

## Chapter 2

# Space Segment

The space segment of an artificial satellite system is one of its three main operational components, the other two being the ground and user segments. It comprises the satellite or satellite constellation and the uplink and downlink satellite links. The overall design of the operational satellite with payload, transponders and satellite bus, ground segment with transmission and antenna systems and end-to-end system of users is a complex task. Satellite communications payload design must be properly coupled with the capabilities and interaction with the spacecraft bus that provides power, stability and environmental support to the payload.

During the last six decades, commercial communication, meteorological, navigation and other satellite networks have utilized Geostationary Satellite Orbits (GEO) extensively to the point where portions of this orbit have become crowded and coordination between satellites is becoming constrained. Recently, Non-GEO satellite systems have grown in importance, both because of their orbit characteristics and Earth coverage capabilities in high latitudes. Some of their special features are described in this section.

This chapter describes orbital mechanics and their significance with regard to satellite use for weather observations and basic physical principles that reveal the shape of a satellite orbit and how to orient the orbital plane in space. Namely, to fully understand and use satellite meteorological data it is necessary to understand the orbits in which satellites are constrained to move and the space geometry with which they view the Earth. This knowledge allows us to calculate the position of a satellite at any time, thus orbit perturbations and their effects on meteorological satellite orbits are then discussed. Next the geometry of satellite tracking and Earth locations of the measurements made from the satellites are explored. This leads to a discussion of space-time sampling.

This theoretical section also describes satellite orbits and their significance with regard to the meteorological observation, the fundamental laws governing satellite orbits and the principal parameters that describe the motion of artificial satellites of the Earth. The types of orbits are also classified and compared from a

communication and meteorological system viewpoint in terms of Earth coverage performance and environmental and link constraints.

More than five decades ago were developed Polar Earth Orbit (PEO) satellites as a first solution for meteorological and navigation satellite applications. Soon later, many other type of satellite networks, such as communication and meteorological, have utilized GEO satellites extensively to the point where portions of these orbits have become crowded and coordination between satellites is becoming constrained. However, PEO as Leo Earth Orbit (LEO) satellite systems have grown in quite importance, both because of their orbit characteristics and coverage capabilities in high latitudes. In addition, can be used High Elliptic Orbits (HEO) and Medium Earth Orbits (MEO) for modern satellite meteorological observation. 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 weather observation viewpoint in terms of coverage and link performances.

## **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 meteorological 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 to identify the most suitable orbit for the objective metrological 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.

### ***2.1.1 Space Environment***

The satellite service begins when a spacecraft is located as a 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 known as the atmosphere, whose density decreases as the altitude increases. 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 friendly and is extremely destructive, mainly because there is no atmosphere, the cosmic radiation is powerful, the vacuum creates high pressure on spacecraft or other bodies and there is the negative influence of 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 History of Motions in Space***

The modern orbit types have been developed based on theories dating back centuries. The early Greeks initiated the orbital theories, postulating that the Earth was fixed, with the planets and other celestial bodies moving around it forming a geocentric universe. About 300 BC, however, Aristarchus of Samos suggested that the Sun was fixed and the planets, including Earth, were in circular orbits around it forming a heliocentric universe. Although Aristarchus was more correct (at least about a heliocentric solar system), his ideas were too revolutionary for that time. Other prominent astronomers and philosophers were held in higher esteem, and since they favored the geocentric theory, so Aristarchus's heliocentric theory was rejected, and the geocentric theory continued to be predominately accepted for many centuries.

In the year 1543, some eighteen centuries after Aristarchus proposed a heliocentric system, a Polish monk named Nicolas Koppernias (better known by his Latin name, Copernicus) revived the heliocentric theory when he published “*De Revolutionibus Orbium Coelestium* (On the Revolutions of the Celestial Spheres)”. This work represented an advance, but there were still some inaccuracies. For example, Copernicus thought that the orbital paths of all planets were circles around the center of the Sun.

Tycho Brahe established an astronomical observatory on the island of Hven in 1576. For 20 years, he and his assistants carried out the most complete and accurate astronomical observations of the period. However, Brahe did not accept Copernicus’s heliocentric theory and instead believed in a geo-heliocentric model that had the Moon and Sun revolving around the Earth, while the rest of the celestial bodies revolved around the Sun.

German astronomer Johannes Kepler, born in 1571, wondered why there were only six planets and what determined their separation. His theories required data from observations of the planets, and he realized that the best way to acquire such data was to become Brahe’s assistant.

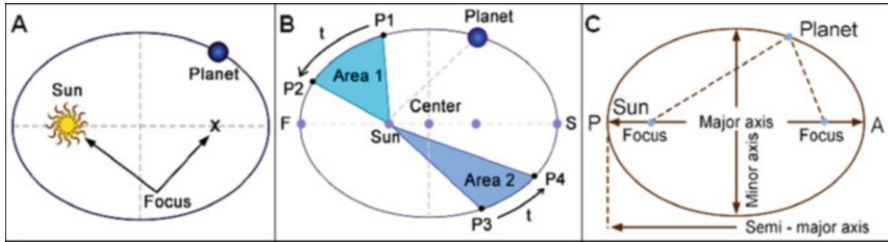
In 1600, Brahe set Kepler to work on the motion of Mars. This task was particularly difficult because Mars’s orbit was the second most eccentric (of the then-known planets) and defied the circular explanation. After Brahe’s death in 1601, Kepler finally discovered that Mars’s orbit (and that of all planets) was represented by an ellipse with the Sun at one of its foci.

### ***2.1.3 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 or any star. 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 Kepler law is also known as The Law of Orbits. As discussed earlier, an ellipse has two foci in which one can be a real with Sun or any star in the focus, and other is unreal. While studying the motion of planets around the Sun, Kepler explained that the path followed was elliptical, with the



**Fig. 2.1** Kepler's laws of satellite motion (Courtesy of Manual: by Ilcev)

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”, which is illustrated in Fig. 2.1 (a). This indicates that the Sun is in one real focus, while the other focus is known as the vacant or empty focus. As seen 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).

2. **Second Law** – The second Kepler law is also known as The Law of Equal Areas, shown in Fig. 2.1 (b). 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 draw 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 Kepler law of planetary motion in ellipse with Perigee (P) and Apogee (A) is alternatively known as The Law of Periods and Harmonic Law, depicted in Fig. 2.1 (c). This law relates the time required by a planet to make a complete trip around the Sun to its mean distance from the Sun. This law 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 centre and when its orbits are not affected by other systems. However, these conditions are not completely fulfilled in the case of Earth 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, 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, Sir Isaac Newton published his breakthrough work "Principia Mathematica" with own syntheses, known as the Three Laws of Motion:

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:

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

where parameter  $m$  = mass of the satellite body;  $t$  = time of satellite orbit;  $R$  = equatorial radius of the Earth ( $6.37816 \times 10^6$  m);  $h$  = altitude of satellite above the Earth's surface;  $G$  = Universal gravitational constant ( $6.67 \times 10^{-11}$  N m<sup>2</sup>/kg<sup>-2</sup>);  $M$  = Mass of the Earth body ( $5.976032 \times 10^{24}$  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.

In such a way, considering that the simple circular orbit and assuming that the Earth is a sphere, it is reasonable that it can be simply treated as 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.

Therefore, 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 these two forces will give the following 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. (2.3) 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. So, substituting in Eq. (2.4) 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:

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

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 an GEO is 42,164 km or about 35,786 km above the Earth's surface.

### 2.1.3.1 Geometry of Elliptical Orbit

The satellite in circular orbit undergoes its revolution at a fixed altitude and velocity, while a satellite in an elliptical orbit can drastically vary its altitude and velocity during one revolution. The elliptical orbit is also subject 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 (II) 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:

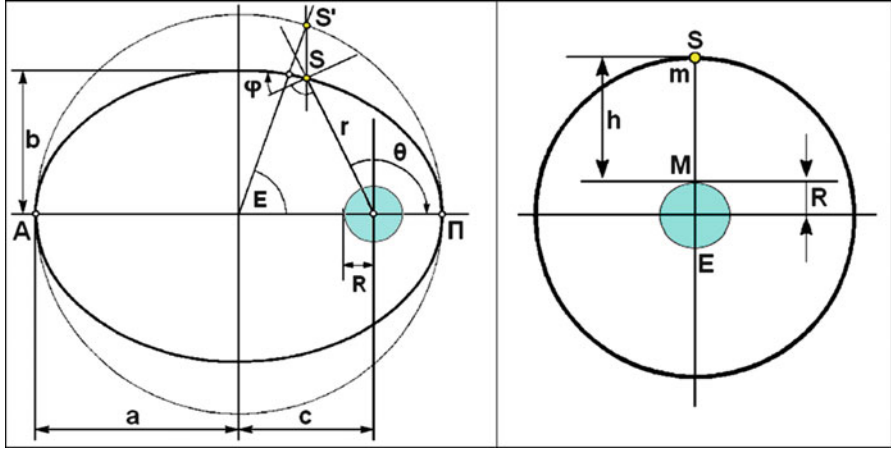
$$\begin{aligned} e &= c/a = \sqrt{1 - (b/a)^2} \text{ or } e = (\sqrt{a^2 - b^2}/a) & p &= a(1 - e^2) \text{ or } p = b^2/a \\ c &= \sqrt{(a^2 - b^2)} & a &= p/1 - e^2 & b &= a\sqrt{1 - e^2} \end{aligned} \quad (2.8)$$

where  $e$  = eccentricity (distance from the centre of the ellipse to one focus/semimajor axis);  $a$  = large semi-major axis of ellipse;  $b$  = small semi-major axis of ellipse;  $c$  = axis between centre of the Earth and centre of ellipse and  $p$  = focal parameter. The equation of ellipse derived from polar coordinates can be presented with the resulting trajectory equation as:

$$r = p/1 + e \cos \Theta \text{ [m]} \quad (2.9)$$

where  $r$  = distance of the satellites from the centre of the Earth ( $r = R + h$ ) or radius of path;  $\Theta$  = true anomaly or  $\vartheta$  = 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  $0^\circ$  to  $360^\circ$ , between the direction of the perigee and the direction of the satellite (S).





**Fig. 2.2** Elliptical and circular satellite orbits (Courtesy of Book: by Galic)

The position of the satellite can also be defined by eccentric anomaly ( $\Theta$ ), 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  $a = 0.5 (40,000 + 500 + 2 \times 6378.16) = 26,628.15 \text{ km}$ .

### 2.1.3.2 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 Law, the solar time ( $\tau$ ) in relation with the right ascension of an ascending node angle ( $\Omega$ ); 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:

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

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  $360^\circ$  for successive passages of the Sun over a point  $0.986^\circ$  further. On the other hand, a sidereal day is the time required for the Earth to rotate one circle of  $360^\circ$  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  $h = 35,786.04 \times 10^3 \text{ (m)}$ . 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:

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

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 than 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  $t / \tau$  values

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

Parameter	Values of time					Unit
t	86,164.00	43,082.05	21,541.23	10,770.61	6052.00	s
h	35,786.00	20,183.62	10,354.71	4162.89	800.00	km
(R + h)	42,164.00	26,561.78	16,732.87	10,541.05	7178.00	km
v	3075.00	3873.83	4880.72	5584.12	7450.00	km/s <sup>-1</sup>

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.

### 2.1.4 Horizon and Geographic Satellite Coordinates

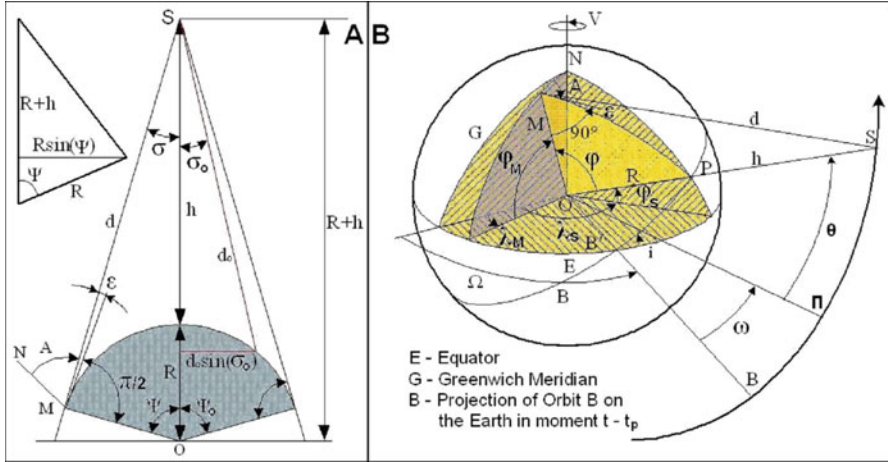
The geographical and horizon coordinates are very important to find out many satellite parameters and equations for better understanding the problems of orbital plane, satellite distance, visibility of the satellite, coverage areas, etc. The coverage areas of a satellite are illustrated in Fig. 2.3 (a) with the following geometrical parameters: actual altitude (h), radius of Earth (R), angle of elevation ( $\epsilon$ ), angle of azimuth (A), distance between satellite and the Earth's surface (d) and central angle ( $\Psi$ ) and sub-satellite angle or declination ( $\delta$ ), which is similar to the angle of antenna radiation.

