

## SPACE SEGMENT

This chapter describes orbital mechanics and their significance with regard to satellite use for mobile communications. Namely, here are introduced the fundamental laws governing satellite orbits and the principal parameters that describe the motion of the Earth's artificial satellites. The types of satellite orbits are also classified, presented and compared from the MSC system viewpoint in terms of coverage and link performances.

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

### 2.1. Platforms and Orbital Mechanics

The platform is an artificial object located in orbit around the Earth at a minimum altitude of about 20 km in the stratosphere and a maximum distance of about 36,000 km. 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 communication satellites project in the phases of orbital design and operations. The orbital design is based on a generic survey of orbits and at an early stage in the MSC project is tasked to identify the most suitable orbit for the objective MSC service. The orbital operation is based on rather short-term knowledge of the orbital motion of the satellite and starts with TT&C maintenances after the satellite is located in orbit. In effect, only a few types of satellite orbits are well-suited for MSC and navigation systems.

#### 2.1.1. Space Environment

The satellite service begins when a spacecraft is located as a space platform in the desired orbital position in a space environment around the Earth. This space environment is a very specific part of the Universe, where many factors and determined elements affect the planet and satellite motions. The Earth is surrounded by a thick layer of many different gasses known as the atmosphere, whose density decreases as the altitude increases. Hence, there is

no air and the atmosphere disappears at about 180 km above the Earth, where the Cosmos begins. The endless environment in space is not very friendly and is extremely destructive, mainly because there is no atmosphere, the cosmic radiation is very powerful, the vacuum creates very high pressure on spacecraft or other bodies and there is the negative influence of very low temperatures.

The Earth's gravity keeps everything on its surface. All the heavenly bodies such as the Sun, Moon, planets and stars have gravity and reciprocal reactions. Any object flying in the atmosphere continues to travel until it meets forces due to the Earth's gravity or until it has enough speed to surpass gravity and to hover in the stratosphere. However, to send an object into space, it first has to overcome gravity and then travel at least at a particular minimum speed to stay in space. In this case, an object traveling at about 5 miles/sec 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 rocket. Rockets or launchers 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 atmosphere or space. As the payload is carried on the top, the rocket is usually separated and drops each stage after burn-out. Finally, a rocket brings a payload up to the required velocity and leaves it in orbit. Sometimes, a rocket is also known as a booster, as a rocket starts with a low velocity and attains some required height, where air drag decreases and it attains a higher velocity.

### 2.1.2. Laws of Satellite Motion

A satellite is an artificial object located by rocket in space orbit following the same laws in its motion as the planets rotating around the Sun. In this sense, three so important laws for planetary motion were derived by Johannes Kepler, as follows:

**1. First Law** – The orbit of each planet follows an elliptical path in space with the Sun in one focus. Motion lies in the plane around the Sun (1602).

**2. Second Law** – The line from the Sun to planet or radius vector ( $r$ ) sweeps out equal areas in equal intervals of time. This is the Law of Areas (1605).

**3. Third Law** – 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 presents 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.

Thus, 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:

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

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 ( $M$ ).

### 2.1.2.1 Parameters of Elliptical Orbit

The satellite in circular orbit undergoes its revolution at a fixed altitude and with fixed velocity, while a satellite in an elliptical orbit can drastically vary its altitude and velocity during one revolution. The elliptical orbit is also subject to Kepler's Three Laws of satellite motion. Thus, the characteristics of elliptical orbit can be determined from elements of the ellipse of the satellite plane with the perigee ( $\Pi$ ) and apogee ( $A$ ) and its position in relation to the Earth, see **Figure 2.1. (A)**. The parameters of elliptical orbit are presented as follows:

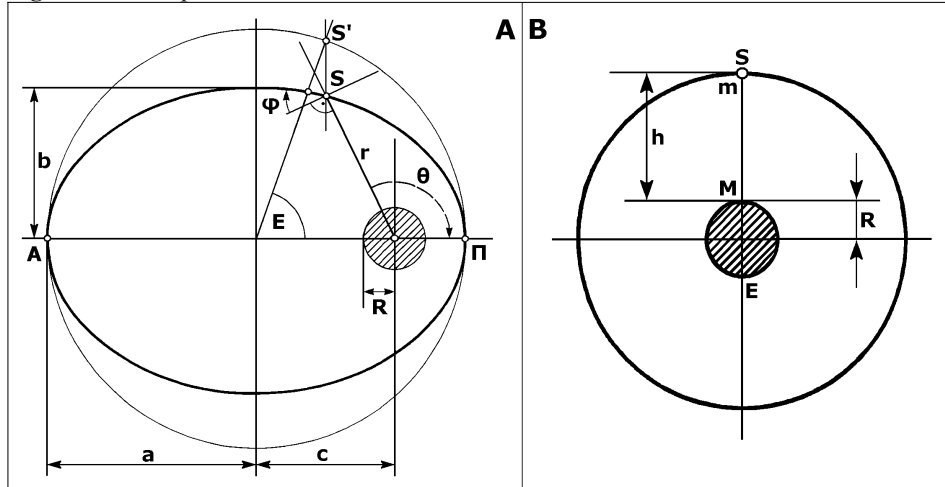
$$e = c/a = \sqrt{1 - (b/a)^2} \quad \text{or} \quad e = (\sqrt{a^2 - b^2}/a) \quad p = a(1 - e^2) \quad \text{or} \quad p = b^2/a \quad (2.2.)$$

$$c = \sqrt{a^2 - b^2} \quad a = p/1 - e^2 \quad b = a \sqrt{1 - e^2}$$

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

$$r = p/1 + e \cos \Theta \quad [\text{m}] \quad (2.3.)$$

where  $r$  = distance of the satellites from the centre of the Earth ( $r = R+h$ ) or radius of path;  $\Theta$  = true anomaly and  $E$  = eccentric anomaly. In this case, the position of the satellite will be determined by the angle called "the true anomaly", which can be counted positively in the direction of movement of the satellite from 0° to 360°, between the direction of the perigee and the direction of the satellite ( $S$ ). The position of the satellite can also be defined by eccentric anomaly, which is the argument of the image in the mapping, which transforms the elliptical trajectory into its principal circle, an angle counted positively in the direction of movement of the satellite from 0 to 360°, between the direction of the perigee and the direction of the satellite. The relations for both mentioned anomalies are given by the following equations:

**Figure 2.1.** Elliptical and Circular Satellite Orbits

Courtesy of Book: "Telekomunikacije satelitima" by R. Galić

$$\cos \Theta = \cos E - e/1 - e \cos E \quad \cos E = \cos \Theta + e/1 + e \cos \Theta \quad (2.4.)$$

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  $\mu=3.99 \times 10^5 \text{ km}^3/\text{sec}^2$ ), the velocity of a satellite in an elliptical orbit may be obtained from the following relation:

$$v = \sqrt{GM (2/r) - (1/a)} = \sqrt{\mu (2/r) - (1/a)} \quad (2.5.)$$

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

$$t = 2\pi \sqrt{(a^3/GM)} = 2\pi \sqrt{(a^3/\mu)} \quad (2.6.)$$

$$t = 3.147099647 \sqrt{(26,628.16 \cdot 10^3)^3 \cdot 10^{-7}} = 43,243.64 \text{ [s]}$$

Therefore, the last equation is the calculated period of sidereal day for the elliptical orbit of Russian-based satellite Molnya with apogee = 40,000 km, perigee = 500 km, revolution time = 719 min and  $a = 0.5 (40,000 + 500 + 2 \times 6,378.16) = 26,628.15 \text{ km}$

### 2.1.2.2. Parameters 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 **Figure 2.1. (B)**. 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:

$$\tau = t / (1 - \Omega t / 2\pi) \quad (2.7.)$$

$$t = 2\pi \sqrt{(r^3/\mu)} = 3.147099647 \sqrt{(r^3 \cdot 10^{-7})} \quad [\text{s}] \quad (2.8.)$$

$$h = [\sqrt[3]{(\mu t^2/4\pi^2)}] - R = 2.1613562 \cdot 10^4 (\sqrt[3]{t^2}) - 6.37816 \cdot 10^6 \quad [\text{m}] \quad (2.9.)$$

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:  $t_E = 23 \text{ h } 56 \text{ min } 4.09 \text{ sec}$ . 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:

$$v = \sqrt{(MG/R + h)} = \sqrt{(\mu/r)} = 1.996502 \cdot 10^{-7}/\sqrt{r} = 631.65 \sqrt{r} \quad [\text{m/s}] \quad (2.10.)$$

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

$$r = \sqrt[3]{[(\mu t) / 2\pi]^2} \quad (2.11.)$$

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:

