrust requirements for orbital control are provided by mono- or bi-propellant fuels. The attitude control thrusters are positioned away from the center of the mass to achieve the maximum thrust, the thrust being applied perpendicular to the direction of a spacecraft's center of mass. The orbit control thrusters are mounted so that the thrust vector passes through the center of mass. The relocation of a satellite from transfer orbit into GEO may be performed by apogee boost motor. In some satellite configurations, this is achieved by a solid or liquid fuel engine. Moreover, the choice between these two motors has a significant effect on the internal arrangements of the satellite.

2.5 Intersatellite Links (ISL)

Intersatellite or cross-links are communication links between satellite constellations. Three types of intersatellite links can be considered:

1. GEO to LEO links between geostationary (GEO) satellites and low Earth orbit (LEO) satellites also called Interorbital Links (IOL);
2. GEO to GEO links between geostationary satellites; and
3. LEO to LEO links between low Earth orbit satellites.

Of course, one could consider intersatellite links between satellites in any type of orbit, but the above configurations are those most considered in practice. Only the transmission aspects are presented here.

An GEO satellite can cover about 40 % of the Earth's surface. Because each satellite sees so much of the Earth, no many commercial GEO satellite to date has an ISL. The ISL solution for GEO satellites is being considered for network connectivity, particularly for transfer of data, and generally to reduce ground infrastructure.

However, if connectivity over a large geographical area is required, the smaller footprint of LEO satellite needs that they have GES or Gateway in the coverage area of each satellite, with onboard store and forward capacity or ISL service. The Big LEO Globalstar has not ISL system, while Big LEO Iridium system using ISL optical links can offer service anywhere and anytime.

The ISL configuration may use either RF or optical technology for interconnections. Thus, a requirement for an optical or RF ISL to operate through solar conjunction, when the Sun is within the field of view of an optical or RF antenna, is a major ISL design driver. Operation through solar conjunction is possible with both microwave and optical systems, but is actually easier with an optical system.

Optical ISL has very high antenna gain because of their short wavelength. Laser diodes have a frequency of the order of $1-6 \times 10^{15}$, or about 105 higher than a 30 GHz RF wave. The optical antenna gain is 100 dB more than a RF antenna of the same size and efficiency. Moreover, a 10-cm-diameter optical antenna has 80 dB more gain than a 1 m RF antenna. This allows a 1 W laser diode to provide the same flux density as a 100 MW RF source. Optical detection is less efficient than RF detection so the above overstates the actual comparison, but gives a clear idea of the advantages of an optical system. Thus, to appreciate the narrowness of the optical beam, a 1-micron laser using a 10-cm antenna would have a bandwidth of about 1 milliradian. The very narrow bandwidth requires a closed-loop tracking system, where each ISL tracks the signal from the communicating satellite.

The choice between radio and optical links depends on the mass and power consumed. In general terms, it can be said that the advantage is with radio links for low throughputs (less than 1 Mb/s). For high-capacity links (several tens of Mb/s) optical links command attention. For a link involving one uplink, one or more optical intersatellite links and one downlink, the overall station-to-station link performance should be established in the same way as the overall link performance for regenerative satellites. Indeed, the implementation of intersatellite links, be it at radio frequency or with optical technology, is mainly of interest when onboard demodulation is available, so as to provide flexible onboard switching.

2.5.1 Direct ISL Data Transmission Over GEO Satellite

In space activities, there is a stronger need of exploring and making usage of advanced technologies to transmit data directly in cross-link from one satellite to another, both conventionally using radio waves or in a revolutionary and new way using laser beams.

Earth observation satellites, for instance, cannot send their data to just one Earth station, because they fly at low altitude of about 800 km. As they scan the Earth's surface, they disappear behind the horizon after a few minutes at the most. The data must therefore be stored in onboard memories, which are far from fail-safe or else transmitted to a large number of distributed Earth stations, which are costly to operate.