The geographical and horizon coordinates of a satellite are presented in Fig. 2.3 (b) with the following, not yet mentioned, main parameters: angular speed of the Earth's rotation ( $\nu$ ), argument of the perigee ( $\omega$ ), moment of satellite pass across any point on the orbit ( $t_o$ ), which can be perigee ( $\Pi$ ), projection of the perigee point on the Earth's surface ( $\Pi'$ ), spherical triangle ( $B'TP$ ), satellite (S), the Point of the Observer or Mobile (M), latitudes of observer and satellite ( $\varphi_M$  and  $\varphi_S$ ), longitudes of observer and satellite ( $\lambda_M$  and  $\lambda_S$ ), 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  $t_o$  ( $\Omega_o$ ).

Otherwise, the right ascension of an ascending node angle ( $\Omega$ ) is the angle in the equatorial plane measured counter clockwise from the direction of the vernal equinox to that of the ascending node, while the argument of the perigee ( $\omega$ ) is the angle between the direction of the ascending node and the direction of the perigee.

#### 2.1.4.1 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 Ground Earth Station (GES) or Fixed Earth Station (FES) and geographic coordinates. Thus, this relation is very simple in the



**Fig. 2.3** Geometric projections of satellite orbits (Courtesy of Manual: by Solobev/Zhilin)

case where the sub-satellite point is in the centre of coverage, while all other samples are more complicated. Thus, the angle of an 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 sub-satellite 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: **(1)** Coverage area or the portion of the Earth's surface that can receive the satellite transmission 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. 2.3 (a) a covered area expressed with central angle ( $2\delta$  or  $2\Psi$ ) 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, such as Mobile Earth Station (MES), and a satellite at altitude ( $h$ ) and elevation angle can be defined in this way:

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

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

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

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  $\pm 40$  km, a time delay of about  $\pm 133 \mu s$  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 ( $\varepsilon$ ) 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 ( $\Omega$ ) and argument of the perigee ( $\omega$ ) are as follows:

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

The limit of the coverage area is defined by the elevation angle from GES 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:

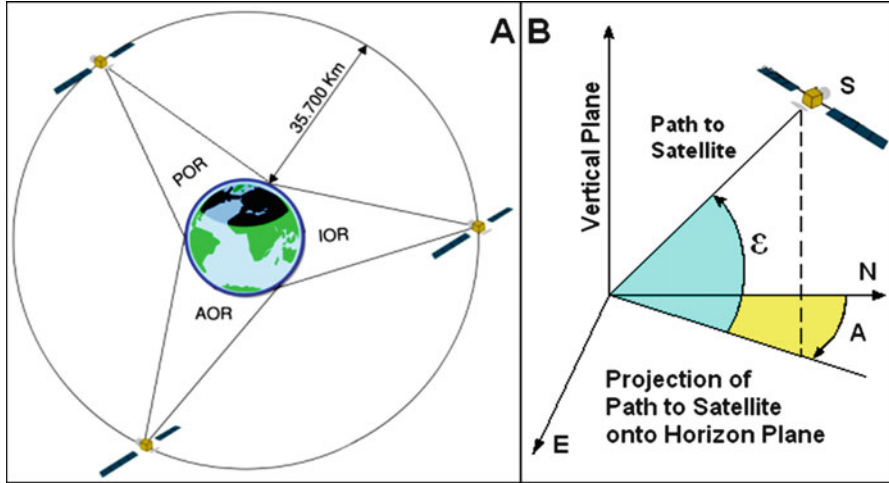


Fig. 2.4 GEO coverage and look angle parameters (Courtesy of Book: by Pratt)

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

All GES 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 apart in the orbit by  $120^{\circ}$  is possible to cover a great area of its surface.

In Fig. 2.4 (a) are illustrated 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  $5^{\circ}$  and  $30^{\circ}$  will be calculated with:

$$\Psi_s = \arccos (k \cos \varepsilon) - \varepsilon \tag{2.27}$$

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

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

The real altitude of satellite over sub-satellite point is as follows:

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

The view angle under which an GEO satellite can see GES/MES is called the “sub-satellite 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 sub-satellite’s 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:

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

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

$$\begin{aligned} \tan \delta &= k \sin \Psi / 1 - k \cos \Psi = 0.15126956 \sin \Psi / 1 - \cos \Psi / 1 - 0.15126956 \cos \Psi \\ \delta_s &= 90^\circ - \Psi_s = 8^\circ 42' 1.82'' \end{aligned} \quad (2.31)$$

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 GES with regard to the satellite can be calculated using Fig. 2.3 (a) 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 GES, which can be calculated by the following equation:

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

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

### 2.1.4.2 Satellite Look Angles (Elevation and Azimuth)

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

The satellite elevation ( $\epsilon$ ) 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. 2.3 (a) the look angle of  $\epsilon$  value can be calculated by the following relation:

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

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 considered. The azimuth value of the satellite and sub-satellite point looking from the point (M) or the hypothetical position of observer 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 sub-satellite point (P), can be calculated as:

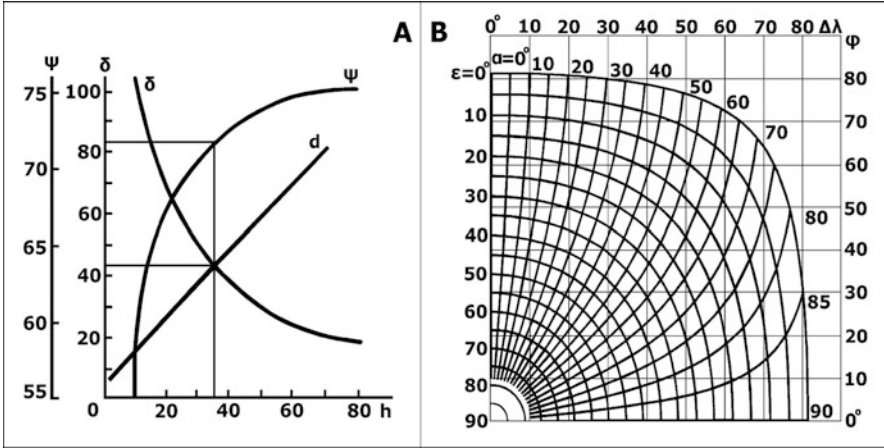
$$\operatorname{tg} A = \sin \Delta\lambda / \operatorname{tg} \varphi \text{ or } \sin A = \cos \varphi \sin \Delta\lambda \operatorname{cosec} \Psi \quad (2.36)$$

However, parameter ( $A'$ ) is the angle between the meridian plane of point (M) and the plane of a big circle crossing this point and sub-satellite 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.

In Fig. 2.5 (a) is presented a correlation of the look angle for three basic parameters ( $\delta$ ,  $\Psi$ , d) in relation to the altitude of the satellite. Inasmuch as the altitude of the satellite is increasing as the values of central angle ( $\Psi$ ), 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 sub-satellite angle or declination ( $\delta$ ) indirectly proportional. An important increase of look angle and duration of communication can 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 centre of look angle, which will have maximum value in the case when the direction is passing across the zenith of the GES. 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:





**Fig. 2.5** Look angle parameters and graphic of geometric coordinates for GEO (Courtesy of Book: by Zhilin)

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

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 user ( $\varphi, \lambda$ ), 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. 2.5 (b) a graphic design is shown which can approximately determine limited zones of satellite visibility from the Earth (user) 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 ( $\varphi$ ) and ( $\Delta\lambda$ ) coordinates of mobile positions.

In such a way, 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 centre shows the geometric positions where the difference in azimuth has the constant value ( $a = 0$ ). This diagram can be used in accordance with Fig. 2.3 (b) in the following order:

1. First, it is necessary to note the longitude values of satellite ( $\lambda_S$ ) and mobile ( $\lambda_M$ ) and the latitude of the mobile ( $\varphi_M$ ), then calculate the difference in longitude ( $\Delta\lambda$ ) and plot the point into the graphic with both coordinates ( $\varphi_M$  &  $\Delta\lambda$ ).
2. The value of elevation angle ( $\epsilon$ ) can then be determined by a plotted point from the group of parallel concentric curves.

**Table 2.2** The form for calculation of Azimuth values

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

3. The difference value of azimuth ( $a$ ) can be determined by a plotted point from the group of fan-shaped curves starting from the centre.
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), then 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, FES or any MES. Let us say, if the position of GES 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 ( $x$ ).

### 2.1.4.3 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 to the MES results from a spherical coordinate system, whose centre is the middle of Earth, is illustrated in Fig. 2.3 (b). In this way, the satellite position in any time can be decided by the geographic coordinates, sub-satellite point and range of radius. Thus, the sub-satellite point is a determined position on the Earth's surface; above it is the satellite at its zenith.

More exactly, the longitude and latitude are geographic coordinates of the sub-satellite point, which can be calculated from the spherical triangle ( $B' \Gamma P$ ), using the following relations:

$$\sin \varphi = \sin (\Theta + \omega) \sin i \text{ and } \operatorname{tg} (\lambda_s - \Omega) = \operatorname{tg} (\Omega + \omega) \cos i \quad (2.39)$$

With the presented equation in previous relation it is possible to calculate the satellite path or trajectory of sub-satellite 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 ( $\varphi_M$ ) and longitude ( $\lambda_M$ ) of the point (M) on the Earth's surface presented in Fig. 2.3 (b), what can be the position of the user, taking into consideration the arc (MP) of the angle illustrated in Fig. 2.3 (a), 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:

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

These relations are useful for any point or area of coverage on the Earth's surface, then for a centre 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 satellites and to get a global coverage from  $75^\circ$  N to  $75^\circ$  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 satellite at distance (d) and the velocity of light ( $c = 3 \times 10^8$  m/s) require 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 effect 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 development of Doppler satellite tracking and determination systems.

### 2.1.5 Orientation in Space

As stated earlier, the position of the satellite is determining by the angle called “the true anomaly”, which can be counted positively in the direction of movement of the satellite from  $0^\circ$  to  $360^\circ$ , between the direction of the perigee and the direction of the satellite (S). In Eq. (2.9) is derived distance of the satellites ( $r$ ) from the centre of the Earth ( $r = R + h$ ) or radius of path;  $\Theta$  = true anomaly or  $\mathcal{O}$  = eccentric anomaly. The value of angle  $90^\circ$  for GEO satellite inside its range was also determined by Eq. (2.17) with main parameters such as: sub-satellite angle or declination ( $\delta$ ), angle of elevation ( $\epsilon$ ) and central angle ( $\Psi$ ).

In such a way, by calculating  $r$  and  $\Theta$  at any time  $t$  is possible to obtain position of orbital plane in the space. To provide this system requires the definition of a coordinate system.

The coordinate system of coordinates that is used in an inertial reference frame, normally in the special theory of relativity is called Earth-Centered Inertial (ECI) coordinate system. Here, the three space coordinates are usually Cartesian coordinates ( $X, Y, Z$ ), and the time coordinate is the time as measured by an observer at rest in the coordinate system. On the other hand, in astrometry an inertial coordinate system is a reference frame formed by assigning coordinates to specific observable objects, such as the positions and proper motions of stars in a fundamental catalogue.

For orientation in the space, the coordinate system must be an inertial coordinate system, that is, a nonaccelerating system in which Newton’s Laws of Motion are valid. Thus, a coordinate system fixed to the rotating Earth is not such a system. Here is adopted an astronomical coordinate system called the right ascension-declination coordinate system, shown in Fig. 2.6 (Left). In this system the axis  $X$  is aligned with the Earth’s spin axis. The axis  $X$  is pointed from the center of the Earth to the Sun at the moment of the vernal equinox, when the Sun is crossing the equatorial plane from the Southern to the Northern Hemisphere. The axis  $Y$  is a right-handed coordinate system. The declination of a point in space is its angular displacement measured northward from the equatorial plane, and the right ascension is the angular displacement, measured counterclockwise from the axis  $X$ , of the projection of the point in the equatorial plane, depicted in Fig. 2.6 (Right).