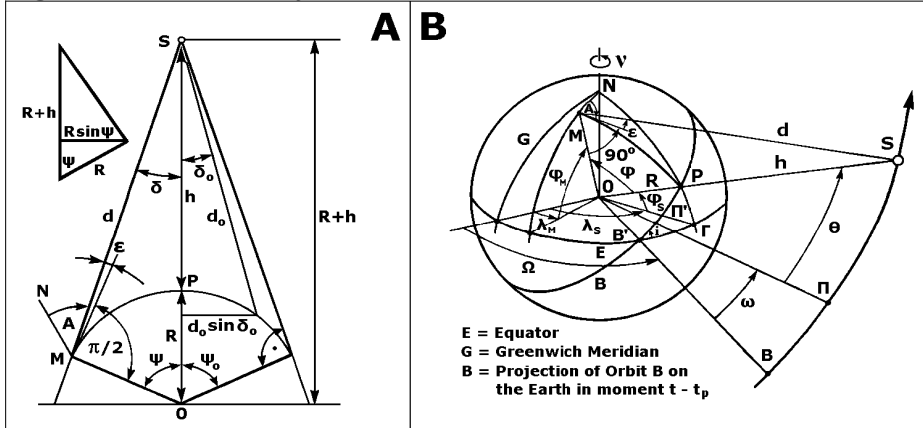
$$t_a = t_E \cdot t/t_E - t \quad (2.12.)$$

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.

**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	6,052.00	s
h	35,786.00	20,183.62	10,354.71	4,162.89	800.00	km
(R+h)	42,164.00	26,561.78	16,732.87	10,541.05	7,178.00	km
v	3,075.00	3,873.83	4,880.72	5,584.12	7,450.00	km/s <sup>-1</sup>

According to **Table 2.1.** and equation (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 1,700, 10,400 and 36,000 km, will have  $t/\tau$  values 2/2,18, 6/8 and 24/zero, respectively. In this case, it is evident that only a satellite constellation at altitudes of about 36,000 km can be synchronous or geostationary.

**Figure 2.2.** Geometric Projection of Satellite Orbits

Courtesy of Books: (A) "Sputnikovaya svyaz na more" by L. I. Novik, I. D. Morozov and V. I. Solobev and (B) "Mezhdunarodnaya sputnikovaya sistema morskoy svyazi – Inmarsat" by V. A. Zhilin

### 2.1.3. 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 **Figure 2.2. (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 ( $\Psi$ ) or sub-satellite angle, which is similar to the angle of antenna radiation ( $\delta$ ).

The geographical and horizon coordinates of a satellite are presented in **Figure 2.2. (B)** with the following, not yet mentioned, main parameters: angular speed of the Earth's rotation ( $v$ ), argument of the perigee ( $\omega$ ), moment of satellite pass across any point on the orbit ( $t_0$ ), 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  $t_0$  ( $\Omega_0$ ).

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.

#### 2.1.3.1. Satellite Distance and Coverage Area

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

$$\delta + \epsilon + \Psi = 90^\circ \quad (2.13.)$$

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

$$R_s = R \sin \Psi \quad (2.14.)$$

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:

$$h_s = R (1 - \cos \Psi) \quad (2.15.)$$

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's transmissions with an elevation angle larger than a prescribed minimum angle; **(2)** The slant range (actual line-of-sight 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 **Figure 2.2. (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:

$$C = \pi (R_s^2 + h_s^2) = 2\pi R^2 (1 - \cos \Psi) \quad (2.16.)$$

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:

$$C/4\pi R^2 = 0,5 (1 - \cos \Psi) \quad (2.17.)$$

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

$$z = [(R \sin \epsilon)^2 + 2Rh + h^2]^{1/2} - R \sin \epsilon \quad (2.18.)$$

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

$$t_d = (3.33) z \quad [\mu\text{sec}] \quad (2.19.)$$

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

$$t_p = (2\Psi/360) (t/1 \pm (t/t_E)) \quad (2.20.)$$

The quotations for right ascension of the ascending node angle ( $\Omega$ ) and argument of the perigee ( $\omega$ ) are as follows:

$$\Omega = 9,95 (R/r)^{3.5} \cos i \quad \text{or} \quad \Omega = \Omega_0 + v (t - t_0) \quad (2.21.)$$

$$\omega = 4,97 (R/a)^{3.5} [5 \cos^2 i - 1/(1 - e^2)^2]$$

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

$$\Psi = \arccos (R \cos \epsilon / r) - \epsilon \quad \text{or} \quad (2.22.)$$

$$\Psi = \pi/2 - \arcsin (R/r) = \arccos (R/r) - \epsilon = \arccos k - \epsilon$$

$$\Psi = \arccos 6,376.16/42,164.20 = \arccos 0.15126956 = 81^\circ 17' 58.18''$$

$$C_{\max} = 255.61 \cdot 10^6 (1 - 0.15126956) = 216.94 \cdot 10^6 \text{ (km}^2\text{)}$$

Therefore, all MES and LES with a position above  $\Psi=81^\circ$  will be not covered by GEO satellites. Since the Earth's square area is 510,100,933.5 km<sup>2</sup> and the extent of the equator is 40,076.6 km, only with three GEO mutually moved apart in the orbit by 120° it is possible to cover a great area of the Earth's surface, see **Figure 2.3. (A)**. The zero angles of elevation have to be avoided, even to get maximum coverage, because this increases the noise temperature of the receiving antenna. Owing to this problem, an equation for the central angle with minimum angle of view between 5° and 30° will be calculated with the following relation:

$$\Psi_s = \arccos (k \cos \epsilon) - \epsilon \quad (2.23.)$$

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

$$l = 2\pi R (2\Psi/360 = 222.64\Psi \text{ [km]}) \quad (2.24.)$$

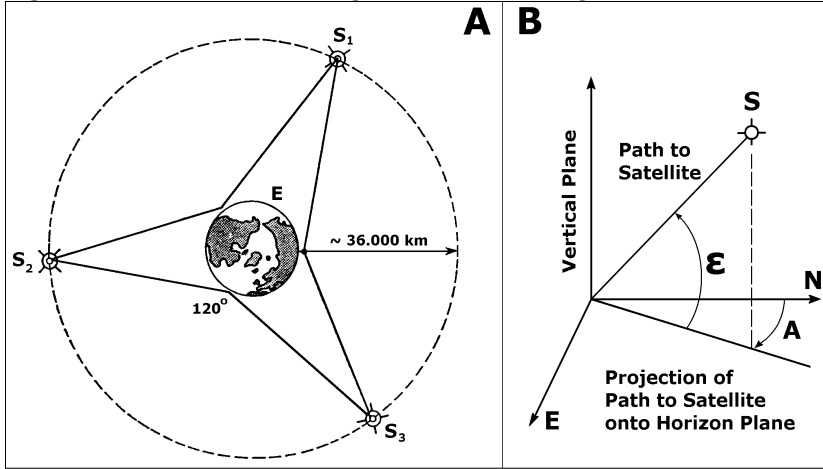
The real altitude of satellite over sub-satellite point is as follows:

$$h = r - R = 42,162 - 6,378 = 35,784 \text{ [km]} \quad (2.25.)$$

The view angle under which a GEO satellite can see LES/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  $\epsilon=5^\circ$ . Such access is very easy to calculate, using simple trigonometry relations:

$$\delta_{\epsilon=0} = \arcsin k \approx 9^\circ \quad (2.26.)$$



**Figure 2.3.** GEO Satellite Configuration and Look Angle Parameters

Courtesy of Books: (A) "Telekomunikacije satelitima" by R. Galić and  
(B) "Satellite Communications" by T. Pratt and Ch. W. Bostian

At any rate, the angle ( $\Psi$ ) is in correlation with angle ( $\delta$ ), which can determine the aperture radiation beam. For example, the aperture radiation beam of satellite antenna for global coverage has a radiation beam of  $2\delta=17.3^\circ$ . According to **Figure 2.2. (A)** it will be easy to find out relations for GEO satellites as follows:

$$\operatorname{tg} \delta = k \sin \Psi / 1 - k \cos \Psi = 0.15126956 \sin \Psi / 1 - \cos \Psi / 1 - 0.15126956 \cos \Psi \quad (2.27.)$$

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

Differently to say, the width of the beam aperture ( $2\delta_s$ ) is providing the maximum possible coverage for synchronous circular orbit. The distance of LES and MES with regard to the satellite can be calculated using **Figure 2.2 (A)**, and equations (2.13.) and (2.22.) by:

$$d = R \sin \Psi / \sin \delta = r \sin \epsilon / \cos \epsilon \quad (2.28.)$$