Fig. 2.40 ISL of GEO spacecraft Artemis—Courtesy of Webpage: by ESA

From its vantage point in GEO at 36,000 km above the Earth such as the ESA Artemis spacecraft can be in constant communication with satellites in LEO orbit for long periods of each orbit and can beam data from those satellites directly to the users, illustrated in Fig. 2.40. To demonstrate the reception of data from other satellites and their onward transmission to users, Artemis is carrying an advanced radio data-relay payload and a revolutionary laser data-relay payload called Semiconductor Laser Intersatellite Link Experiment (SILEX). SILEX is the world's first civil intersatellite data-relay system using lasers as carriers for the signal transmission.

Using laser communications, very high data rates can be achieved via small terminals on LEO satellites, such as existing Iridium. The radio data-relay payload SKDR (S/Ka-band Data Relay) features the use of two frequencies for relaying data at low, medium, or high rates again both ways. The first users of SKDR include ESA's environmental ENVISAT-1 satellite scheduled for launch in 1999. Under an agreement with NASDA, the Japanese space agency, the ARTEMIS satellite, will also serve several Japanese spacecraft and the Japanese module of the International Space.

2.5.2 Radio Frequency (RF) ISL

Table 2.7 indicates the frequency bands allocated to ISL by the radiocommunication regulations. These frequencies correspond to strong absorption by the atmosphere and have been chosen to provide protection against interference between intersatellite links and terrestrial systems. However, these bands are shared with

Table 2.7 Frequency bands for ISL

Intersatellite service	Frequency bands
Radio frequency	22.55–23.5 GHz
	24.45–24.75 GHz
	32–33 GHz
	54.25–58.2 GHz
Optical	0.8–0.9 μm (AlGaAs laser diode)
	1.06 μm (Nd:YAG laser diode)
	0.532 μm (Nd:YAG laser diode)
	10.6 μm (CO_2 laser)

other space services and the limitation on interference level is likely to impose constraints on the choice of the defining parameters of intersatellite links (CCIR Reports 451, 465, 874, 951). The same table also indicates the wavelengths envisaged for optical links. These result in the transmission characteristics of the components.

Propagation losses reduce to free-space losses since there is no passage through the atmosphere. Antenna pointing error can be maintained at around a tenth of the beamwidth and this leads to a pointing error loss of the order of 0.5 dB. The antenna temperature in the case of a GEO–GEO link, in the absence of solar conjunction, is of the order of 10 K. Table 2.8 indicates typical values for the terminal equipment. For practical applications, antenna dimensions are of the order of 1–2 m. In such a way, considering a frequency of 60 GHz and transmission and reception losses of 1 dB leads to:

- A receiver figure of merit G/T of the order of $25\text{--}29\text{ dBK}^{-1}$; and
- A transmitter EIRP of the order of 72–78 dBW.

Because of the relatively wide beamwidth of the antenna (0.2° at 60 GHz for a 2 m antenna), establishing the link is not a problem. Each satellite orientates its receiving antenna in the direction of the transmitting satellite with a precision of the order of 0.1° to acquire a beacon signal, which is subsequently used for tracking. The development of high capacity RF ISL between GEO satellite systems implies reuse of frequencies from one beam to another. In view of the small angular separation of the satellites, it is preferable to use narrow-beam antennas with reduced side lobes in order to avoid interference between systems. Consequently, and in view of the limited antenna size imposed by the launcher and the technical complexity of the deployable antennas, the use of high frequencies is indicated. The use of optical links may be usefully considered in this context.

Table 2.8 Typical values for terminal equipment of an ISC

Receiver noise factor (dB)	Transmitter power (W)
3–4.5	150
4.5	75
9	30

2.5.3 Optical ISL

Except RF links, ISL can use optical or laser links as well.

In comparison with radio links, optical links have specific characteristics, which are important for establishment a link.