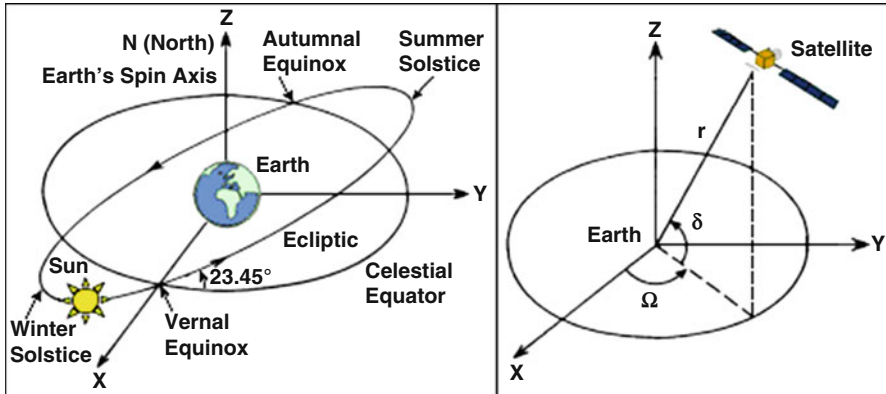
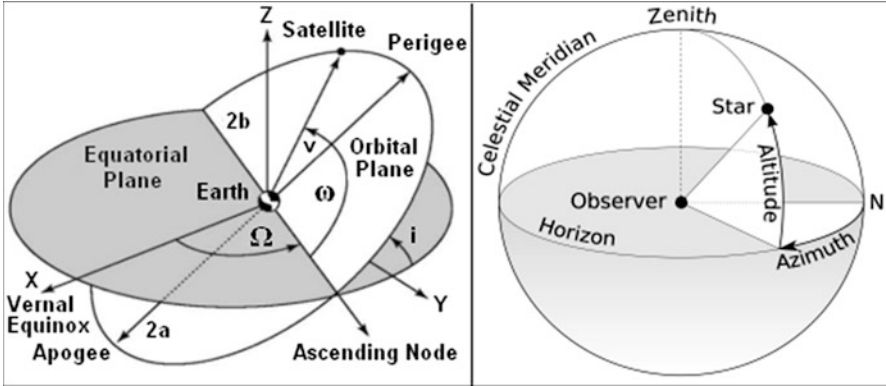


Fig. 2.6 Look angle parameters and coordinates (Courtesy of Book: by Kidder)

In Fig. 2.7 (Left) are shown three angles used to position an elliptical orbit in the right ascension declination (celestial) coordinate system between orbital and equatorial planes, inclination angle ( $i$ ), right ascension of ascending node ( $\Omega$ ) and argument of perigee ( $\omega$ ).

1. **Inclination Angle** – It is the angle between the equatorial plane and the orbital plane. By convention, the inclination angle is 0 if the orbital plane coincides with the equatorial plane and if the satellite rotates in the same direction as the Earth. If the two planes coincide then the satellite rotates opposite to the direction of Earth and the inclination angle is  $180^\circ$ . In addition, prograde orbits are those with inclination angles less than  $90^\circ$ , while retrograde orbits are those with inclination angle greater than  $90^\circ$ .
2. **Ascending Node** – It is the point where the satellite crosses the equatorial plane going North (ascends). The right ascension of this point is the right ascension of ascending node, which is measured in the equatorial plane from the axis Z (vernal equinox) to the ascending node. In practice, the right ascension of ascending node has a more general meaning. It is the right ascension of the intersection of the orbital plane with the equatorial plane; thus it is always defined, not just when the satellite is at an ascending node.
3. **Argument of Perigee** – It is angle measured in the orbital plane, for Earth-centered orbits, between the ascending node (equatorial plane) and the perigee. For other specific types of orbits, words such as perihelion is for Sun-centered orbits, periastron is for orbits around stars and so on may replace the word periapsis. The argument of periapsis (also called perifocus or pericenter) is one of the orbital elements of an orbiting body. The angle from ascending node to its periapsis, measured in the direction of motion, is parametrical.

In Fig. 2.7 (Right) is depicted an example of the star position in the sky with observer in the coordination centre. The star is the point of interest, while the reference plane is the horizon or the surface of the sea and the reference vector



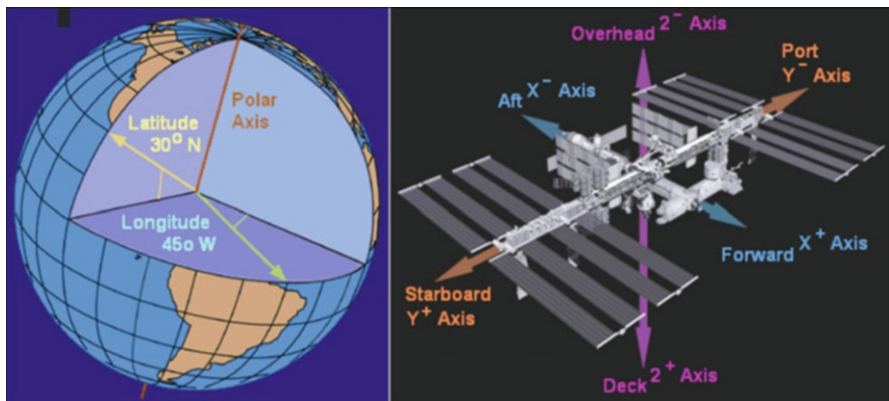
**Fig. 2.7** Right ascension declination and star's position (Courtesy of Manual: by Ilcev)

points North (N). The azimuth is the angle between the North vector and the perpendicular projection of the star down onto the horizon. Thus, similar to the satellite platform, this is the direction of a celestial object, measured clockwise around the observer's horizon from North. In such a way, an object due North has an azimuth of  $0^\circ$ , one due East  $90^\circ$ , South  $180^\circ$  and West  $270^\circ$ . Altitude, sometimes referred to as elevation, is the angle between the star or other astronomical object and the observer's local horizon, while for visible objects it is an angle between  $0^\circ$  and  $90^\circ$ . Azimuth and altitude are usually used together to give the direction of an object in the topocentric coordinate system.

The system oriented by the spin axis of the Earth has special points at the North and South Poles, which uses lines of latitude and longitude as geocentric local coordinate system to pinpoint a location of Earth's surface or to demarcate the surface, which is illustrated in Fig. 2.8 (Left). In such a way, latitude is measured away from the equator, while the starting point for longitude is Greenwich meridian.

As stated earlier, here are used two orthogonal global coordinates to describe a position, such as declination, like a celestial latitude and right ascension, like a celestial longitude. As with latitude, declination is measured away from the celestial equator, while with longitude, right ascension has starting point known as "vernal equinox", which is shown in Fig. 2.7 (Left). This point at which the Sun appears to cross the celestial equator from South to North as it moves through the sky during the course of a year.

To understand better positioning of objects in the space will be necessary to describe the basic nomenclature of satellite orientation in the space, which is illustrated in Fig. 2.8 (Right) using Reference guide to the International Space Station (ISS). Alternatively, movement relative to these three axes could be described using shipping terms roll, pitch and yaw, to describe attitude of a satellite, but these don't really denote the sides, merely rotation of the body with respect to the X, Y and Z (depicted in this case as red, blue and green) axes in Cartesian coordinate system, respectively. A Cartesian coordinate system is a coordinate



**Fig. 2.8** Coordinates of the earth and ISS (Courtesy of Manual: by Pidwirny/NASA)

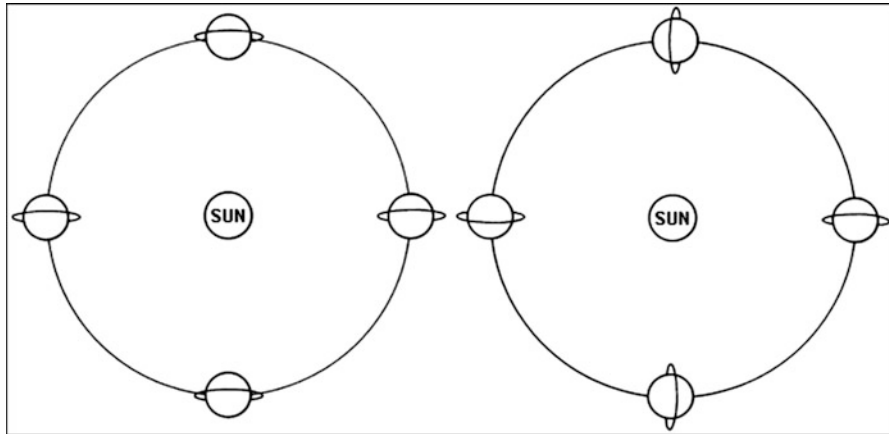
system that specifies each point uniquely in a plane by a pair of numerical coordinates, which are the signed distances to the point from two fixed perpendicular directed lines, measured in the same unit of length.

There might be other terms referring to the sides from the perspective of the spaceships and relative to its movement, such as nadir, which is direction directly below (opposite zenith), and zenith, which is directly above (opposite nadir). With regards to the maritime terms related to the vessels sides, there are additional sides, such as port, which is direction to the left side (opposite starboard) and starboard, which is direction to the right side (opposite port); And the remaining two sides mentioned in the figure, such as: ram (forward) or wake (aft) pointing, which also have maritime origin. NASA's Guide to the International Space Station Laboratory Racks Interactive however names direction towards nadir as deck and direction towards zenith overhead, and alternatively also the  $+/-$  axial values that follow the right-hand rule more commonly used by astronaut pilots during navigation or to describe station's attitude, such as during the docking.

### 2.1.6 Satellite Orbit Perturbations

After launching a satellite in circular orbits undergoes uniform angular velocity. By Kepler's Second Law, however, satellite in an elliptical orbit cannot have uniform angular velocity, so it must travel faster when is closer to Earth. The position of the satellite as a function of time can be found by applying Kepler's equation:

$$M_a = n(t - t_p) = e - \mathfrak{I} \sin e, \text{ where } n = 2\pi/T = \sqrt{Gm_e/a^3} \quad (2.46)$$



**Fig. 2.9** Seasonal changes for Keplerian and sunsynchronous orbit (Courtesy of Book: by Kidder)

where values  $n$  = mean motion constant,  $t$  = time,  $t_p$  = time in perigee passage,  $M_a$  = means anomaly,  $\mathfrak{E}$  = eccentric anomaly,  $e$  = eccentricity and  $m_e$  = mass of the Earth.

In the other words, satellite orbits in which the classical orbital elements (except  $M_a$ ) are constant are called Keplerian orbits. More exactly, viewed from space, Keplerian orbits are simple. The satellite moves in an elliptical path with the center of the Earth at one focus. The ellipse maintains a constant size, shape and orientation with respect to the stars, which seasonal changes are illustrated in Fig. 2.9, for Keplerian (Left) and Sunsynchronous Orbit (Right). Perhaps surprisingly, but the only effect of the Sun's gravity on the satellite orbit is to move the focus of the ellipse (the Earth) in an elliptical path around the Sun (the Earth's orbit). Viewed from the Earth, Keplerian orbits appear complicated because the Earth rotates on its axis as the satellite orbits the Earth, which orbit of a representative satellite as viewed from a point rotating with the Earth is shown in Fig. 2.10.

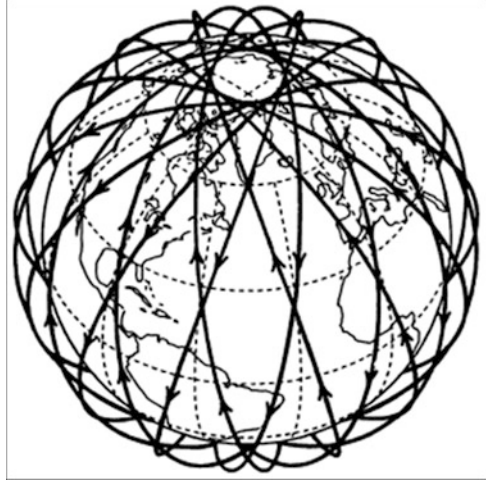
The rotation of the Earth beneath a fixed orbit results in two daily passes of the satellite near a point on the Earth (assuming that the period is substantially less than a day and that the inclination angle is greater than the latitude of the point). Thus, one pass occurs during the ascending portion of the orbit; the other occurs during the descending portion of the orbit. This usually means that one pass occurs during daylight and one during darkness.

There is some variation in how the orbital elements are specified. For example, ESA system substitutes true anomaly for mean anomaly. Also, in less formal descriptions of satellite orbits, one frequently sees the height of the satellite above the Earth's surface substituted for the semimajor axis. Since the Earth is not round, the height of a satellite will vary according to its position in the orbit. Specifying the semimajor axis is a much better way to describe a satellite orbit.

The eccentric anomaly is useful to compute the position of a point moving in a Keplerian orbit. As for instance, if the body passes the periastron at coordinates



**Fig. 2.10** Representative satellite orbit (Courtesy of Book: by Kidder)



$x = a(1 - e)$ ,  $y = 0$ , at time  $t = t_0$ , then to find out the position of the body at any time, it will be at first to calculate the mean anomaly from the time and the mean motion  $n$  by the formula  $M_a = n(t - t_0)$ , then solve the Kepler equation above to get  $\Theta$ , then get the coordinates from:

$$x = a(\cos \Theta - e) \text{ and } y = b \sin \Theta \quad (2.47)$$

In reality, Kepler's equation is a transcendental equation because sine is a transcendental function, meaning it cannot be solved by  $\Theta$  algebraically. More exactly, numerical analysis and series expansions are generally required to evaluate  $\Theta$ .

Although satellites travel in nearly Keplerian orbits, these orbits are perturbed by a variety of forces, see Table 2.3.

Forces arising from the last five listed processes are small and can be viewed as causing essentially random perturbations in the orbital elements. Operationally they are dealt with simply by periodically observing the orbital elements and adjusting the orbit with on-board thrusters. Forces due to the nonspherical Earth cause secular (linear with time) changes in the orbital elements.