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

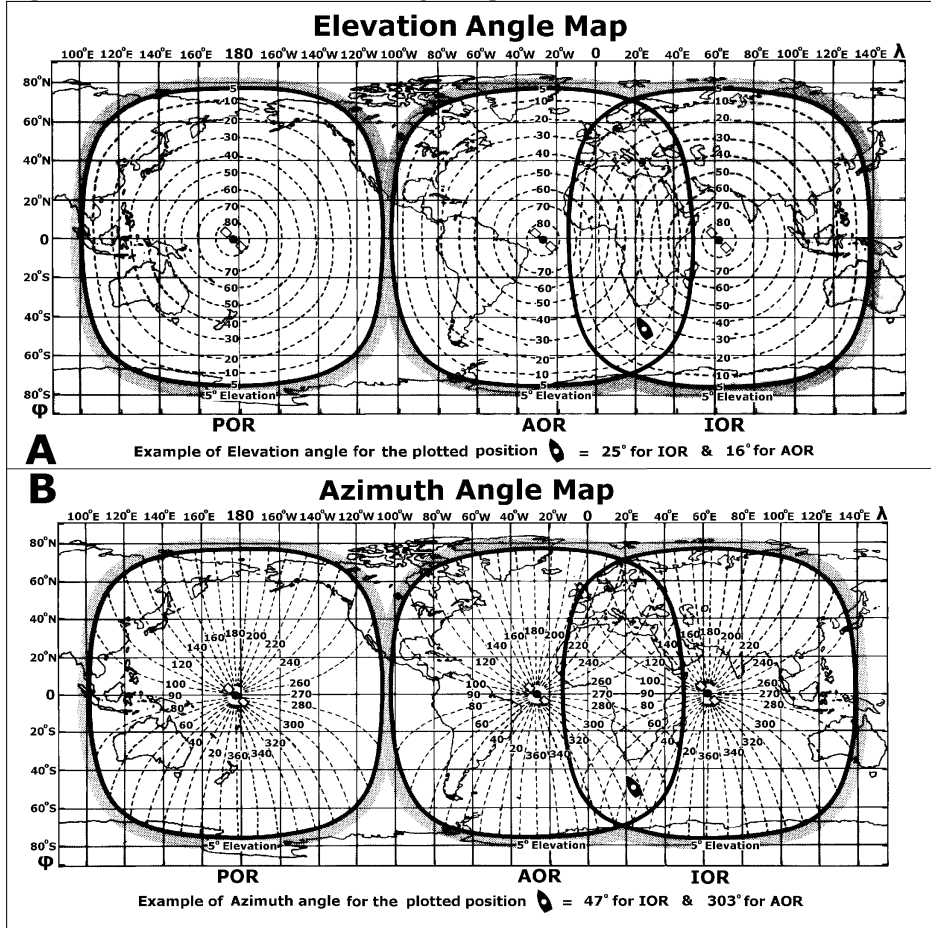
$$d = \sqrt{[(R + r)^2 - 2Rr \cos \Psi]} \quad \text{or} \quad (2.29.)$$

$$d = h \sqrt{[1 + 2(1/k)(R/h)^2(1 - \cos \varphi \cos \Delta \lambda)]} \quad \text{or}$$

$$d = r [1 - (R \cos \epsilon / r)^2]^{1/2} - R \sin \epsilon$$

Accordingly, when the position of any MES 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 MES will have a further position from (P) when the central angle exceeds  $\Psi = 81^\circ$ , when  $d_{\max}=41,643$  km.

Figure 2.4. Elevation and Azimuth Angle Maps



Courtesy of Manual: "Saturn 3 – Installation" by EB Communications

### 2.1.3.2. Satellite Look Angles (Elevation and Azimuth)

The horizon coordinates are considered to determine satellite position in correlation with an Earth observer, LES and MES terminals. These specific horizon coordinates are angles of satellite elevation and azimuth, shown in **Figure 2.2. (A and B)** and **Figure 2.3. (B)**.

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 **Figure 2.2. (A)** the look angle of  $\epsilon$  value can be calculated by the following relation:

$$\text{tg } \epsilon = \cos \Psi - k / \sin \Psi \quad (2.30.)$$

In **Figure 2.4. (A)** is illustrated the Mercator chart of the 1<sup>st</sup> Generation Inmarsat space segment, using three ocean coverage areas with projection of elevation angles and with one

example of a plotted position of a hypothetical ship (may also be aircraft or any mobile). Thus, it can be concluded that SES or any type of MES at designated position ( $\epsilon=25^\circ$  for IOR and  $\epsilon=16^\circ$  for AOR) has the possibility to use either GEO satellites over IOR or AOR to communicate with any LES inside the coverage areas of both satellites.

The satellite azimuth (A) is the angle measured eastward from the geographical North line to the projection of the satellite path on the horizontal plane at the observer point. This angle varies between 0 and  $360^\circ$  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 MES can be calculated as follows:

$$\operatorname{tg} A' = \operatorname{tg} \Delta\lambda_M - k/\sin \Psi \quad (2.31.)$$

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

$$\operatorname{tg} A = \sin \Delta\lambda / \operatorname{tg} \varphi \quad \text{or} \quad \sin A = \cos \varphi \sin \Delta\lambda \operatorname{cosec} \Psi \quad (2.32.)$$

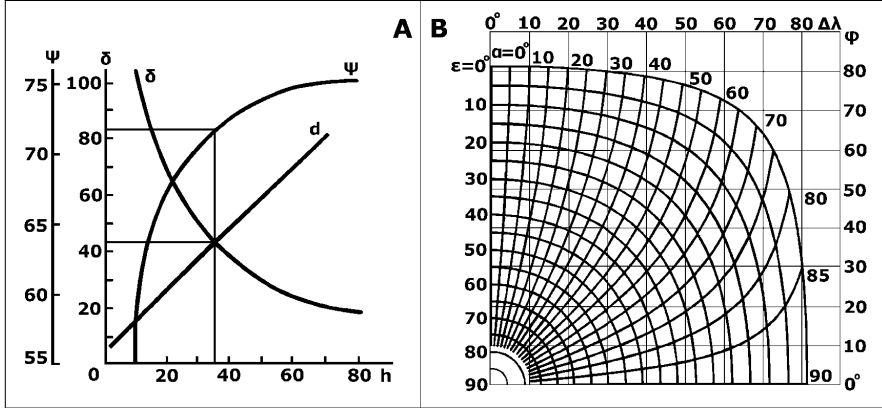
In **Figure 2.4. (B)** is illustrated the Mercator chart of 1<sup>st</sup> Generation Inmarsat 3-satellite or ocean coverage areas with projection of azimuth angles, with one example for the plotted position of a hypothetical ship ( $\epsilon=47^\circ$  for IOR and  $\epsilon=303^\circ$  for AOR). Any mobile inside of both satellites' coverage can establish a radio link to the subscribers on shore via any LES. However, parameter (A') is the angle between the meridian plane of point (M) and the plane of a big circle crossing this point and 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 **Figure 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 ( $\delta$ ) is 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 LES. The single angle of the satellite in circular orbit depends on the  $t/2$  value, which in area of satellite look angle, can be found in the duration of the time and is determined as:

$$t_c = \Psi / \pi \quad (2.33.)$$

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 SES ( $\varphi$ ,  $\lambda$ ), 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 **Figure 2.5. (B)** a graphic design is shown which can approximately determine limited zones of satellite

**Figure 2.5.** Look Angle Parameters and Graphic of Geometric Coordinates for GEO

Courtesy of Book: "Mezhdunarodnaya sputnikovaya sistema morskoy svyazi – Inmarsat" by V. A. Zhilin

visibility from the Earth (MES) by using elevation and azimuth angles under the condition that  $\delta = 0$ . This graphic contains two groups of crossing curves, which are used to compare  $(\varphi)$  and  $(\Delta\lambda)$  coordinates of mobile positions. Thus, the first group of parallel concentric curves shows the geometric positions where elevation has the constant value ( $\epsilon=0$ ), while the second group of fan-shaped curves starting from the centre shows the geometric positions where the difference in azimuth has the constant value ( $\alpha = 0$ ). This diagram can be used in accordance with **Figure 2.2. (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.
3. The difference value of azimuth ( $\alpha$ ) 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**.

**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	$A = 360^\circ - a$

Inasmuch as the position of SES 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 AES, then the elevation angle will be compensated by the following parameter:

$$x = \arccos(1 - H/R) \quad (2.34.)$$

where  $H$  = height above sea level of observer or MES. Let us say, if the position of LES is a height of  $H = 1,000$  m above sea level, the value of  $x \approx 1^\circ$ . This example can be used for

the determination of AES compensation parameters, depending on actual aircraft altitude. In such a way, the estimated value of elevation angle has to be subtracted for the value of the compensation parameter ( $x$ ).