2.5.3.1 Establishing an Optical ISL

Two aspects should be indicated to establish an optical ISL:

1. The small diameter of the telescope is typically of the order 0.3 m. In this way, one is freed from congestion problems and aperture blocking of other antennas in the satellite payload.
2. The narrowness of the optical beam is typically 5 microradians. Notice that this width is of several orders of magnitude less than that of a radio beam, and this is an advantage for protection against interference between systems. But it is also a disadvantage since the beamwidth is much less than the precision of satellite attitude control (typically 0.1° or 1.75 mrad). Consequently, an advanced pointing device is necessary; this is probably the most difficult technical problem.

There are three basic phases to optical communications:

1. Acquisition: The beam must be as wide as possible in order to reduce the acquisition time. But this requires a high-power laser transmitter. A laser of lower mean power can be used which emits pulses of high peak power with a low duty cycle. The beam scans the region of space where the receiver is expected to be located. When the receiver receives the signal, it enters a tracking phase and transmits in the direction of the received signal. On receiving the return signal from the receiver, the transmitter also enters the tracking phase. The typical duration of this phase is 10 s.

2. Tracking: The beams are reduced to their nominal width. Laser transmission becomes continuous. In this phase, which extends throughout the following, the pointing error control device must allow for movements of the platform and relative movements of the two satellites. In addition, since the relative velocity of the two satellites is not zero, a lead-ahead angle exists between the receiver and the transmitter LOS. On the other hand, as demonstrated below, the lead-ahead angle is larger than the beamwidth and must be accurately determined.

3. Communications: Information is exchanged between the two ends.

Otherwise, the narrow optical beam gives rise to several problems for optical cross-links as:

1. **Difficult acquisition**—The very narrow beam requires very precise knowledge of the location of the two satellites establishing the link and the special acquisition mode.

2. **Sensitivity to spacecraft vibration**—A fast-tracking system, of the order of 100 Hz, is necessary so that naturally occurring vibrations, from thruster valves, momentum wheels, solar array devices and so on, do not degrade performance.

As both satellites are on the same circular orbit, shown in Fig. 2.39 (Right), the velocity vectors V_{S1} and V_{S2} , which are tangential to the orbit, have equal modulus:

$$|V_{S1}| = |V_{S2}| = \omega(R_0 + R_E) = 3075 \text{ m/s} \quad (2.62)$$

where ω is the angular velocity of a GEO = 7.293×10^{-5} rad/s; R_0 is the altitude of a GEO satellite = 35,786 km; and R_E is the Earth radius = 6378 km.

The component vectors V_{T1} and V_{T2} , perpendicular to the line joining S1 and S2 at time t , both lie in the plane of the orbit and are opposite. They are at an angle $(\pi/2 - \alpha/2)$ with respect to vectors V_{S1} and V_{S2} . Therefore, following these values is giving relation:

$$|V_{T1} - V_{T2}| = 2\omega(R_0 + R_E) \cos(\pi/2 - \alpha/2) = 2\omega(R_0 + R_E) \sin(\alpha/2) [\text{m/s}] \quad (2.63)$$

This equation is possible to express the following relation:

$$\beta = 2/V_{T1} - V_{T2}/c = 4\omega(R_0 + R_E) \sin(\alpha/2)/c [\text{rad}] \quad (2.64)$$

At this point, the lead-ahead angle β as a function of the separation angle α between the two GEO satellites. Here is important to note that for a separation angle larger than 15° , the lead-ahead angle is larger than the beamwidth (typically 5 microradians). For instance, $\beta = 10.6$ microradians for $\alpha = 30^\circ$, $\beta = 20.5$ microradians for $\alpha = 60^\circ$, and $\beta = 35.5$ microradians for $\alpha = 120^\circ$.