These forces can be predicted theoretically and indeed are useful. The gravitational potential of Earth is a complicated function of the Earth's shape, the distribution of land and ocean, and even the density of crystal material. As a first-order correction to a spherical shape, we may treat the Earth as an oblate spheroid of revolution. In cross section the Earth is approximately elliptical.

The distance from the center of the Earth to the equator is, on average, 6378.140 km, whereas the distance to the poles is 6356.755 km. Thus, one can think of the Earth as a sphere with a 21 km thick "belt" around the equator. Therefore, the gravitational potential of the Earth is approximately given by the following equation:

**Table 2.3** Forces and sources of orbital perturbations

Force	Source
Nonspherical gravitational field	Nonspherical, nonhomogeneous Earth
Gravitational attraction of auxiliary bodies	Moon, planets
Radiation pressure	Sun's radiation
Particle flux	Solar wind
Lift and drags	Residual atmosphere
Electromagnetic forces	Interaction of electrical current in the satellite with Earth's magnetic field

$$U = -GM/r \left[ 1 + \frac{1}{2} J_2 (r_e/r)^2 (1 - 3 \sin^2 \mathcal{D}) + \dots \right] \quad (2.48)$$

where  $r_e$  = equatorial radius of the Earth,  $\mathcal{D}$  = declination angle and  $J_2$  = coefficient of the quadrupole term. The higher-order terms are more than two orders of magnitude smaller than the quadrupole term and will not be considered here, although they are necessary for very accurate calculations.

How does this belt of extra mass affect a satellite's orbit? Therefore, one might expect it to cause the satellite to orbit at a different speed, and indeed it does. The time rate of change of the mean anomaly ( $dM_a/dt$ ) is given by the mean motion constant  $n$  in the unperturbed orbit and by the anomalistic mean motion constant  $N$ , in a perturbed orbit. Considering only the quadrupole term anomaly can be formulated:

$$N = dM_a/dt = n \left[ 1 + 3/2 J_2 (r_e/a)^2 (1 - e^2)^{-3/2} (1 - 3/2 \sin^2 i) \right] \quad (2.49)$$

where  $a$  = semimajor axis and  $i$  = inclination.

When the inclination angle is less than  $54.7^\circ$ ,  $N$  is greater than  $n$ , so the satellite orbits faster than it would in an unperturbed orbit. However, for larger inclinations, the satellite orbits more slowly than it otherwise would.

Because the belt exerts an equator ward force, one might also expect that it would have an effect on the inclination angle. In such a way, this force affects the right ascension of the ascending node rather than the inclination angle.

Just as the force of gravity causes a top to precess rather than to fall over, so the attraction of the belt causes the orbit to precess about the  $x$  axis rather than to change its inclination angle. The rate of change of the right ascension of ascending node ( $\Omega$ ) can be determined by the following relation:

$$d\Omega/dt = -N \left[ 3/2 J_2 (r_e/a)^2 (1 - e^2)^{-2} \cos i \right] \quad (2.50)$$

The final effect of the belt is to cause the argument of perigee ( $\omega$ ) to rotate or precess, which relation is presented as:

$$d\omega/dt = -N \left[ 3/2 J_2 (r_e/a)^2 (1 - e^2)^{-2} (2 - 5/2 \cos^2 i) \right] \quad (2.51)$$

The other three orbital elements,  $a$ ,  $e$  and  $i$ , undergo small, oscillatory changes that may be neglected. If SI Units are used, Eqs. (2.46), (2.49), (2.50) and (2.51) result respectively in values of  $n$ ,  $N$ ,  $d\Omega/dt$  and  $d\omega/dt$  whose units are radians per second. The anomalistic period of a perturbed orbit is simply presented as:

$$T_a = 2\pi/N \quad (2.52)$$

However, because  $M_a$  is measured from perigee, the anomalistic period is the time for the satellite to travel from perigee to moving perigee. Of more use is the synodic or nodal period ( $T_s$ ), which is the certain time for the satellite to travel from one ascending node to the next ascending node. An exact value of  $T_s$  must be calculated numerically; however, to very good approximation as follows:

$$T_s = 2\pi/[N + (d\omega/dt)] \quad (2.53)$$

In summary, therefore, the first-order effects of the nonspherical gravitational potential of the Earth consist of a slow, linear change in two of the classical orbital elements, than the right ascension of ascending node and the argument of perigee, and a small change in the mean motion constant.

## 2.2 Spacecraft Launching and Station-Keeping Techniques

The launch of the satellite and controlling support services are a very critical point in the creation of space communications, meteorological or other systems and the most expensive phase of the total mission cost. 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. There are multi-stage 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 is very expensive operation, 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 centre 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 in the orbit, namely by direct ascent and by Hohmann transfer ellipse.

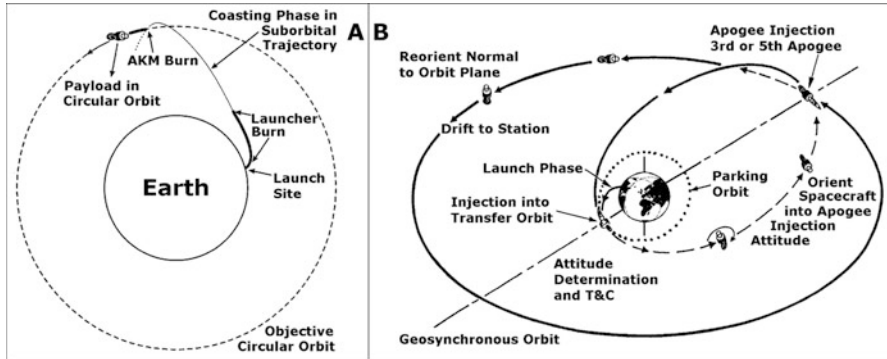
#### **2.2.1.1 Direct Ascent Launching**

A satellite may be launched into a circular orbit by using the direct ascent method, shown in Fig. 2.11 (a). 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 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. 2.11 (b). The Hohmann transfer ellipse method enables a satellite to be placed in an orbit at the desired altitude using the trajectory that



**Fig. 2.11** Satellite installation in circular and synchronous orbit (Courtesy of Book: by Pascall)

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 centres: Land-based and Sea-based launch systems.

#### 2.2.2.1 Expendable Launching Vehicles

The great majority of communication 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), H-II (Japan), Soyuz (Russia) and so on. In addition, a new generation of launchers have 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 proven Soyuz launch vehicle is one of the world's most reliable and frequently used launch vehicle. After the US Space Shuttle program ended in 2011, Soyuz rockets became the only launch vehicle able to transport astronauts to the International Space Station (ISS).

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 enters 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. 2.12 (a – Left) and (a – Right), respectively and also Russian Angara A-5 and Zenit (Ukraine). Otherwise, these types of vehicles do not need an AKM because direct-injection 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 lift-off the total weight of Proton is about 688 tons and this vehicle has the capability of placing a maximum of 4500 and 2600 kg into GTO and GEO, respectively. The Proton SL12/D1e has four stages and is the largest currently available space launcher from the former-USSR since 1967. The Proton SL13/D1 variant has three stages.

### 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

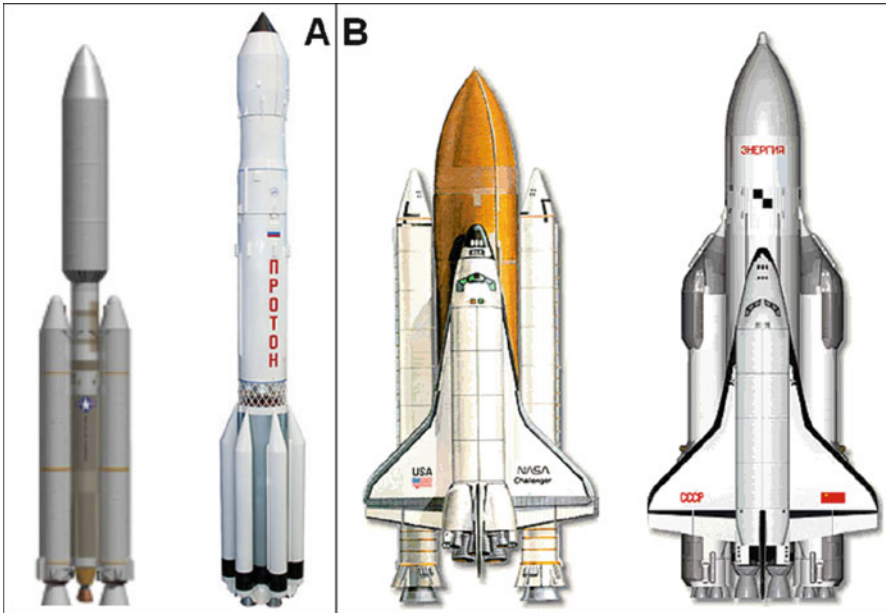


Fig. 2.12 Types of launch vehicles (Courtesy of Book: by Pascall)

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 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 development of these small vehicles it is important to realize whether any of them could carry sufficient weight and be able to put communication 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  $28.5^\circ$ , with an altitude of up to 431 km, which is illustrated in Fig. 2.12 (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 launches programme of government and scientific vehicles. The final shuttle mission was completed with the landing of Atlantis on 21 July 2011, closing the 30-year Space Shuttle program.

2. **Energia/Buran Spaceplane** – 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. 2.12 (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 communication satellites to be placed in orbit. Energia flew for the first time on 15 May 1987, carrying a spacecraft mock-up and later on 18 November 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 in 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 spaceplane Kliper by NPO





**Fig. 2.13** Russian spaceplane Kliper (Courtesy of Brochure: by Zak)

Energia, see Fig. 2.13. 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 Centres** – 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 Centres** – Russian satellites are launched from two main launch centres 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 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. The newest cosmodrome in Russia is Vostochny Cosmodrome "Eastern Spaceport". This cosmodrome is under construction on the 51° North in the Amur Oblast, in the Russian Far East. When completed in 2018, it is intended to reduce Russia's dependency on the Baikonur Cosmodrome in Kazakhstan. The first launch took place on 28 April 2016 at 0201 UTC.
3. **European Launch Centres** – The main European launch Cosmodrome is the Guiana Space Centre 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 Centres** – The principal launch sites in China are Jiuquan and Xi Chang, for launching Long March vehicles. In the meantime, the Xi Chang launch centre has also become most used for launches into the GEO for the international market.
5. **Japanese Launch Centres** – The Japan's Tanegashima Space centre 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 next generation of J-I Japanese vehicles. The new Yoshinobu launch complex has been constructed next to the Osaki centre 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 centres, has located at 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.

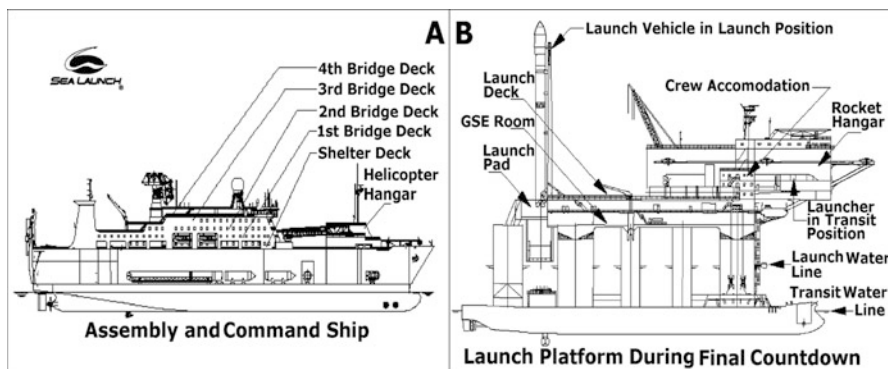
The partner team of contractors has developed an innovative approach to establishing Sea Launch as a reliable, cost-effective and flexible commercial launch system. Each partner is also a supplier to the venture, capitalizing on the strengths of these industry leaders. The System consists in two main modules: Assembly (Command and Control Ship) and Launch Platform, both illustrated in Fig. 2.14 (a, 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 pre-launch 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 Centre (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 infrastructure include: a rocket assembly compartment; the LCC with viewing room; helicopter capability; spacecraft contractor and customer work areas and spacecraft contractor and customer accommodation. Moreover, 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 fuelling 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.