### 2.1.3.3. Satellite Track and Geometry (Longitude and Latitude)

The satellite track on the Earth's surface and the presentation of a satellite's position in correlation to the MES results from a spherical coordinate system, whose centre is the middle of Earth, is illustrated in **Figure 2.2. (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.

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

$$\sin \varphi = \sin (\Theta + \omega) \sin i \quad (2.35.)$$

$$\operatorname{tg} (\lambda_S - \Omega) = \operatorname{tg} (\Theta + \omega) \cos i$$

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 **Figure 2.2. (B)**, what can be the position of the MES, taking into consideration the arc ( $MP$ ) of the angle illustrated in **Figure 2.2. (A)**, the central angle can be calculated by the following relations:

$$\cos \Psi = \cos \varphi_S \cos \Delta\lambda \cos \varphi_M + \sin \varphi_S \sin \varphi_M \quad \text{or} \quad (2.36.)$$

$$\cos \Psi = \cos \text{arc } MP = \cos \varphi_M \cos \Delta\lambda$$

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

$$\cos \Psi = \cos \varphi \cos \Delta\lambda \quad \text{and} \quad \operatorname{tg} A = \sin \Delta\lambda / \operatorname{tg} \varphi, \quad \text{respectively} \quad (2.37.)$$

$$\sin \varphi = \sin \Psi \cos A \quad \text{and} \quad \operatorname{tg} \Delta\lambda = \operatorname{tg} \Psi \sin A$$

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:

$$n = 180^\circ / \Psi \quad (2.38.)$$

For instance, if  $\delta = 0$  and  $\Psi = 81^\circ$  it will be necessary to put into orbit only 3 GEO, and to get a global coverage from  $70^\circ$  N to  $70^\circ$  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) requires a propagation time equal to:

$$T = d/c \quad (\text{s}) \quad (2.39.)$$

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). Namely, 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. In effect, the Doppler shift at a transmitting frequency ( $f$ ) and radial velocity ( $v_r$ ) between the observer and the transmitter can be calculated by the following relation:

$$\Delta f_D = f v_r / c \quad \text{where} \quad v_r = dR/dt \quad (2.40.)$$

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

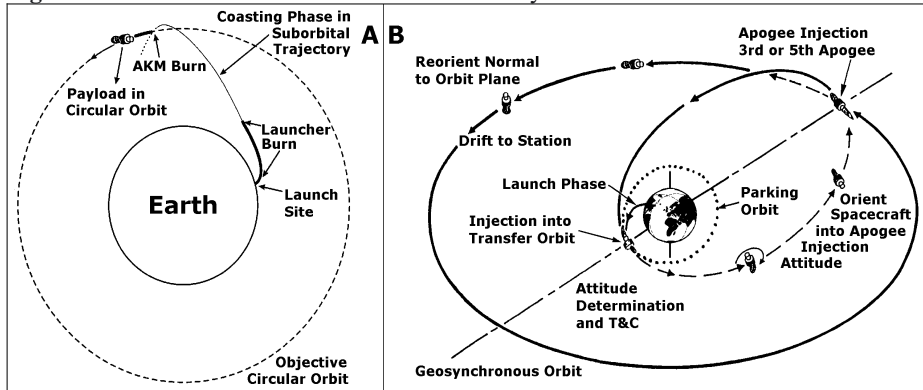
$$v_r = dr/dt = (dr/\Theta) (d\Theta/dt) \quad (2.41.)$$

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.

## 2.2. 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-based communication technology and the most expensive phase of the total system cost. At the same time, the need to make a satellite body capable of surviving the stresses of the launch stages is a major element in their design phase. Satellites are also designed to be compatible with more than one model of launch vehicle and launching type. In a more determined sense, there are 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 communications satellite is based mostly on launching into an equatorial circular orbit, in particular the GEO but broadly similar processes or phases are used for all types of orbits. Otherwise, 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.

**Figure 2.6.** Satellite Installation in Circular and Synchronous Orbit

Courtesy of Book: "Commercial Satellite Communications" by S.C. Pascall and D.J. Withers

### 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 3,070 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, 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 **Figure 2.6. (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 **Figure 2.6. (B)**. The Hohmann transfer ellipse method enables a satellite to be placed in an orbit at the desired altitude using the trajectory that requires the least energy. At the first sequence the launch vehicle propels the satellite into a low parking orbit by the direct ascent method. The satellite is then injected into an elliptical transfer orbit, the apogee of which is the altitude of the desired circular synchronous orbit. At the apogee, additional thrust is applied by an AKM to provide the velocity increment necessary for the attainment of the required synchronous orbit. In practice it is usual for the direct ascent method to be used to inject a satellite into a LEO and for the Hohmann transfer ellipse to be used for higher types of 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) and H-II (Japan). 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 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 8,000 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 **Figure 2.7. (A)** and **(B)**, respectively and also 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 4,500 and 2,600 kg into GTO and GEO, respectively.

#### **2.2.2.2. Reusable Launch Vehicles**

Reusable launch vehicles have already been developed in the USA (Space Shuttle) and former USSR (Energia/Buran), which have as their aim the development of vehicles that could journey into space and return, all or much of their structure being reusable and thus, the satellite launching will cost less. Moreover, in using these launchers there will be less burnt-out upper stages than with expendable vehicles. What remains in space, the small pieces in transfer orbits for many years and much small debris, remains in LEO for a long time, adding to the growing space junk hazard for operational satellites and future space operations.

There are other projects for development of similar vehicles such as a small manned reusable space shuttle called Hermes (Europe) and Hope (Japan). In the UK an unmanned space plane Hotol is proposed, while in Germany and the USA two similar vehicles are projected: TAV (TransAtmospheric 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.

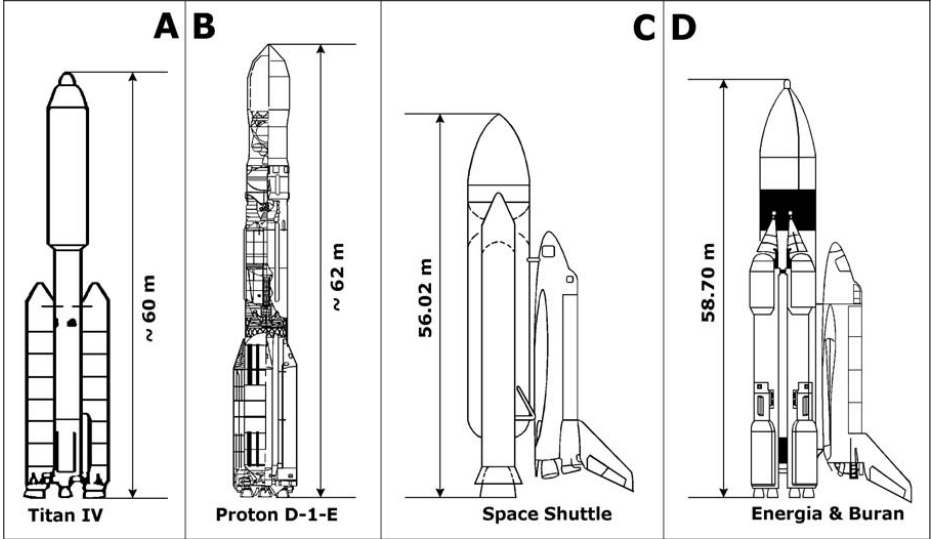
**1. Space Shuttle** – The US-based NASA developed a fleet of manned reusable vehicles of Space Transportation System (STS) called Space Shuttle, which are capable of lifting a satellite of up to 29.5 tons into a parking orbit, inclined at  $28.5^\circ$ , with an altitude of up to 431 km, shown in **Figure 2.7. (C)**. 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.

**2) Energia/Buran Space plane** – The launcher Energia is the most powerful operational reusable vehicle in the world, capable of carrying about 100 tons into space, whose four first-stage booster units are recoverable for reuse. In particular, it can launch the Buran space plane, enabling it to acquire a LEO and to land with the aid of its own rocket engine, shown in **Figure 2.7. (D)**. 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. Thus, 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 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. This compact shuttlecraft could be placed on top of a Proton launcher.