2.5.3.3 ISL Relations Between GEO and LEO Satellites

The relative velocity of the two satellites varies with time and so does the value of the lead-ahead angle, as shown in Fig. 2.42. Its maximum value is obtained when the LEO satellite crosses the equatorial plane. Denoting as i the LEO satellite orbit inclination, then:

$$|V_{T1} - V_{T2}| = \left\{ |V_{S1}|^2 + |V_{S2}|^2 - 2|V_{S1}||V_{S2}| \cos i \right\} \quad (2.65)$$

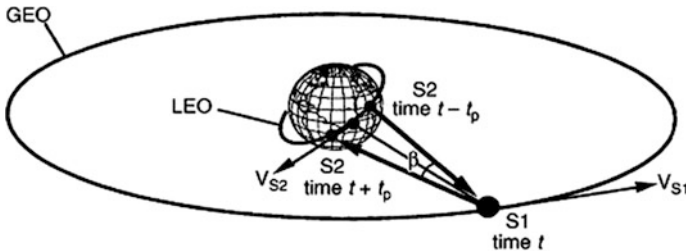


Fig. 2.42 Lead-ahead angle for link between GEO and LEO satellites—Courtesy of Book: by Maral

Using this equation is possible to determine both values with the following relations:

$$\begin{aligned} |V_{S1}| &= \omega_{\text{GEO}}(R_0 + R_E) = 3,075 \text{ [m/s]} \\ |V_{S2}| &= \omega_{\text{LEO}}(h + R_E) \end{aligned} \quad (2.66)$$

where h is the LEO satellite altitude and $\omega_{\text{LEO}} + \mu^{1/2} (h + R_E)^{-3/2}$ is the LEO satellite angular rate ($\mu = 3986 \times 10^{14} \text{ m}^3/\text{s}^2$). From Eq. 2.56, the lead-ahead angle is given by:

$$\beta = 2/V_{T1} - (2/c) \left\{ |V_{S1}|^2 + |V_{S2}|^2 - 2|V_{S1}||V_{S2}| \cos I \right\} \text{ [rad]} \quad (2.67)$$

The lead-ahead angle is the same for the two satellites. Considering $i = 98.5^\circ$ and $h = 800 \text{ km}$, $\beta = 57$ microradians. Note this value is even larger than for intersatellite links between two geostationary satellites.

2.5.4 Transmission and Reception of Optical Sources

The ISL relay link from a LEO satellite to the ground via high-altitude platforms (HAP) relay and from a LEO satellite over a GEO satellite and HAP is presented in Fig. 2.43.

The “last mile” downlink from the HAP or stratospheric communication platform (SCP) to GES would be either bridged by a high-bandwidth optical or microwave link.

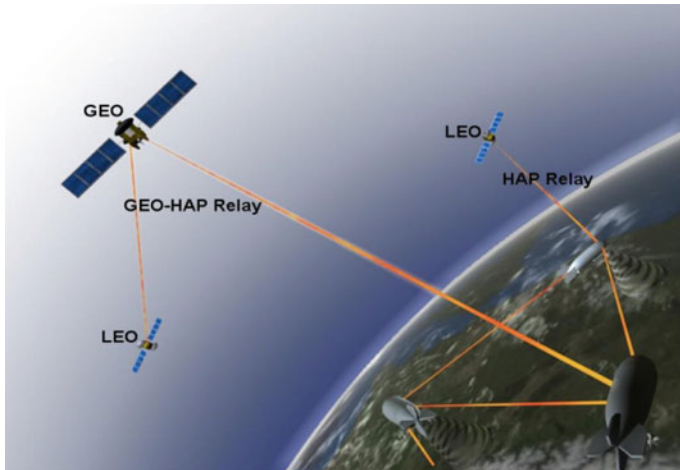


Fig. 2.43 ISL between GEO, LEO, and HAP—Courtesy of Paper: by Knappek

Optical or laser sources operate in single- and multifrequency modes. In single frequency mode, spectral width varies between 10 kHz and 10 MHz. In multifrequency mode, it is from 1.5 to 10 nm. The power emitted depends on the type of laser. Modulation can be internal or external. Internal modulation implies direct modification of the operation of the laser. External modulation is a modification of the light beam after its emission by the laser. The intensity, the frequency, the phase, and the polarization can be modulated. Phase and polarization modulation are external. Intensity and frequency modulation can be internal or external. Polarization modulation requires the presence of two detectors in the receiver, one for each polarization. Because of this, it is preferable to reserve polarization for multiplexing of two channels. The intensity distribution of a laser beam, as a function of angle with respect to the maximum intensity, follows a Gaussian law. If total beamwidth is reduced, there is benefit in gain but the pointing error loss increases. In general, for a pointing error of any kind, the beamwidth may be adapted to the pointing error. In addition to losses due to pointing error, transmission losses and degradation of the wavefront in the emitting optics occur.