**Fig. 2.14** Sea launch modules (Courtesy of Manual: by Sea Launch)

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 three 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 rocket 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 international 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 communications systems are distributed between the ACV and LP, which includes CCTV, telephones, intercom, videoconferencing, public address and vessel-to-vessel radiocommunications using Line-of-

Sight (LOS) direct system. It links external communication system and interconnects the various segments of the Sea Launch program. The external communication system includes Intelsat and two ground stations 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 the ACV and launch platform PABX systems to provide voice connectivity, while 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 is sent to a separate TDRSS receiver. Zenit data is received shortly after lift-off at approximately 9 s and continues until Zenit Stage 2/Block DM separation, at around 9 min. These data are routed from the NASA White Sands GES to the Sea Launch Brewster GES and to the ACV. The data are also recorded at White Sands and Brewster for later playback to the KB Yuzhnoye design centre. 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 is 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 onboard 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 centre station. 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 centre. 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 Meteorological and Other Satellite Systems

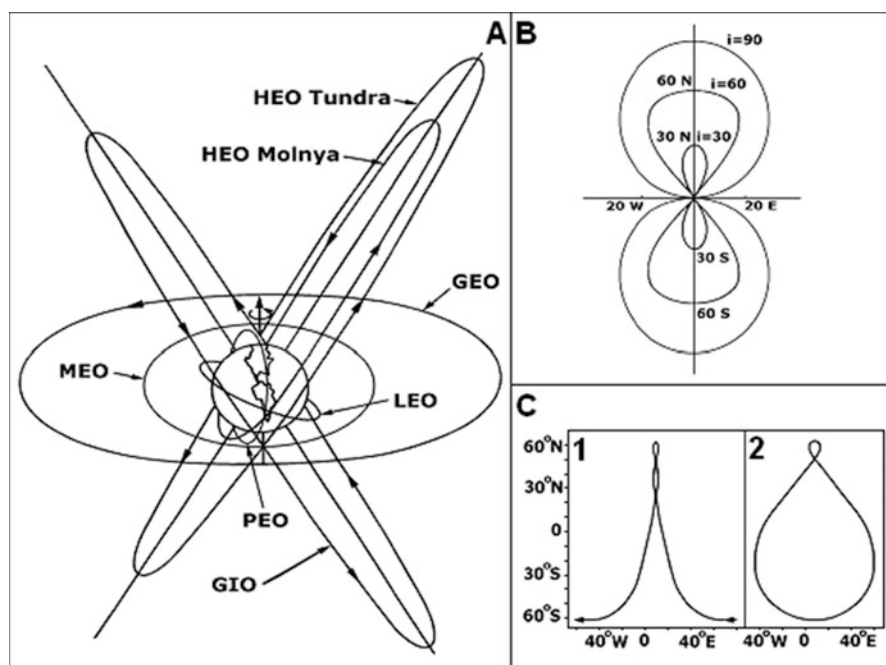
An orbit is the circular or elliptical path that the satellite traverses through space. This path appears in the chosen orbital plane in the same or different angle to the equatorial plane. All communication satellites always remain near the Earth and keep going around the same orbit, directed by centrifugal 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 meteorological or communication systems. The one most commonly used orbit for satellite meteorology are GEO and PEO (LEO) constellations. In addition, can be also used Highly Elliptical Orbit (HEO) and latterly Geosynchronous Inclined Orbits (GIO) and Medium Earth Orbit (MEO), depicted in Fig. 2.15 (a). Thus, 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 onboard.

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 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 track of the satellite varies from 0 to  $360^\circ$ , which is illustrated in Fig. 2.15 (b). The track of the GEO satellite is at a point in the centre of the coordinate system; two tracks are apparent movements of the GIO satellite with respect to the ascending node of both  $30^\circ$  and  $60^\circ$  inclination angles and the last is the track of the PEO satellite with an inclined orbit plane to the equator of  $90^\circ$ . The tracks of HEO Molniya (part of the track) and Tundra (complete track) orbits are depicted in Fig. 2.15 (c 1/2), respectively. These two tracks pass over the African Continent and almost all of Europe.

It is sufficient to see Table 2.4 and to understand that the major reasons for LEO problems are enormous 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.



**Fig. 2.15** Type of satellite orbits and tracks (Courtesy of Book: by Ilcev)

### 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 Earth surface and below the Inner Van Allen Belt. The orbit period at these altitudes varies between 90 min and 2 h, shown in Fig. 2.15 (a). The radius of the footprint of a communications satellite in LEO constellation varies from 3000 to 4000 km. 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. The traffic to a LEO satellite has to be handed over much more frequently than all other types of orbit. When an LEO satellite, which is serving particular users, 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. Satellites in LEO are not affected at all by radiation damage, but are affected by atmospheric drag, which causes the orbit to gradually deteriorate. For 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. Moreover, a perturbation is unlikely to produce a serious effect on the operation of a multi-satellite constellation since it will usually affect all satellites of the

**Table 2.4** The properties of four major orbits

Orbital properties	LEO/PEO	MEO	HEO	GEO
Development period	Long	Short	Medium	Long
Launch and 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	$\frac{1}{2}$ Sidereal day	1 Sidereal day
Inclination	90°	45°	63.4°	Zero
Coverage	Global	Global	Near global	Near global
Altitude range (km <sup>-3</sup> )	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

configuration in equal measure. It is obvious that satellites in LEO and MEO constellation are subject to orbital perturbation.

Today, the US NOAA SMAP and OCO-2 Series and Russian Kanopus-V 1–2 and Resurs-P 1–2 are operational LEO meteorological satellites, which will be introduced in Chap. 6.

### 2.3.2 Circular Orbits

The GEO together with and geosynchronous satellite constellation has great advantages for meteorological observations, and in the future MEO satellites will have significant success in these fields.

#### 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 to 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 is shown in Fig. 2.15 (a). Compared to a LEO system a MEO constellation can only be in circular orbit.



Doppler Effect and handover is less frequent and propagation delay is about 70 ms and free space loss is greater. These satellites are affected by radiation damages from the Inner Van Allen Belt only during the launching period and are subject to orbital perturbation.

Cosmic radiation in this orbit is lower, with subsequently longer life expectancy for the complete MEO configuration, fewer eclipse cycles means that battery lifetime will be more than 7 years, higher average elevation angle from users to satellite minimizes probability of LOS blockage and higher RF output power required for both indoor and handheld MSC terminals. 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 existing US GPS and Russian GLONASS. If Galileo at all became operational, it will use MEO configuration, which 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°.

However, MEO satellite configuration is successfully used as MEOSAR of Cospas-Sarsat system and can be effective for metrological observations as well. At present is developed O3b new MEO satellite constellation for fixed and mobile communications, which can include meteorological instruments and transponders as well in satellite payload.

### 2.3.2.2 Geostationary Earth Orbits (GEO)

A GEO has circular orbit in the equatorial plane, with orbital period equal to the rotation of the Earth of 1 sidereal day, which is achieved with the orbital radius of 66,107 (Equatorial) Earth Radii, or orbital height of 35,786 km, which is depicted in Fig. 2.15 (a). This satellite will appear fixed above the surface of the Earth and remain in a stationary position relative to the Earth itself. For instance, this orbit is with zero inclination and track as a point but in practice, the orbit has small non-zero values for inclination and eccentricity, causing the satellite to trace out a small figure eight in the sky, which track is illustrated in Fig. 2.15 (b). The footprint or service area of a GEO satellite covers almost 1/3 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 communication, metrological, GEOSAR Cospas-Sarsat system and other services with the following advantages:

- (a) Satellite remains stationary with respect to one point on the Earth's surface and so the GES satellite antennas can be beamed exactly towards the focus of the GEO satellite without periodical tracking.

- (b) Inmarsat GEO satellite constellation consisting in three or four satellites can cover all three-ocean regions with overlapping longitudes, except for the Polar Regions beyond latitudes of  $75^\circ$  North and South.
- (c) 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 GES within satellite coverage.

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

- (a) Long signal delay is due to the large distance of about 35,800 km if the satellite is in zenith for GES and about 41,000 km at the minimum elevation angle of about  $5^\circ$ . 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.
- (b) Required higher RF output power and the use of directional satellite antennas.
- (c) 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.

An 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. Using satellite spot beam antennas GEO coverage can be confined to smaller spot areas with bigger power and higher speed of transmission. 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. The GEO satellite constellation seems likely to continue to dominate in the meteorological satellite observation world, including satellite communication and broadcasting systems, providing near global coverage with low and high-power transmission.

### 2.3.2.3 Geosynchronous Inclined Orbits (GIO)

The GIO constellations would consist in four satellites at 6-h intervals around the Earth an inclination of  $45^\circ$  to the equatorial plane, which is shown in Fig. 2.15 (a). This orbit provides polar coverage for 6 h either side of their most Northerly and Southerly movement and needs special GES with full tracking antennas. The 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 in such a way, this movement also has constant angular velocity.

Thus, the projection of this satellite 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. In Fig. 2.15 (b) is presented tracks of  $30^\circ$  and  $60^\circ$  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. Second, usually there is no need to control inclination because tracking GES antennas are required for other reasons. Some GEO satellites may last beyond their planned lifetime if run low on fuel and cease inclination control. In effect, the GIO constellation with non-zero 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 continuous service. At this point, blocking of the beam due to occlusion of the satellite by buildings, mountains, hills and trees is minimized. Besides, multiple trajectories caused by successive reflection of various obstacles are also reduced in comparison with systems operating with low elevation angles, like GEO.

The apogee altitude combines polar coverages with nearly synchronous advantages. Thus, minimum two special GES in both northern and southern Polar Regions are required to serve MES terminals. The GES 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 towards 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 GES 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 GES, 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  $63.4^\circ$  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 centre to the apogee and the Earth's surface will always occur at latitude of  $63.4^\circ\text{N}$ . 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 8 h at a minimum.

When there is an orbit in HEO plane of non-zero 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  $90^\circ$ . By orienting the apsidal line, namely the line between perigee and apogee, in the vicinity of the perpendicular to the line of nodes (when  $\omega$  is close to  $90^\circ$  or  $270^\circ$ ), 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 GES, FES or MES terminals 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  $2\pi$ ) 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. 2.15 (a). The ESA proposed Archimedes system employs a so-called “M-HEO” 8-h orbit. This produces three apogees spaced at  $120^\circ$ . 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  $63^\circ$  semi-synchronous orbit planes. With 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 8 h 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  $90^\circ$  apart, but the ascending node of one constellation is shifted  $90^\circ$  from the other, i.e., the Eastern Hemisphere ascending nodes are approximately  $65^\circ$  and  $155^\circ$ E, respectively. Although the system was designed to support the Russian Orbita TV network, a principal function was to service Russian government and military communications traffic via a single 40 W 1.0/0.8 GHz satellite transponder.

**Table 2.5** 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 to 0.75	0.25 to 0.4
Perigee altitude ( $h_p$ )	$a(1 - e) - R$	$a(1 - e) - R$
(e.g.: $e = 0.71$ )	1250 km	25,231 km
Apogee altitude ( $h_a$ )	$a(1 + e) - R$	$a(1 + e) - R$
(e.g.: $e = 0.71$ )	39,105 km	46,340 km

The hypothetical Russian Molniya network can employ minimum 3 HEO satellites in three 12-h orbits separated by  $120^\circ$  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 2.5.

The only one-track cycles of a total of two satellite tracks on the surface of the Earth is shown in Fig. 2.15 (c-1) for a perigee argument equal to  $270^\circ$ . The shape of this track is cycles of one orbit only near Greenwich Meridian, so the centre, of the next identical track is around  $180^\circ$  westward. Therefore, the satellite at apogee passes successively on each orbit above two points separated by  $180^\circ$  in longitude. The apogee is situated above regions of  $63^\circ$  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  $270^\circ$ ). 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  $63.45^\circ$ . A value different from this leads to a drift, which is non-zero but remains small for value of inclination, which does not deviate too greatly from the nominal value. By way of example, for an inclination angle  $i = 65^\circ$ , that is variation of  $1.55^\circ$ , the drift of argument of the perigee has a value of around  $6.5^\circ$  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 GES must use tracking antenna systems, so a terminal with only one antenna will have an outage during handover (switching) from one satellite to another.

Using experience and efficiency of Molniya HEO satellites in 2009 Russia developed the first meteorological HEO satellite known as Arctica, which will be introduced in Chap. 6.

### 2.3.3.2 Tundra Orbit

The Russian Tundra HEO system employs 2 satellites in two 24-h orbits separated by  $180^\circ$  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 2.5. This orbit has only one track on the Earth's surface, as shown in Fig. 2.15 (c-2), for a perigee argument equal to  $270^\circ$ , 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 a 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 towards 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 towards the East or West, with respect to the point of maximum latitude, by changing the value of argument of the perigee ( $\omega$ ) and the eccentricity.

### 2.3.3.3 Loopus Orbit

The proposed Loopus system, which employs 3 satellites in three 8-h orbits separated by  $120^\circ$  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 GES 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.

### 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  $90^\circ$  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  $90^\circ$  to each other, the time between satellites passes over any given point will be halved, which orbit is shown in Fig. 2.15 (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 solutions 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. As stated, Cospas-Sarsat system has developed MEOSAR using MEO satellites, GEOSAR using three GEO satellites for global distress communications satellite beacons in combination with first 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. 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 2 h 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 handover and minimize waiting time between transits. This problem can be solved with additional GES terminals over pole area.

In Fig. 2.16 is illustrated the Earth track of ten successive orbits of satellite in Low PEO with an altitude of 1000 km. The GES 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 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 every day. 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 as 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.