Figure 2.7. Expendable and Reusable Launch Vehicles



Courtesy of Book: “Commercial Satellite Communications” by S.C. Pascall and D.J. Withers

### 2.2.2.3. Land-Based Launching Systems

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

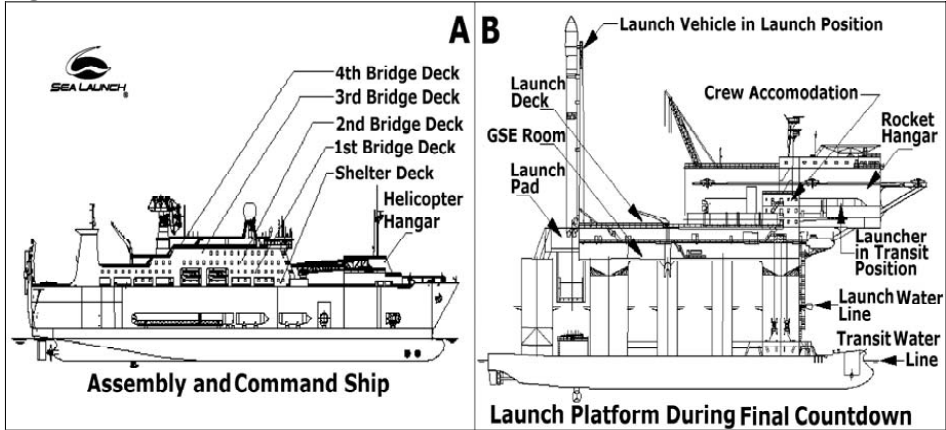
- 1. US-Based Launch Centres** – The USA launches satellites from two main locations, in Florida Cape Canaveral, suitable for direct equatorial orbit and the Vandenberg Air Force Base in California, suitable for polar orbit missions.
- 2. Russian Launch Centres** – Russian satellites are launched from two main launch centres named Baikonur and Northern Cosmodrome. Baikonur lies north of Tyuratam in Kazakhstan, with all the 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.
- 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 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.

**Figure 2.8.** Sea Launch Modules

Courtesy of Manual: "User's Guide" by Sea Launch Company

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 Ship (ACS), 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 **Figure 2.8. (A)** and **(B)**, respectively. However, transit for the ACS and the LP from Home Port in Long Beach to the launch site on the equator takes 10 to 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 ACS 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 ACS 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 ACS include: a rocket assembly compartment; the LCC with viewing room; helicopter capability; spacecraft contractor and customer work areas and spacecraft contractor and customer accommodation. The rocket assembly compartment, which is located on the main deck of the ACS, 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 ACS before sailing from the port of Long Beach to the designated launch site. A launcher with a payload will then be transferred in the horizontal position to the launch pad on LP and raised to a vertical position for fueling and launching.

During the launch sequence, the crew of the LP will be transferred to the ACS, which will initiate and control the launch from a position about 3 miles away from the LP pad. The LP is an extremely stable sea platform from which to conduct the launch, control and other operations. The LP rides catamaran-style on a pair of large pontoons and is self-propelled by a four-screw propulsion system (two in each lower hull, aft), which is powered by four direct-current double armature-type motors, each of which are rated at 3,000 hp. The LP in navigation has normal draft at sea water level but once at the launch location, the pontoons are submerged to a depth of 22.5 m to achieve a very stable launch position, level to within approximately 1°. The ballast tanks are located in the pontoons and in the lower part of the columns. Six ballast pumps, three in each pontoon, serve them. The LP has an overall length of approximately 133 m at the pontoons and the launch deck is 78 by 66.8 m. The Zenit-3SL launcher is a two-stage liquid propellant launch vehicle solution capable of transporting a spacecraft to a variety of orbits. The original two-stage Zenit was designed by KB Yuzhnoye quickly to reconstitute former Soviet military satellite constellations. The design emphasizes robustness, ease of operation and fast reaction times. The result is a highly automated launch capability using a minimum complement of launch personnel. The launcher as an integrated part of the Sea Launch system is designed to place spacecraft into a variety of orbits and is capable of putting 5,250 kg of payload into GEO.

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

**1. Communications** – Internal communications systems are distributed between the ACS and LP. This system includes CCTV, telephones, intercom, video teleconferencing, public address and vessel-to-vessel radiocommunications, known as the Line-of-Sight (LOS) direct system.

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

**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 sec and continues until Zenit Stage 2/Block DM separation, at around 9 minutes. These data are routed from the NASA White Sands LES to the Sea Launch Brewster LES and to the ACS. Otherwise, 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 ACS. The data are received on board the ship through the Intelsat communications terminal and are routed to Room 15 for upper-stage data and Room 94 for PLU data. Simultaneously, Brewster routes Block DM data to the Energia Moscow control centre station. However, the TDRSS coverage continues until after playback of the recorded Block DM data.

**3. Telemetry System** – Sea Launch uses LOS telemetry systems for the initial flight phase, as well as the TDRSS for later phases. The LOS system, which includes the Proton antenna and the S-band system, is located on the ACS. 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 ACS has a self-contained WX station, which includes a motion-stabilized C-band Doppler radar, surface wind instruments, wave radar, upper-atmospheric balloon release station, ambient condition sensors and access to satellite imagery and information from an on-site buoy.

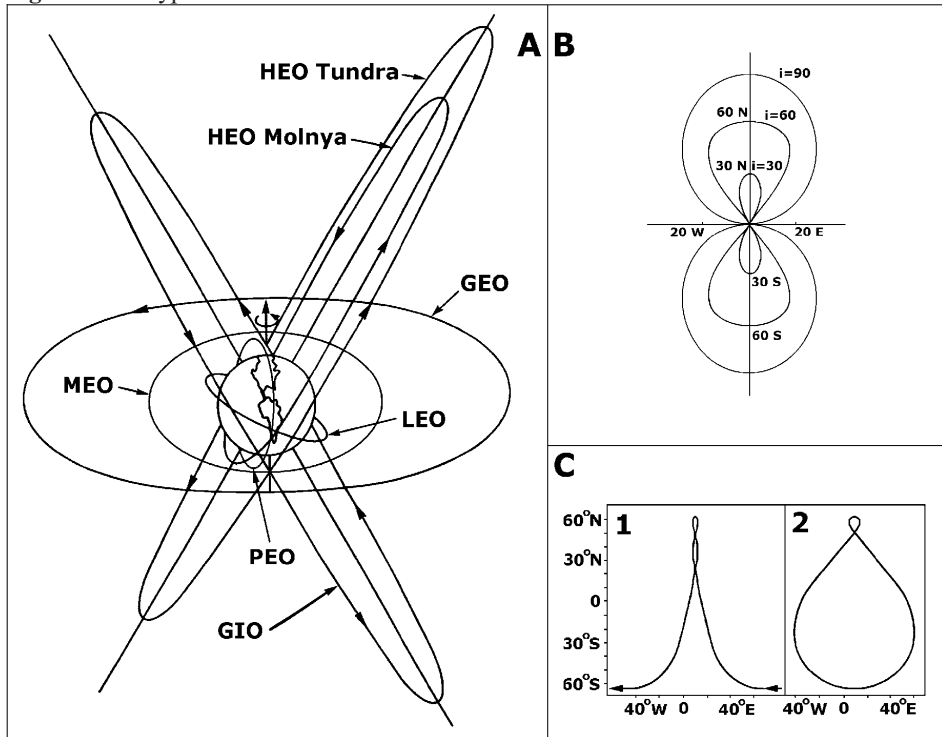
### 2.3. Types of Orbits for Mobile and Other Satellite Systems

An orbit is the circular or elliptical path that the satellite traverses through space. This path appears in the chosen orbital plane in the same or different angle to the equatorial plane. All 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 fixed and mobile satellite communication systems. In general, the one most commonly used orbit for satellite communications is GEO constellations, after which HEO and latterly GIO, PEO, LEO and MEO, shown in **Figure 2.9. (A)**.

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



**Figure 2.9.** Type of Satellite Orbits and Tracks

Courtesy of Book: "Satellite Communication Systems" by B.G. Evans

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 extensive use of GEO is showing that it provides something good. Inmarsat is the biggest GEO operator whose service and revenue confirm this point of view. The advantages of Inmarsat MSC solutions can be realized if someone uses them such, as operators on board mobiles and finds out how powerful they are. Most of other regional GEO worldwide networks, such as ACeS, Optus and Thuraya are also more successful than other Non-GEO constellations.

Especially Big LEO and ICO systems or hybrid constellations such as Ellipso have had several years of serious economical and concept difficulties. It is sufficient to see **Table 2.3.** to understand that the major reasons for LEO problems are 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.

**Table 2.3.** The Properties of Four Major Orbits

Orbital Properties	LEO	MEO	HEO	GEO
Development Period	Long	Short	Medium	Long
Launch & Satellite Cost	Maximum	Maximum	Medium	Medium
Satellite Life (Years)	3–7	10–15	2–4	10–15
Congestion	Low	Low	Low	High
Radiation Damage	Zero	Small	Big	Small
Orbital Period	<100 min	8–12 hours	½ 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

The track of the satellite varies from 0 to 360°, see **Figure 2.9. (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° and 60° inclination angles and the last is the track of the PEO satellite with an inclined orbit plane to the equator of 90°. The tracks of HEO Molniya (part of the track) and Tundra (complete track) orbits are shown in **Figure 2.9. (C-1/C-2)**, respectively. These two tracks pass over the African Continent and almost all of Europe.