The receiver can be of a direct detection or a coherent detection receiver. With direct detection, the incident photons are converted into electrons by a photodetector. The subsequent baseband electric current at the photodetected output is amplified then detected by a matched filter. With coherent detection, the optical signal field associated with the incident photons is mixed with the signal from a local laser. The resulting optical field is converted into a bandpass electric current by a photodetector and is subsequently amplified by an intermediate frequency amplifier. The demodulator detects the useful signal either by envelope detection or by coherent demodulation. The receiving losses include optical transmission losses and, for coherent detection, losses associated with the degradation of the wavefront. The quality of the wavefront is an important characteristic for optimum mixing of the received signal field and the local oscillator field at the photodetector frontend. Filtering, to reject out-of-band photons, also introduces losses, since the transmission coefficient reduces with bandwidth. A typical filter width is from 0.1 to 100 nm.

2.5.5 Iridium ISL and Mobility System

Figure 2.44 illustrates a typical satellite communication setup of the Iridium Big LEO constellation ISL and mobility system. In distinction from the Globalstar Big LEO, Iridium is providing optical ISL, which improves connections between users and GES terminals, although there is not direct LOS. The Iridium ISL system employs frequencies of 22.55–23.55 GHz as these RF are heavily absorbed by water and hence are not useful for establishment Earth to satellite links.

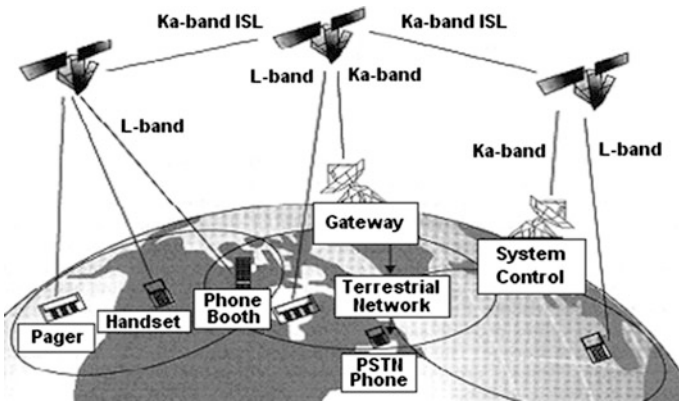


Fig. 2.44 Iridium big LEO ISL system—Courtesy of Book: by Maini

2.5.5.1 LEO Satellite ISL

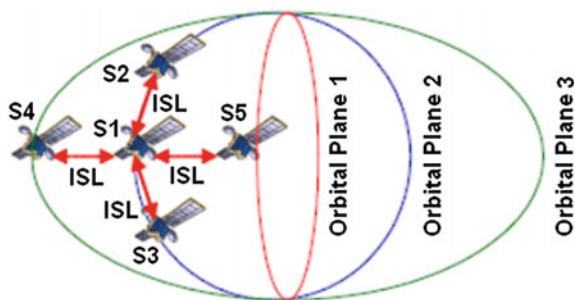
The LEO network topology is constructed by the ISL, which establish a network between satellites. In order to establish ISL network, each satellite needs to have extra equipment, including transceivers and antennas. This additional equipment will increase the weight and the cost of the satellite, but on the other hand a satellite system that has this ISL features does not require a GES to establish a long distance connection. This advantage reduces the dependency of the satellite network on terrestrial systems. As stated earlier, in the satellite constellation, there are two types of ISL solutions.

The permanent ISL between satellites in the same plane (intraplane satellite link) and semi permanent between satellites in the neighboring planes (interplane satellite link) will function as edges in LEO satellite constellation with LEO satellites as the nodes.