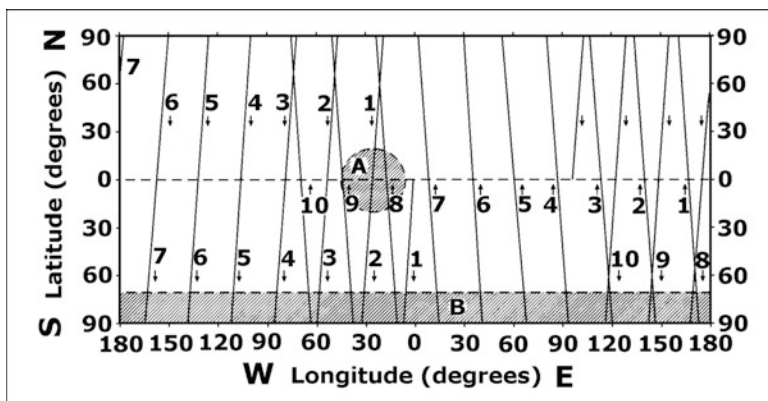


Fig. 2.16 Type of satellite orbits and tracks (Courtesy of Book: by Pascall)

## 2.4 Main Characteristics of Metrological Satellite Orbits

Nearly all-present meteorological satellites are using one of two satellite orbits, such as sunsynchronous (PEO) or geostationary (GEO), but other orbits are also useful, such as LEO and HEO.

### 2.4.1 *Sunsynchronous Polar Orbits*

The nonspherical gravitational perturbation of Earth, far from being a problem, has a very useful application. As shown in Fig. 2.9 (Left), the angle between the lines that join the Sun and the ascending node to the center of the Earth changes in a Keplerian orbit because the orbital plane is fixed while the Earth rotates around the Sun. This causes the satellite to pass over an area at different times of the day. For example, if the satellite passes over near noon and midnight in the spring, it will pass over near 6:00 am and 6:00 pm in the winter. In a such a manner, several additional problems result, among them are: (1) the data do not fit conveniently into operational schedules, (2) orientation of solar cell panels is difficult and (3) dawn or dusk visible images may not be as useful as images made at other times. Fortunately, the perturbations caused by the nonspherical Earth can be employed to keep the Sun-Earth-satellite angle constant.

The Earth makes one complete revolution about the sun ( $2\pi$  radians) in one tropical year (31,556,925.9747 s). Thus, the right ascension of the sun changes at the average rate of  $1.991064 \times 10^{-7} \text{ rad s}^{-1}$  ( $0.9856473^\circ \text{ day}^{-1}$ ). If the inclination of the satellite is correctly chosen, the right ascension of its ascending node can be made to precess at this same rate. At this point, a satellite orbit that is so synchronized with the Sun is called a sunsynchronous orbit. For a satellite with a semimajor axis of 7228 km and zero eccentricity, Eq. (3.53) requires an inclination of  $98.8^\circ$  to

be sunsynchronous. In Fig. 2.9 (Right) is shown the change with season of a sunsynchronous orbit.

Because the sun-Earth-ascending node angle is constant in a sunsynchronous satellite orbit, the satellite is often said to cross the equator at the same local time every day. Unfortunately, local time is an ambiguous term. This can be used to express equivalence as follows:

$$LT \equiv t_U + \lambda/15^0 \quad (2.54)$$

where  $t_U$  = coordinated universal time in hours and  $\lambda$  = longitude in degrees of a particular point. Equator Crossing Time (ECT) is the local time when a satellite crosses the equator:

$$ECT \equiv t_U + \lambda_N/15^0 \quad (2.55)$$

where  $\lambda_N$  = longitude of ascending or descending node. Therefore, if  $\lambda_N$  is constant, as it is for a sunsynchronous satellite, then ECT is constant.

Sunsynchronous satellites are classified by their ECT values. Thus, noon satellites (or noon-midnight satellites) ascend (or descend) near noon LT (local time). They must, therefore, descend (or ascend) near local midnight. Morning satellites ascend (or descend) between 06 and 12 h LT, and descend (or ascend) between 18 and 24 h LT. Afternoon satellites ascend (or descend) between 12 and 18 h LT, and descend (or ascend) between 00 and 06 h LT.

The highest latitude reached by the subsatellite point (in any orbit) is equal to the inclination angle (or the supplement of  $i$ , in the case of retrograde orbits). Since sunsynchronous orbits reach high latitudes, they are referred to as near polar orbits. This is frequently shortened to polar orbits, although they do not cross directly over the poles. These orbits are also called LEO to distinguish them from geostationary orbits (GEO). Note, however, that PEO is a general term for a satellite that passes near the poles, and LEO is a general term for a satellite that orbits not far above the Earth's surface.

While sunsynchronous satellites are of necessity polar orbiters and LEO, the converse is not necessarily true. The ground track of a satellite is the path of the point on the Earth's surface that is directly between the satellite and the center of the Earth (the subsatellite point). Thus, in Fig. 2.17 is illustrated the ground track for three orbits of the typical sunsynchronous NOAA 11 satellite, which main parameters are presenting the following values:  $a = 7229.606$  km,  $i = 98.97446^0$ ,  $e = 0.00110058$ ,  $M_a = 192.28166^0$ ,  $\Omega = 29.31059^0$ ,  $\omega_0 = 167.747540$ , Epoch Time = 22 March 1990 1 h 15 min and 55.353 s UTC and Nodal Period = 102.0764 min.



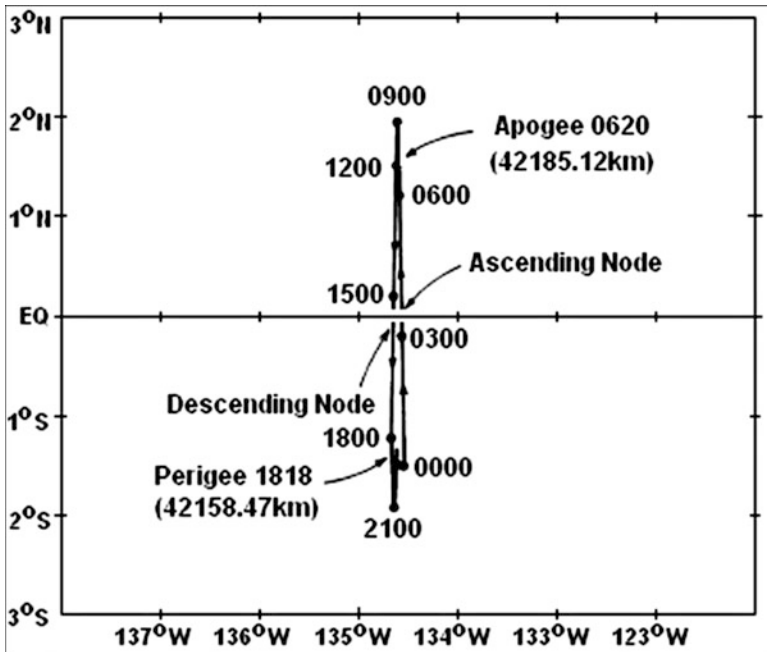


Fig. 2.18 Ground track of geostationary satellite (Courtesy of Book: by Kidder)

Thus, the principal GEO parameters are presenting by the following values:  $a = 42171.798 \text{ km}$ ,  $i = 1.97310^\circ$ ,  $e = 0.0003160$ ,  $M_a = 147.020^\circ$ ,  $\Omega = 80.259^\circ$ ,  $\omega_0 = 223.89100$ , Epoch Time = 6 March 1990 4 h 8 min and 20 s UTC and Nodal Period = 1431.297 min.

As stated above and in the other words, the satellite ground track is an imaginary line on the Earth's surface that traces the course of another imaginary line between Earth's centre (O) and an orbiting satellite (S).

In this sense, near-circular satellite orbits, which may be considered as circular in a first approximation, constitute a very important and frequently encountered case. Let now to study some notions developed specifically for these orbits, such as the equatorial shift or the apparent inclination. In this sense, when the orbit is circular, the motion of satellite is uniform with angular frequency ( $n$ ), which is discussed earlier as mean motion.

The three Euler angles  $\theta$ ,  $\Phi$  and  $\psi$  are serving to specify the orbit and its perigee in space. In this case is necessary to specify satellite (S). Obtaining the correspondence between the Euler angles and the orbital elements is giving as follows:  $\theta = \Omega$ ,  $\Phi = i$  and  $\psi = \omega + \Xi$ .

Although they are fixed for the Keplerian orbit, the ascending node angle ( $\Omega$ ); argument of the perigee ( $\omega$ ) and  $M - nt$  vary in time for a real orbit, however, the inclination  $i$  remain constant. The distance from S to the centre of attraction O or distance of the satellites from the centre of the Earth ( $r = R + h$ ), known also as

radius of path, is given by expressed in terms of semi-major axis ( $a$ ), true anomaly ( $\Theta$ ) and eccentricity ( $e$ ) as:

$$r = a(1 - e^2) / (1 + e \cos \Theta) \quad (2.56)$$

Using the notation introduced above, stated value of  $\psi = \omega + \Theta$  can be replaced by:

$$\psi = n(t - t_{AN}) \quad (2.57)$$

Therefore, the ground track of the satellite is determined by two quantities relating to the ascending node taken as origin, namely, its longitude  $\lambda_0$  and the crossing time  $t_{AN}$ , which constitute the initial conditions of the uniform motion.

### 2.4.3 Other Satellite Orbits

Geostationary and sunsynchronous are only two of infinite possible orbits. Others have been and will become useful for meteorological satellites. The US LEO Earth Radiation Budget Satellite (ERBS) was launched in 1984 by Space Shuttle in orbit at an altitude of 600 km with an inclination angle of  $57^\circ$ . It was placed in this orbit so that it would precess with respect to the Sun and sample all local times over the course of a month.

The former Soviet Union placed its Meteor satellites in low Earth orbit with inclination angles of about  $82^\circ$ . The former Soviet Union also used a highly elliptical orbit for Molniya communications satellites. Thus, it has been suggested that this orbit would be useful for meteorological observations of the high latitudes (Kidder and Vonder Haar 1990). This satellite has an inclination angle of  $63.4^\circ$ , at which the argument of perigee is motionless thus the apogee, from which measurements are made, stays at a given latitude.

The semimajor axis is chosen such that the satellite makes two orbits while the Earth turns once with respect to the plane of the orbit. The eccentricity is made as large as possible so that the satellite will stay near apogee longer. The eccentricity must not be so large that the satellite encounters significant atmospheric drag at perigee. A semimajor axis of 26,554 km and an eccentricity of 0.72 result in a perigee of 7378 km (1000 km above the equator), an apogee of 45,730 km (39,352 km above the equator), and a period of 717.8 min. The attractiveness of this orbit is that it functions as a high-latitude, part-time, nearly GEO satellite. For about 8 h centered on apogee, the satellite is synchronized with the Earth and is nearly stationary in the sky. For a meteorological satellite in a Molniya orbit, the rapid imaging capability, useful from GEO, would be available in the high latitudes as well.

In such a way, as meteorological satellite and instruments become more specialized, more custom orbits are likely to be used in the future, which include GEO, PEO, LEO and HEO.

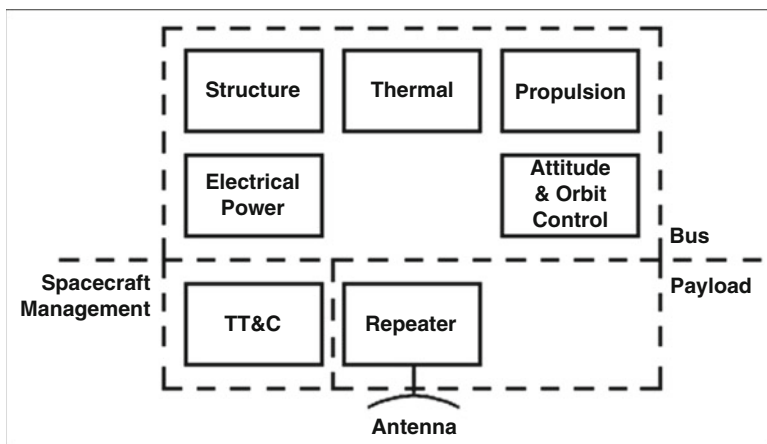
To provide global coverage of modern meteorological satellite observation system can be deployed special satellite constellation known as Hybrid Satellite Orbits (HSO) such as:

1. **Combination of GEO and HEO Constellations** – In development this kind of HSO can be included combination of minimum three GEO and four HEO satellites;
2. **Combination of GEO and PEO Constellations** – This HSO combination can include minimum 3 GEO and four PEO satellites;
3. **Combination of GEO and LEO Constellations** – This HSO combination needs minimum three GEO and two LEO satellites in Sun synchronous orbit;
4. **Combination of MEO and HEO Constellations** – This HSO combination will need minimum eight MEO and four HEO satellites; and
5. **Combination of MEO and LEO Constellations** – This HSO combination needs minimum three GEO and minimum two LEO satellites in Sun synchronous orbit.

## 2.5 Meteorological Satellite Payloads and Antenna Systems

An onboard communication element of meteorological satellite consists many in two major functional units: payload and bus. The primary function of the payload is to provide communication between ground sensors known as Data Collection Platform (DCP) repeaters and ground segment of Direct Readout Ground (DRG) Earth Stations. However the main function of satellite payload is to transmit collected meteorological data from different onboard sensors to DRG. While the bus provides all the necessary electrical and mechanical support to the payload and all satellite missions illustrated in Fig. 2.19.