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

### 2.3.1. Low Earth Orbits (LEO)

The LEO systems are either elliptical or more usually circular satellite orbits between 500 and 2,000 km above the surface of the Earth and bellow the Inner Van Allen Belt. The orbit period at these altitudes varies between ninety minutes and two hours. The radius of the footprint of a communications satellite in LEO varies from 3,000 to 4,000 km. Therefore, the maximum time during which a satellite in LEO orbit is above the local horizon for an observer on the Earth is up to 20 minutes. In this case, the traffic to a LEO satellite has to be handed over much more frequently than all other types of orbit. Namely, when a satellite which is serving a particular user, moves below the local horizon, it needs to be able to quickly handover the service to a succeeding one in the same or adjacent orbit. Due to the relatively large movement of a satellite in LEO constellation with respect to an observer on



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

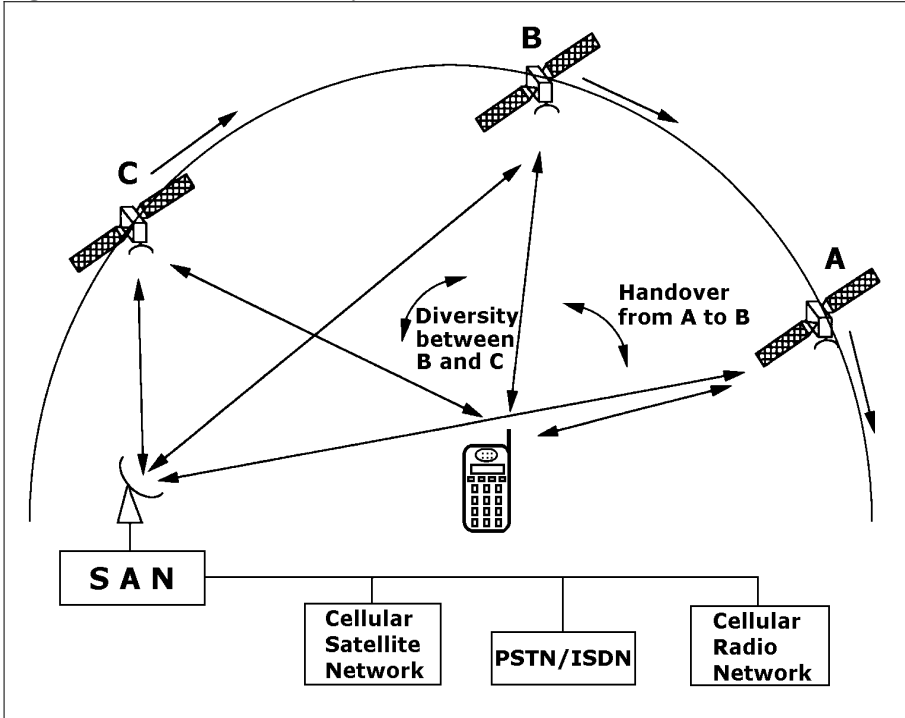
The major advantages of LEO are as follows:

- a)** The LEO system may become important in the field of MSC using handheld terminals with global roaming and to be exceedingly useful in areas not covered by cellular systems. The LEO constellations cover almost the entire Earth's surface and some of them provide polar coverage and show promise in the fields of mobile data and Internet and FSS networks for broadband data transmission and communications.
- b)** High Doppler shift allows the LEO system to be used for satellite positioning, tracking and determination.
- c)** The relatively small distance between LES and LEO results in much lower power and smaller user terminals. Furthermore, the one-way speed-of-light propagation delay of at least 0,25 sec using GEO is obviated with LEO, which effect can be annoying in two-way voice transmission. For example, for two-way voice via a satellite at an altitude of about 1,000 km, the delay is only 13 ms in total for uplink and downlink.
- d)** Satellite path diversity eliminates signal interruption due to path obstruction. In **Figure 2.10**, handover from satellite A to satellite B is demonstrated and path diversity between satellite B and C. This figure illustrates the LEO MSS space and ground architecture with utilization of handheld personal terminals (PES). On the other hand, the Satellite Access Node (SAN) is the LES providing a link between PES terminals through satellites and ground telecommunications infrastructures.

The disadvantages of LEO are as follows:

- a)** The orbit period at about 1,000 km altitude is in the order of approximately 100 min and the visibility at a point on the Earth is only some 10 min, requiring 40 to 80 satellites in six to seven planes for global coverage. Thus, in reality a GMSC system using this type of orbit requires a large number of satellites, in a number of different inclined orbits, which increases the total cost of the network.
- b)** Frequently handover is necessary for uninterrupted communications. Satellite visibility for MES could be improved by using more satellites. The optimum number of satellites of about 48 inclined in the constellation in a carefully optimized pattern of orbit planes will provide continuous visibility of one or other of the satellites at any location on Earth.
- c)** During times of the year that the orbital plane is in the direction of the Sun, a satellite in LEO is eclipsed for almost one-third of the orbit period. Consequently, there is a significant demand on battery power, with up to 5,000 charge/discharge cycles per year, which with existing NiCd types of batteries, reduces satellite lifetimes to 3–7 years.
- d)** The launch cost is low, with direct injection into the orbit of several satellites but the total cost is very high, with a minimum of 40 satellites being produced.

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

**Figure 2.10.** LEO MSS Diversity and Handover

Courtesy of Book: "Telecommunications Transmission Systems" by Hurdeman

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

### 2.3.1.1. Little LEO

The Little LEO mobile satellite systems are a category of LEO solutions that utilize birds of small size and low mass for low-bit-rate transmission under 1 Kb/s. Thus, the Little LEOS are constellations of very small Non-GEO satellites, which operate in LEO orbits providing mainly mobile data messaging and tracking services for vehicles and ships and other FSS and broadcasting services. The FCC has allocated a frequency band of 137–138 MHz for the downlink and 148–149.9 MHz for the uplink to Little LEO systems, which is a heavily utilized spectrum for private and government services worldwide, such as Orbcomm, Falsat, Leo One, VITASat, Starnet and other systems. The mass of satellites in these solutions range from 40 kg in Orbcomm to 150 kg in the Starnet system. Otherwise, these systems prefer a spectrum below 1 GHz, because it enables the use of cost-effective equipment. In this way, nonvoice two-way messaging and positioning with low cost transceiver, which would be equipped with an alphanumerical display, are the major characteristics of these systems.

### **2.3.1.2. Big LEO**

The Big LEO is a larger Non-GEO satellite system, which operates in LEO constellations and provides mainly mobile telephony, Fax, data and RDSS services. Compared to the Little LEO systems, satellites in Big LEO systems are expected to be bigger in body and to have more power and bandwidth to provide a different service to their subscribers. This system will use the underutilized spectrum available in the L-band, because of the commercial failure of proposed RDSS service. Currently, the frequency spectrum of 1610–1626.5 MHz for uplinks and 2483.5–2500 MHz for downlinks are assigned to these MSC systems. It is interesting to note that although the names of these systems include LEO, their frequencies are the ones usually utilized in MEO and GEO satellite systems. For this reason a new ICO system is systematized in the category of Big LEO constellations together with real Big LEO systems, such as Iridium and Globalstar, which are located at a lower altitude than ICO, at about 700–1,500 km from the Earth's surface. Thus, all of the new proposed Big LEO systems would offer global handheld telephone service by means of satellites on lower altitudes moving very fast, instead of fixed GEO relays. The bigger size of the satellites enables them to carry a transponder on board with more complex data processing facilities than the simple store-and-forward feature of the Little LEO satellite configuration. Hence, an important fact is that these systems are networking with cellular and spreading their roaming and billing capabilities in real global coverage.

### **2.3.2. Circular Orbits**

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

#### **2.3.2.1. Medium Earth Orbit (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. A LEO constellation for MSS global coverage requires around 10 satellites in two or three orbital planes, each plane inclining  $45^\circ$  to the equator. Their orbit period measures about 6 to 8 hours, providing slightly over 1 hour local visibility above the horizon for an observer on the Earth and handover from one to the next satellite is every 6 hours minimum. The MEO satellites are operated in a similar way to Big LEO systems providing global coverage. However, compared to a LEO system a MEO constellation can only be in circular orbit; Doppler effect and handover is less frequent; propagation delay is about 70 ms and free space loss is greater; satellites are affected by radiation damages from the Inner

Van Allen Belt only during the launching period; fewer eclipse cycles means that battery lifetime will be more than 7 years; cosmic radiation is lower, with subsequently longer life expectancy for the complete MEO configuration; higher average elevation angle from users to satellite minimizes probability of LOS blockage and higher RF output power required for both indoor and handheld terminals.