As illustrated in Fig. 2.45, ISL between satellites 1 (S1), satellite 2 (S2), and ISL between satellite 1 (S1) and satellite 3 (S3) in orbital plane 1 are interplane ISL. The ISL between S1 in orbital plane 2, S4 in orbital plane 3 and the ISL between S1 and S5 in orbital plane 1 are intraplane ISL. Iridium is the first satellite communication system that provides ISL. The ISL system uses radio or laser media for their direct communication for mobile and semi-fixed systems. Iridium uses a GSM-based telephony architecture, and a geographically controlled system access process. Each satellite is connected to its four neighboring satellites through their ISL network. Connections between the IRIDIUM network and the Public Switched Telephone Network (PSTN) are provided via GES installations or Gateway. By having ISL the location of the GES can be flexibly located.

As stated, satellite links in the LEO constellation are constructed from two different types of ISL solutions. The first one is the interplane satellite connection, which is constructed between two satellites in the same plane, whereas interplane satellite connection is an ISL which is constructed between two neighboring

Fig. 2.45 Iridium big LEO ISL system—Courtesy of Theses: by Septiawan



satellites in a deferent plane. The intraplane satellite connection remains continuous for the whole period of time, while interplane satellite connection in some LEO topology will be turned on for a period of time. For example, in the near-polar constellation, intraplane satellite connection is switched on in the polar region, to reduce the signal interference. In the case of Iridium intraplane, ISL will only be maintained between latitudes of approximately 60° North or South of the equator. A first-degree ISL only provides a connection between satellites with its direct neighbors. A second-degree ISL will provide a connection between satellites with not only its direct neighbor satellites, but also with the next neighbor satellites.

Besides intersatellite handover, there is another type of satellite handover, which is called intrasatellite handover. This is a handover between spot beams in one satellite or handover occurs between satellites in the same plane. Interplane satellite handover: This type of handover occurs when a connection is handed over to another satellite in another plane, due to the unavailability of a satellite in the same plane, or due to the unavailability of satellite channels to accommodate this connection. The last type of handover is a handover that occurs once the ISL systems between satellites (either on the seam region or near polar region) are turned off.

2.5.5.2 LEO Satellite Handover

The mobility management in LEO satellite network is specific because a LEO satellites movement is relatively higher than the movement of any object on Earth, when an LEO satellite becomes the moving object in this context, while the mobile user on Earth has a relatively fixed position. The LEO mobility aspect is more concerned with the movement of the satellite footprint on Earth, in which an on-going connection of MES users has to be handed over from one satellite to another. Thus, these transitions have to be smooth and seamless and mobility management has to be able to maintain any on-going network connections and perform a handover process if needed. There is an advantage in a LEO satellites environment, in which the movement of the LEO satellite is roughly predictable.

Prior to the disappearance of viewed satellite, a new connection needs to be established. A handover procedure to the new approaching satellite needs to be

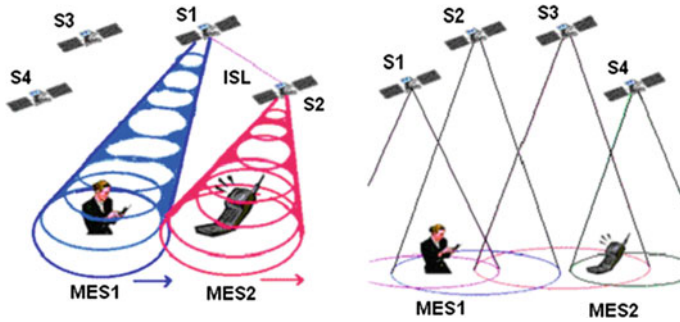


Fig. 2.46 Handover between neighboring satellites and soft handover procedure—Courtesy of Theses: by Septiawan

constructed. In case there is only one visible satellite at a time, the handover procedure needs to be initiated before the currently used satellite becomes invisible. This case is shown in Fig. 2.46 (Left), where a connection between MES1 and MES2 is constructed using satellite 1 (S1) and satellite 2 (S2), via ISL between these two satellites. The MES1 unit is currently inside the service coverage of S1, while MES2 is located inside S2 service coverage. Both satellites move in clockwise direction, so MES2 starts losing service coverage from S2 and starts to obtain S1 coverage. Since currently MES2 is only covered by S2, it needs to start the handover procedure to allocate the connection to the next available satellite S1.