The payload is made up of the multipurpose repeaters and antenna systems. The repeater performs the required processing of the signal and the antenna receives signals from GES and to transmit signals to FES or MES in the coverage area and vice versa. Two main types of satellite repeaters are possible for onboard utilization: Transparent and Regenerative transponders however are developed many other types for different satellite applications.



**Fig. 2.19** Spacecraft sub-system (Courtesy of Book: by Richharia)

### ***2.5.1 Transparent or Bent-pipe Communication Transponder***

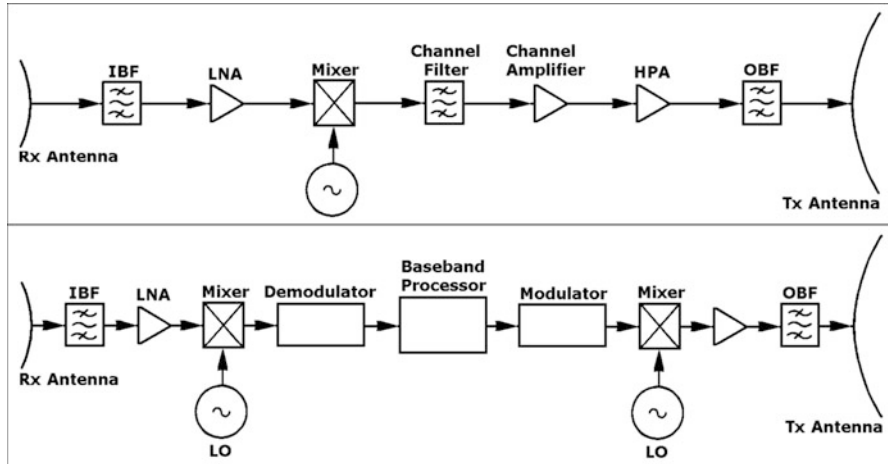
The basic function of the satellite transponder is to isolate individual carriers or groups of carriers of signals and to boost their power level before they are retransmitted to the ground stations. The carrier frequencies are also altered as the carriers pass through the satellite. Satellite repeaters that process the carrier in this way are typically referred to as transparent or bent-pipe transponders, which are illustrated in Fig. 2.20 (Above). 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 amplification. Initially, the received uplink signals from ground-satellite-ground by Receiver (Rx) antennas 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 from the down conversion, 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 Transmitter (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 to a lower





**Fig. 2.20** Configuration of spacecraft transponders (Courtesy of Handbook: by ITU)

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 (TWT). In a more complex transponder design, in order to achieve higher RF power, it may be possible to combine the output of several amplifiers.

### 2.5.2 Regenerative Communication Transponder

Other satellite system designs go through a more complex onboard process to manipulate the carrier's formats, by using on-board processing architecture. This payload architecture offers advantages over the transparent alternative, including improved transmission quality and the prospect of compact and inexpensive ground terminals.

A typical on-board processing system will implement some or all of the functions that are performed by the ground-based Tx and/or Rx 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, which is illustrated in Fig. 2.20

**(Below).** This type of satellite transponder provides demodulation and modulation capacity completely on board the satellite.

The received uplink signal goes along the down-converter segment prior to coming into the on-board 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 on-board modulator passes along the up-converter 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 downlink by the on-board demodulator and modulator, supported by the base band processor.

A regenerative transponder with base band processing permits reformatting of data without limitation to ground 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 ground Rx G/T and demodulation performance.

### ***2.5.3 Satellite Meteorological e Communication Transponder***

The meteorological 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 ground stations and to transmit signals to the ground receiving and processing facilities in the coverage area and vice versa.

In Fig. 2.21 are illustrated main components of meteorological satellite payloads, which contain the following main components of instruments: Scan Motor Drive, Optical Sensors, Picture Element Buffer and additional radiometers.

On the other hand, meteorological satellite has communication payload, which contain two repeaters. First is Dissemination Repeater, which Receiver (Rx) and Transmitter (Tx) with both spacecraft antennas uses 2 GHz, while Rx and Tx with both antennas of onboard Data Collection Repeater uses 0.4 GHz.

The duties of the meteorological satellite payloads are to contain radio Rx, which acquire operational commands and meteorological satellite data from the ground and retransmit this data via Tx in Data Collection Repeater to the readout ground receiving and processing facilities. In such a way, second Tx in Dissemination Repeater can retransmit data from the ground and send data collected by the onboard instruments to Earth stations.

Then meteorological satellite may carry main components shown in Fig. 2.22, including computers that process the data from the onboard instruments and the tape recorders on which data are recorded for later transmission to Earth. Usually meteorological GEO satellites carry onboard several different payloads, not just one. The primary payloads are meteorological radiometer, which measures Earth

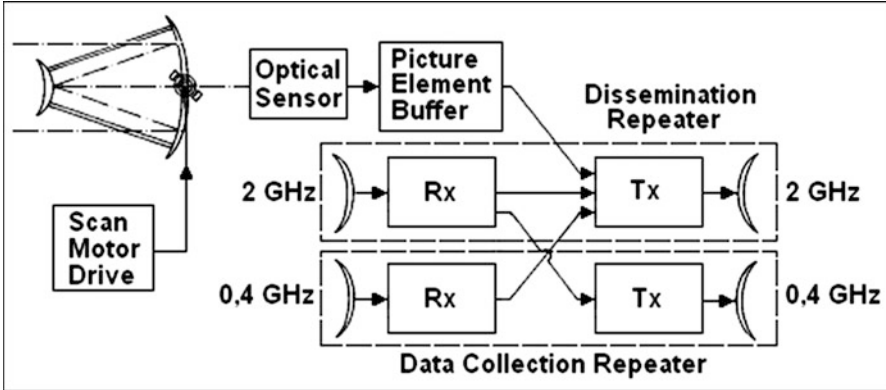


Fig. 2.21 Meteorological satellite payloads (Courtesy of Book: by Berlin)

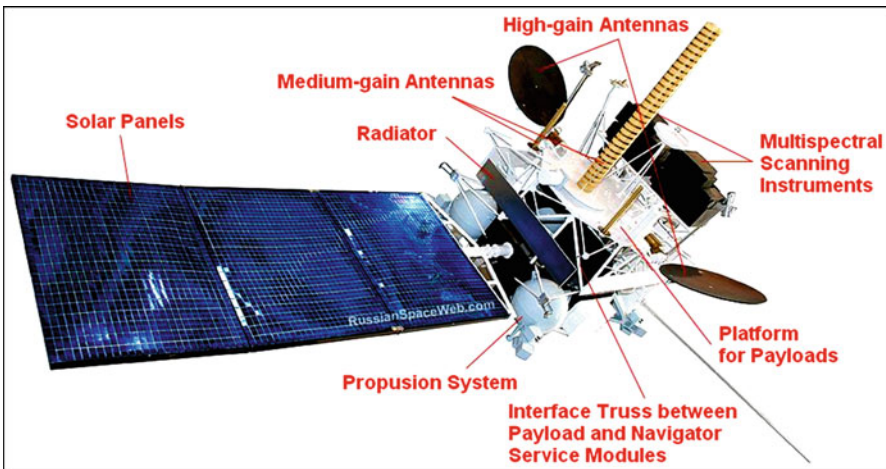


Fig. 2.22 Main components of Russian electro GEO satellite (Courtesy of Booklet: by Roscosmos)

radiances in the visible ( $0.4\text{--}1.0\ \mu\text{m}$ ) and thermal infrared ( $10\text{--}13\ \mu\text{m}$ ) spectral bands. Same satellites carry and additional payloads not related to meteorology.

For example, the Search and Rescue (SAR) payload of Cospas-Sarsat system detects signals from downed ships, land vehicles, persons and aircraft, giving the precise estimate of their location to aid in rescue operations. The Space Environment Monitor (SEM) measures energetic particles (protons, electrons and alpha particles) for solar and ionospheric studies. Finally, the Data Collection System (DCS) relays meteorological and other data transmitted from ground-based instruments.

### 2.5.4 *Diagram of VSAT GEO Satellite Communication Repeater*

As stated earlier, a transparent satellite payload makes no distinction between uplink carrier and uplink noise, and both signals are forwarded to the downlink facilities. On the other hand, however, at the Earth station receiver, one gets the downlink noise together with the uplink-retransmitted noise.

Moreover, a regenerative satellite payload entails onboard demodulation to process of the uplink carriers. Onboard satellite regeneration is most conveniently performed on digital carriers. In fact, the bit stream obtained from demodulation of a given satellite uplink carrier is then used to modulate a new carrier at downlink frequency. This signal carrier is noise-free; hence a regenerative satellite 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. 2.23. It should be emphasized that today's commercial and other type of satellites can be used for VSAT services, which are not equipped with regenerative payloads but only with transparent ones. In reality, 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. For instance, 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.

In the more determined sense, it is important to underline that DVB-RCS technique and technology is very important and useful for the future more reliable and successful high speed file transfer of meteorological data and images using VSAT transponders onboard satellites meteoroloskim. The DVB-RCS standards will improve data and image transmissions with the following

1. **DVB-S Standard** – This standard provides DVB adopted technique nearly in all parts of the world (except US and Japan) for transfer of any kind of digital data using Multiplex of MPEG-II video/audio and IP packets. The cost of VSAT and other hardware is very low for about 50 \$ available (PC card) and plugs into existing PC or laptop Dish, plus receiver front-end for another 50 \$.
2. **DVB-S2 Standard** – This DVB standards benefits from recent developments and improvements in transmission Voice, Data and Video (VDV) VSAT technology, powerful error-correction coding, fade-mitigation to overcome impairments of the channel due to unfavourable propagation conditions (rainstorms)

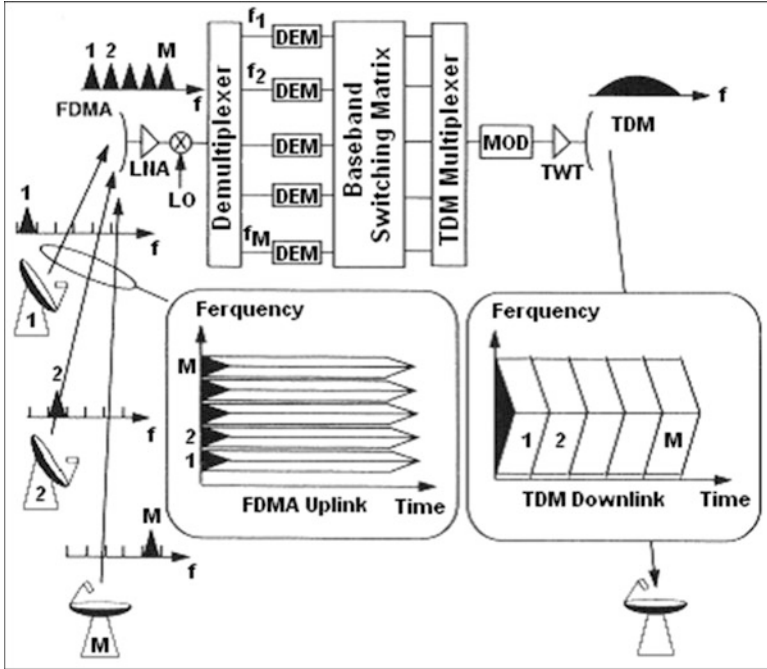


Fig. 2.23 Diagram of VSAT spacecraft transponders (Courtesy of Book: by Maral)

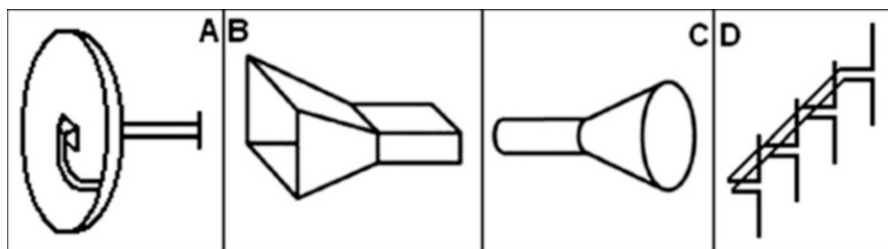
and provide typically 30–35% capacity increase over DVB-S under same transmission conditions.

### 2.5.5 Antenna System onboard Metrological Satellites

The antenna systems of onboard meteorological satellites can be many types and may have different shapes. The spacecraft antenna system mounted on the spacecraft structure similar is composed of two main integrated elements: the receiving antennas, which are detecting all signals from the ground stations, and transmitting antennas, which are sending data and video signals to the ground receiving and processing stations.

The transmit antenna systems are providing a global (wide) beam on the Earth's surface via GEO satellites, and local footprint via PEO satellites. 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  $17.4^\circ$ . Antenna beams  $5.8^\circ$  wide can reuse three frequency bands twice in providing Earth disc coverage.