An example of MEO satellite constellation is ICO system in currently developing cycles as a former Inmarsat-P system (10 operational + 2 spare satellites in 2 inclined planes of  $45^\circ$  and at 10,355 km altitude) and the abandoned Odyssey system (12 + 3 satellites in 3 inclined planes at 10,355 km). Accordingly, a Non-GEO satellite system known as the ICO Global Communications network, which operates in MEO constellation, will provide MSC mobile telephony, Fax and data services, including Internet access and was scheduled to be operational in 2004.

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 GSNN systems such as Navstar (GPS), Navsat, GLONASS and the newly developed Galileo. In all, complete implementation of this orbit configuration would have 24 satellites in 3 orbital planes equidistant from each other, at an altitude of 20,000 km and at an inclination of  $55^\circ$ . In comparison with existing GSNN the new Galileo system will have 30 satellites in high MEO of about 28,000 km and at a similar inclination of  $56^\circ$ . At this point, its interest to polar MSC would be the eventual prospect of satellite sharing with navigation services, in a similar fashion to a high PEO with minimum of 3 satellites in the same orbital plane.

### 2.3.2.2. Geostationary Earth Orbit (GEO)

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

The footprint or service area of a GEO satellite covers almost 1/3 of the Earth's surface or  $120^\circ$  in longitude direction and up to  $75^\circ$ – $78^\circ$  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  $120^\circ$ , although the best solution is to employ four GEO satellites for better overlapping. This type of orbit is essentially used for commercial communication services for both FSS and MSS with the following advantages:

**a)** The satellite remains stationary with respect to one point on the Earth's surface and so the LES antennas can be beamed exactly towards the focus of the GEO satellite without periodical tracking. Only mobile high-gain antennas need auto-tracking systems, while low-gain omnidirectional antennas are free of tracking systems.

**b)** The new Inmarsat GEO space constellation consisting in four satellites can cover all three-ocean regions with four overlapping longitudes, except for the polar regions beyond latitudes of  $75^\circ$  North and South. Otherwise, the polar regions can be covered for maritime and aeronautical MSS applications with current HF Radio systems or in combination with PEO or HEO satellite constellations.

c) The Doppler shift, affecting synchronous digital systems caused by satellites to drift in orbit (affected by the gravitation of the Moon and to a lesser extent of the Sun) is small for all LES and MES within satellite coverage.

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

a) The long signal delay is due to the large distance of about 35,800 km if the satellite is in zenith for MES and about 41,000 km at the minimum elevation angle of about  $5^\circ$ . For the EM waves traveling at the speed of light this causes a round-trip signal delay of 240 to 270 ms and full duplex delay of 480 to 540 ms. Thus, the voice used via satellite can experience some disturbance but echo cancellation devices developed in the 1980s can reduce the problem. Besides, for data transmission equipment, especially when using error-correction protocols that require retransmission of blocks with detected errors, complex circuitry with special high-capacity buffer devices is required to overcome delay problems. In addition, practical experience has shown that given good control of the echo, a telephone connection which includes one hop in each direction via a GEO satellite is acceptable to public users.

b) The required higher RF output power and the use of directional antennas aggravate GEO operation slightly for use with handheld terminals, although it is not critical, because some GEO operators provide this service, such as Thuraya and ACeS.

c) The launch procedure to put a satellite in GEO is expensive but the total cost of 4 satellites is less than the cost of a minimum of 12 or 40 for MEO and LEO, respectively.

As stated earlier, the major disadvantage of a GEO satellite in a voice transmission is the round-trip delay between satellite and LES of approximately 2.5 sec, which can be successfully solved with current and newly advertised echo cancellation circuits. Because of the enormous use of the GEO constellation for many space applications, some parts of the GEO are becoming congested, owing to only one radius and latitude. This orbit is geostationary and so its track is one point called the sub-satellite point and obviously, handover and Doppler effect does not apply to GEO. A GEO satellite is at essentially fixed latitude and longitude, so even a narrow-beam Earth antenna can remain fixed. Satellites in GEO can use high and recently low-gain antennas, which helps to overcome the great distances in achieving the required EIRP at ground level. On the other hand, using satellite spot beam antennas, coverage can be confined to smaller spot areas, bigger power and higher speed of transmission, such as new generation of Inmarsat-3 spacecraft.

Furthermore, a variety of perturbing forces causes the GEO satellite to drift out of its path and assigned position towards so-called inclined orbit (GIO). By far the most important perturbations are the lunar and at a lesser degree the solar gravitational forces, which cause the satellite to drift in latitude or North–South direction. However, the longitudinal drift in East–West direction is caused by fluctuations in the gravitational forces from the Earth, due to its nonspherical shape and by fluctuations in solar radiation pressure. Thus, to counteract these perturbations the satellite needs station-keeping devices. The GEO satellites pass through both Van Allen Belts only on launch, so their effect is insignificant. After reaching the end of operational life a satellite has to be removed from its orbital slots into a graveyard orbit some 200 km above the GEO plane.

Thus, the GEO satellite constellation seems likely to continue to dominate in the satellite communications world, especially in MSS, providing near global coverage with low and high-power transmission. Especially attractive is the reliable and economical use of the Inmarsat standard-C low-power transceiver and low-gain omnidirectional antenna for maritime, land vehicles and aeronautical two-way data/messaging and telex and one-way E-mail service.

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

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

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

### 2.3.2.3. Geosynchronous Inclined Orbit (GIO)

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

Otherwise, a GIO satellite has a period of orbit equal to or very little different from a sidereal day (23h 56 min and 4.1 sec), which is time for one complete revolution of the Earth. The satellite movement speed has only very little difference from the angular velocity of the Earth, so this movement also has constant angular velocity. Otherwise, the projection of this movement on the equatorial plane is not at a constant velocity. There is an apparent movement of the satellite with respect to the reference meridian on the surface of the Earth and that of the satellite on passing through the nodes. The orbit may be inclined at any angle, which produces a repeating ground track. In **Figure 2.9. (B)** are 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. Sometimes there is no need to control inclination because tracking LES antennas are required for other reasons, while mobiles such as ships and aircraft require tracking antennas. Some GEO satellites may last beyond their planned lifetime if run low on fuel and cease inclination control. In effect, the GIO constellation with 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-hour period of visibility, which is especially required for continuous Euro-Asian regional coverage providing land MSC service. At this point, blocking of the beam due to occlusion of the satellite by buildings, mountains, hills and trees is minimized. 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 LES in both northern and southern polar regions are required to serve MES terminals. The LES tracking can be reached by a fairly directive fixed antenna while the satellite is in its slow apogee sector, the HEO space constellation is namely designed to cover the area under the apogee. Tracking of the satellite is facilitated on account of the small apparent movement and the long visibility duration. Otherwise, it is even possible to use antennas whose 3 dB bandwidth is a few tens of degrees, with fixed pointing 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 LES antenna, from the ground or due to interference from other radio systems and atmosphere, is also minimized due to the high elevation angles. At any rate, these advantages have led the former USSR to use these orbits for a long time in order to provide coverage of high latitude territories for mobile systems.

The HEO satellite two-way voice transmission has a similar delay as a GEO at the apogee of about 0.25 sec. Therefore, free space loss and propagation delay for HEO is comparable to that of the GEO constellation. Compared with GEO, the launch and satellite cost of the HEO constellation is reasonably low; this constellation is free of congestion because of only a few current and projected new HEO systems and provides high elevation angles for LES, which reduces atmospheric losses. Due to the relatively large movement of a satellite in HEO with respect to an observer on Earth, satellite systems using this model of orbit need to be able to cope with large Doppler shifts, 14 kHz for Molnya and 6 kHz for Tundra orbits in L-band 1.6 GHz. However, as the former USSR's experience has shown, satellites in this orbit tend to have rather a short lifetime due to the repetitive crossing of both Van Allen Belts. The rest of the disadvantages are the necessity of constant satellite tracking at the MES, compensation of signal loss variation, long eclipse periods and complex control system of MES and spacecraft.