In the case that two or more LEO satellites are visible at the same time, a soft handover can be used to allocate the ongoing call into another visible satellite, which has a better signal performance or better visibility than the current one. This is shown in Fig. 2.46 (Right), when MES2 is currently in the coverage of S3 and S4. Soon, S3 and S2 will cover MES2. Every time there should be two visible satellites for each user. MES2 compares signal strength between these two satellites and chooses the best satellite to build a connection.

Therefore, due to the rapid movement of satellites, MES terminals can only use the same LEO satellite for short periods of times and before they lose their LOS to the current satellite, their connection must be handed over to the next available satellite.

It is necessary to conclude that a handover is initiated from one satellite to another satellite. The term handover is used in US cellular standard documents, while in ITU documents the term handover is used as well. Both terms have the same meaning. Handover in LEO satellite systems differs from the one in the cellular terrestrial system in term of the mobile and fixed units. In the LEO satellite systems, handover means a procedure of changing the assignment of a fixed unit, a MES terminal on Earth, from one mobile unit, a LEO satellite, to another as the LEO satellites move, while in the terrestrial cellular system a handover means a procedure of changing the assignment of a mobile unit from one fixed unit (Base Station) to another, as the mobile unit moves permanently.

The time in which a satellite is visible from a MES is called the sliding window of this satellite constellation. The visibility period of a satellite is defined as the maximum time duration that a MES is located inside a footprint and can directly communicate with that satellite (about 15 min). The satellite footprint is divided into several spot beams, and each spot beam can use several deferent frequencies. Because the spot beam coverage area is much smaller than the footprint coverage area, the maximum visibility of a spot beam is around 1–2 min. This means that a spot beam handover occurs more frequently than satellite handover. A handover can be initiated by the network, in which the decision is made by network measurements of received signals from MES on the ground. Alternatively, MES on the ground can provide a feedback to the LEO satellite concerning the signal received at the MES user. Various performance metrics may be used in making a decision, such as call dropping probability due to the handover itself, probability of unsuccessful handover due to an execution of a handover with an inadequate reception condition, call blocking probability due to unavailable capacity, rate of handover the maximum number of handovers per unit time, and handover delay due to the distance of the point that handover should occur and the point that the handover does occur.

Stallings outlines several handover strategies, which firstly suggests a handover can be initiated by measuring relative signal strength between two LEO satellites and the MES user on the ground. The second strategy depends on threshold signal strength. A handover is initiated when the signal strength between the LEO satellite and the MES user is lower than a certain threshold value. In the third strategy, a handover can only occur if the signal strength of the new LEO satellite is stronger by a margin than the current signal strength. The last strategy uses by using prediction techniques. The handover decision is based on the expected future value of the received signal depending on the movement of LEO satellites. A satellite handover takes place when an ongoing connection needs to be handed over from one satellite to another satellite. There are three types of satellite handovers:

1. **Intra-satellite Handover**—Handover occurs between satellites in the same plane. The Gateway monitors the signal strength and the MES unit position relative to the satellite. Since the Gateway has information about the MES unit positions and the satellite positions, when the currently used satellite moves away from the MES unit, the Gateway will contact the next available satellite in the same plane, to replace the currently used satellite.

2. **Interplane Satellite Handover**—It occurs when a connection is handed over to another satellite in another plane, due to the unavailability of a satellite in the same plane, or due to the unavailability of satellite channels to accommodate this connection.

3. **Interbeam Handover**—This handover occurs while the MES decides to use another frequency in the adjacent candidate beams, due to the weakness of RF signal power from the used frequency. Once the RF power strength in the current beam is lower than the MES initiates, a handover requests to hand over a user to the new beam.

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