For instance, from GEO the Earth subtends an angle of  $17.4^\circ$ . Thus, in this content will be explained four types only, such as follows:



**Fig. 2.24** Four types of spacecraft antennas (Courtesy of Book: by Ilcev)

### 2.5.5.1 Reflector Antennas

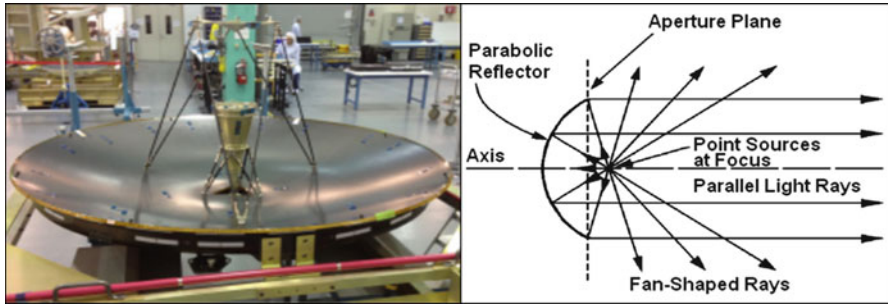
The parabolic reflector is a good example of reflectors at radio microwave frequencies, shown in Fig. 2.24 (a). In the past, these antenna were used mainly in space applications onboard spacecraft, but today they are very popular and used by almost everyone who wishes to receive the large number of television or broadcasting channels transmitted all over the globe providing gain in values of 20–30 dB. Reflector antennas are typically used when very high gain or narrow main beam is required. Gain is improved and the main beam narrowed with increase in the reflector size. Large reflectors are however difficult to simulate as they become very large in terms of wavelengths. Reflector antennas are usually illuminated by one or more horns and provide a larger aperture than can be achieved with a horn alone, which prototype made by the US company Lockheed Martin is shown in Fig. 2.25 (Left).

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, shown in Fig. 2.25 (Right).

The paraboloid, however, 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. At this point, the preferred way of making directional spacecraft antennas is to make a dish to reflect the waves rather than to refract them with a lens.

These dishes are easy to make and unlike a lens, which needs to be, solid and they need only a very thin reflective surface. When the reflector is given a specific shape called a parabola and the receiver is placed at the focus the waves received are parallel. This makes this antenna the preferred type for radio telescopes and other extreme range applications.

However, the reflector dish using focal point the waves are leaving the lens parallel to each other and therefore will go great distances with out scattering.



**Fig. 2.25** Parabolic reflector antenna systems (Courtesy of Manual: by Ilcev)

While this works it has problems mostly that is transmission the waves moving to the left are unaffordable.

### 2.5.5.2 Aperture Antennas (Horn Antennas)

A horn is an example of aperture antennas, which are used in satellite spacecraft more commonly, which pyramidal horn is shown in Fig. 2.24 (b), while conical horn is shown in Fig. 2.24 (c). Rectangular or pyramidal horn antenna is one of the simplest and most widely used antennas. Horns have been used for more than a hundred years, and today they used in satellite communications, radio astronomy in communication dishes as feeders, in measurements, etc.

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 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. In such a way, it is difficult to obtain gains much greater than 23 dB or beamwidths narrower than about  $10^\circ$  with horn antennas. In such a way, for higher gains or narrow beamwidths a reflector antenna or array must be used.

### 2.5.5.3 Array Antennas

A grouping of several similar or different antennas forms a single array antenna, shown in Fig. 2.24 (d). The control of phase shift from element to element is used to scan electronically the direction of radiation. Antenna arrays are able to produce radiation patterns that combined, have characteristics that a single antenna would not. The antenna elements can be arranged to form a 1 or 2 dimensional antenna array. A number of antenna array specific aspects will be outlined using 1dimensional arrays for simplicity reasons. Antennas exhibit a specific radiation pattern,

which overall radiation pattern changes when several antenna elements are combined in an array.

The array factor quantifies the effect of combining radiating elements in an array without the element specific radiation pattern taken into account. The overall radiation pattern results in certain directivity and thus gain linked through the efficiency with the directivity. Directivity and gain are equal if the efficiency is 100%.

### 2.5.6 Characteristics of Spacecraft Antenna System

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 an 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.

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 inverse proportional with  $4\pi$  and square of distance. The relations for  $P_R$  and  $G_T$  are presented as follows:

$$P_R = P_T G_T A_R / 4\pi R^2 \text{ and } G_T = 4\pi A_T / \lambda^2 \quad (2.58)$$

where  $G_T$  = effective area of transmit antenna and  $\lambda$  = 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 ( $G_R$ ) presented as:

$$P_R = P_T G_T G_R \lambda^2 / (4\pi R)^2 \text{ or } P_R = P_T G_T G_R / L_P L_K \text{ [W]} \quad (2.59)$$

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

$$P_R/N = P_T G_T (G_R/T_R) (1/K L_P L_K) \quad (2.60)$$

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 ( $1.38 \times 10^{-23}$  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 ( $N$ ), i.e., 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 GES/FES/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 onboard 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. 2.25 (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 towards 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:

$$G_a = \eta (4\pi A/\lambda^2) = 4\pi A_E/\lambda^2 \text{ and } G_p = \eta (\pi D)^2/\lambda^2 \quad (2.61)$$

where  $\eta$  = 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 the following relations:

$$G_p = \eta (\pi D f/c)^2 = 60,7 (D f)^2 \quad (2.62)$$

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

$$G_T = 10 \log G_p \quad (2.63)$$

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, 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:

$$F(\delta_o) = s(\delta_o)/s(\delta_o = 0) \quad (2.64)$$

where parameter  $s(\delta_o)$  = flow density of radiation in the hypothetical satellite angle ( $\delta_o$ ) and  $s(\delta_o = 0)$  = flow density in the middle of the coverage area. Looking the geometrical relations in Fig. 2.3 (a) follows the relation:

$$F(\delta_o) = d_o/h = \cos \delta \sqrt{(k^2 - \sin^2 \delta_o)} / (1 - k) \quad (2.65)$$

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

$$F(\delta) = k \cos \delta \quad (2.66)$$

For GEO satellite the value of  $\Delta L$  is given as a function of angle  $\delta$ , which is the distance from the centre of the coverage area, where the function diagram of the radiation is as follows:

$$\begin{aligned} F(\delta) = \Delta L &= 20 \log R/(R + h) \cos \delta \\ &= 10 \log R/(1 + 2R/h) \text{ [dB]} \end{aligned} \quad (2.67)$$

Therefore, in the case of GEO satellites the losses of antenna propagation are greater around the periphery than in the centre 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:

$$L_P = (4\pi d/\lambda)^2 \text{ and } P_R/S = P_T G_T / 4\pi d^2 L_K \quad (2.68)$$

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

## 2.6 Satellite Bus

The satellite bus is usually called a platform and consists in several sections, shown in Fig. 2.19. 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.

### ***2.6.1 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.

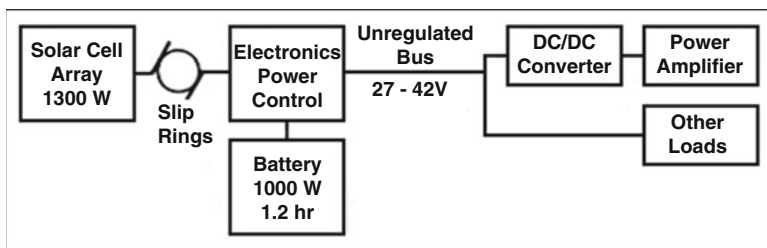
It comprises a skeleton on which the equipment modules are mounted and a panel, which covers and provides protection for sensitive parts during the operational phase from micrometers and helps to shield the equipment from extremes of heat, coldness, vacuum and weightlessness, including the relatively 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 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.

### ***2.6.2 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) Power electric generator, usually solar cell arrays located on the spinning body of a spin-stabilized satellite or on the paddles for a three-axes stabilized satellite; (2) Reliable electrical storage devices, such as batteries, for operating during periods of solar eclipses; (3) 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 TT&C satellite system.



**Fig. 2.26** Satellite electric power subsystem (Courtesy of Book: by Gordon)

The solar arrays are the motor during entire life of the satellite, providing sufficient power to all active components. Occasionally the Earth or Moon blocks the Sun, then researchable batteries provides secondary electrical power. A typical satellite power subsystem is shown in Fig. 2.26. 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 parallel 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. During exploitation, batteries are sometimes reconditioned by intentionally discharging them to a low charge level and recharging again, which prolongs their life.

The operational status of onboard batteries including recharge, in-service or reconditioning is remotely controlled by a special ground segment. Thus, 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 life-time. These batteries provide a low specific energy of about 30 to 40 Wh/kg. The latest type of Ni-H (Nickel-Hydrogen) batteries can store at least 50% more energy per kilogram.

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.

### 2.6.3 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 communication satellite is very important factor during entire satellite lifetime, which is necessary to achieve normal temperature balance and proper performance of all subsystem. In spacecraft design, the TC system has the function to keep all parts of spacecraft within acceptable temperature ranges during all mission phases. It is essential to guarantee the optimum performance and success of the mission, because if a component encounters a temperature, which is too high or too low, so could be damaged or affected.

Thermal stress to the satellite constructions results from high temperature effects from the Sun and from 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. In such a way, 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  $1 \text{ kW/m}^2$ , while other parts are facing the shadow side at a temperature of about  $-270^\circ\text{C}$ . In addition, an eclipse causes temperature variation from around  $-180^\circ$  to  $+60^\circ\text{C}$ , when the ambient temperature falls well below  $0^\circ\text{C}$  and rises rapidly from the moment the satellite emerges from the eclipse. In fact, all these extremes in space have to be eliminated or moderated for normal satellite operations, especially because all electronic devices need optimum temperatures between  $-5^\circ$  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.

### 2.6.4 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. Many 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  $0.1^\circ$  and  $0.01^\circ$ . In Fig. 2.27 is shown 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 is to keep the antenna RF beam pointing at the intended areas on the Earth, which procedure involves as follows:

- (a) Measuring the attitude of the satellite by sensors;
- (b) Comparing the results of measurements with the required values;
- (c) Calculating the corrections to reduce eventual errors and
- (d) Introducing these corrections by operating the appropriate torque units.

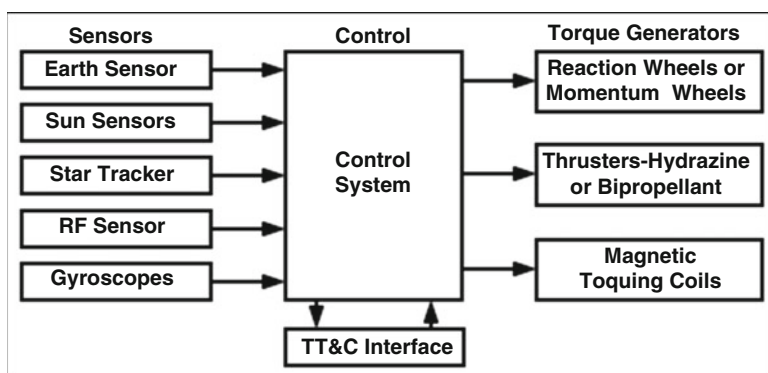


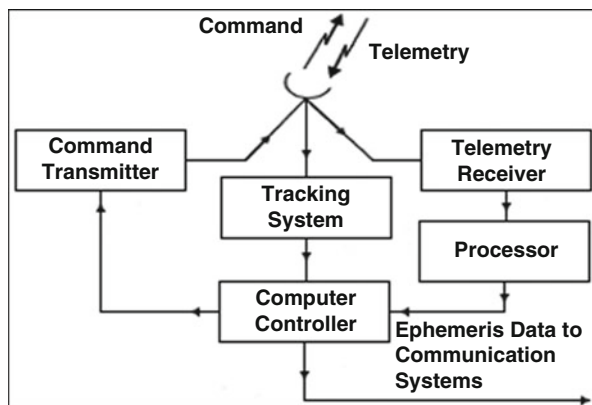
Fig. 2.27 Satellite AOC subsystem (Courtesy of Book: by Gordon)

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 reduces 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 a combination of 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** – On-board 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, iridium-coated rhenium chambers for chemical propellants, etc. 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 GES terminals. 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 depends 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 towards 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 towards 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.

### ***2.6.5 Telemetry, Tracking and Command (TT&C)***

The telemetry, tracking, command and communication equipment enables data to be sent continuously to the Earth stations, received from these stations and allows

**Fig. 2.28** Satellite TT&C  
(Courtesy of Book: by Maini)



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 like switching the transponders in and out of service, switching between redundant units, etc. The TT&C subsystem monitors and controls the satellite functions right from the lift-off stage and to the end of its operational life in space. So, it is 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. 2.28 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 sub-systems. The main functions of a TT&C are as follows:

1. **Telemetry Sub-System** – 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 Centre (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 Sub-System** – 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 GES 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.

3. **Command Sub-System** – This sub-system 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 follows: 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.

### ***2.6.6 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 centre of the mass to achieve the maximum thrust, the thrust being applied perpendicular to the direction of a spacecraft's centre of mass.

The orbit control thrusters are mounted so that the thrust vector passes through the centre 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.

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