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

When there is an orbit in HEO plane of 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 LES or MES located at high latitudes. Although the satellite remains for several hours in the vicinity of the apogee, it does move with respect to the Earth and after a time dependent on the position of the MES, the satellite disappears over the horizon as seen from the mobiles. However, to establish permanent links it is necessary to provide several suitably phased satellites in similar orbits, which are spaced around the Earth (with different right ascensions of the ascending node and regularly distributed between 0 and  $2\pi$ ) 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 Molnya, Tundra, Loopus, Borealis of the Ellipso system and Archimedes. The ESA proposed Archimedes system employs a so-called “M-HEO” 8-hour orbit. This produces three apogees spaced at  $120^\circ$ . In effect, 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. Molnya Orbit

The first prototype HEO Molnya 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 Molnya 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 spacebased 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 minutes. The laws of orbital mechanics dictate that the spacecraft orbital velocity is greatly reduced near apogee, allowing broad visibility of the Northern Hemisphere for periods up to eight hours at a time. Thus, by carefully spacing 3 or 4 Molniya spacecraft, continuous communications can be maintained. This type of orbit was pioneered by the USSR and is particularly suited to high latitude regions, which are difficult or impossible to service with GEO satellites.



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

The hypothetical Russian Molniya network can employ minimum 3 HEO satellites in three 12-hour orbits separated by  $120^\circ$  around the Earth, with apogee distance at 39,354 km and perigee at 1,000 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 hours. The characteristics of an example Molniya orbit are given in **Table 2.4**.

**Table 2.4.** Molniya and Tundra Orbit Parameters

Characteristics	Molniya Orbit	Tundra Orbit
Orbital Period ( $t$ )	12 h	24 h
Sidereal Period	11 h 58 min 2 s (half day)	23 h 56 m 4 s (full day)
Semi-major Axis ( $a$ )	26,556 km	42,164 km
Inclination ( $i$ )	$63.4^\circ$	$63.4^\circ$
Eccentricity ( $e$ )	0.6 to 0.75	0.25 to 0.4
Perigee Altitude ( $h_p$ ) (e.g.: $e=0.71$ )	$a(1 - e) - R$ 1,250 km	$a(1 - e) - R$ 25,231 km
Apogee Altitude ( $h_a$ ) (e.g.: $e=0.71$ )	$a(1 + e) - R$ 39,105 km	$a(1 + e) - R$ 46,340 km

The only one-track cycles of a total of two satellite tracks on the surface of the Earth is shown in **Figure 2.9. (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  $i = 65^\circ$ , 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 orbit 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 hours per day, while with four satellites handover is every 6 hours. The LES must use tracking antenna systems, so a terminal with only one antenna will have an outage during handover (switching) from one satellite to another.

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

### 2.3.3.2. Tundra Orbit

The Russian Tundra HEO system employs 2 satellites in two 24-hour 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 hours 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 hours. The characteristics of an example orbit of this type are given in **Table 2.4**. This orbit has only one track on the Earth's surface, as shown in **Figure 2.9. (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 figure 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-hour orbits separated by  $120^\circ$  around the Earth, has an apogee distance at 39,117 km and perigee at 1,238 km. This orbit has similar advantages and disadvantages as for the Molniya orbit. One of the problems encountered by the LES is that of repointing the antenna during the handover (changeover) from one satellite to another. With orbits whose track contains a loop, it is possible to use only the loop as the useful part of the track in the trajectory. Handover between two satellites is performed at the crossover point of the track. At this instant the two satellites are seen from the LES in exactly the same direction and it is not necessary to repoint the antenna. To achieve continuous coverage of the region situated under the loop, the transmit time of the loop must be a sub-multiple of the orbit period and the number of satellites. Hence, the coverage can be extended to one part of the hemisphere by increasing the number of satellites in orbit regularly spaced about the globe.

### 2.3.4. Polar Earth Orbits (PEO)

The PEO constellation is today a synonym for providing coverage of both polar regions for different types of meteorological observation and satellite determination services. Namely, a satellite in this orbit travels its course over the geographical North and South Poles and will effectively follow a line of longitude. Certainly, this orbit may be virtually circular or elliptical depending upon requirements of the program and is inclined at about  $90^\circ$  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, for instance, two such satellite orbits are spaced at  $90^\circ$  to each other, the time between satellites passes over any given point will be halved.

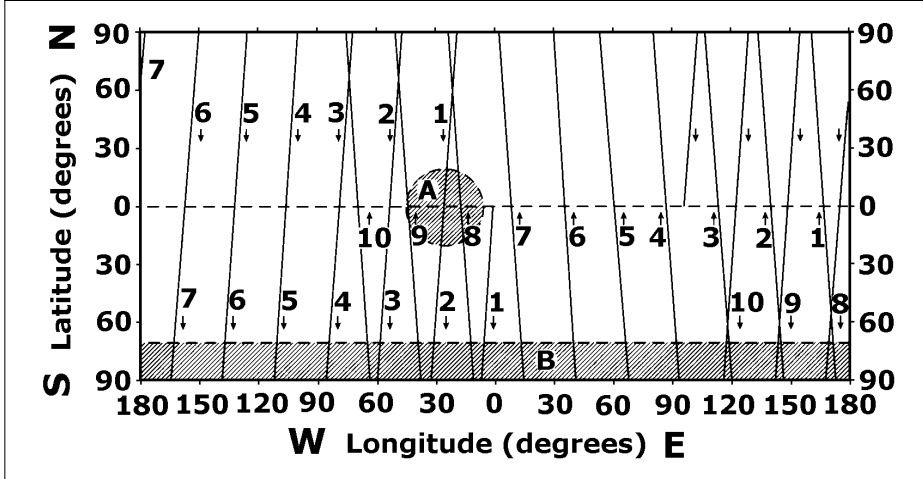
The PEO system is rarely used for communication purposes because the satellite is in view of a specific point on the Earth's surface for only a short period of time. Any complex steerable antenna systems would also need to follow the satellite as it passes overhead. At any rate, this satellite orbit 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 1,400 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 1,000 km, reaching a peak at about 5,000 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 a way between MEO and GEO satellite planes.

These two specific systems studied by Inmarsat are Cospas-Sarsat Low PEO at 1,000 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 Molnya constellation of four satellites; GIO  $45^\circ$  inclined orbit of four satellites and  $55^\circ$  inclined circular MEO at 20,000 km altitude for GPS and GLONASS satellite navigation systems. In the meantime, the Cospas-Sarsat system has developed a special GEOSAR system using three GEO satellites for global distress communications satellite beacons in combination with already developed LEOSAR systems using four PEO satellites.

**Figure 2.11.** Type of Satellite Orbits and Tracks

Courtesy of Book: "Commercial Satellite Communication" by S.C. Pascall and D.J. Withers

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

#### 2.3.4.1. Low PEO

The Low PEO satellite constellation similar to the LEO mostly employs both polar and near-polar orbits for communications and navigation utilities. Thus, a particular example of a system that uses this type of orbit is the Cospas-Sarsat SAR system for maritime, land and aeronautical applications. This system uses 8 satellites in 4 near-polar orbits: four US-based Sarsat satellite constellations at 860 km orbits, inclined at  $99^\circ$ , which makes them sun-synchronous and four Cospas satellite configurations at 1,000 km orbits, inclined at  $82^\circ$ . However, this orbit was also suitable for the first satellite navigation systems Transit and Cicada, developed by the USA and the former USSR, respectively.

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

In **Figure 2.11**, is illustrated the Earth track of ten successive orbits of satellite in Low PEO with an altitude of 1,000 km. The MES in shaded area A (4,200 km in diameter) would see the satellite, in the absence of environmental screening, at an angle of elevation not less than  $10^\circ$ , 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.

The South Pole coverage area at a single pass of the satellite is shown 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  $90^\circ$  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  $50^\circ$  would have better visibility between  $60^\circ$  N and  $60^\circ$  S latitude than a PEO satellite but it would have no visibility at all of the polar regions. The Low PEO configuration is attractive for mobile distress communications for two reasons. First, the transmission path loss is relatively low, allowing reliable communication with a low powered satellite beacon and PEO spacecraft. An altitude of about 1,000 km is the upper limit for good reception of signals at 243/406 MHz sent from emergency distress beacons. Secondly, the Doppler shift is high, approximately 30 kHz at 1.6 GHz, allowing accurate location of the distress transmitter. On the other hand, there are several significant disadvantages. However, as mentioned earlier, PEO coverage is not continuous unless there is simultaneous communication between a distress buoy and a ground terminal because of the small footprint of each individual satellite. Accordingly, storage and retransmission of distress messages on-board processing would be necessarily adding to the distress alert delay time and also to satellite mass and complexity.

The short visibility period during a transit and the uneconomic need for large numbers of satellites for continuous coverage makes a Low PEO unattractive for communications considerations. If this orbit configured well as an economic solution for distress coverage in polar regions to be used for communications purposes, users would have to operate with the following restrictions: (1) Only burst mode, non-simultaneous data communication would be possible; (2) Transmission time and/or bit rate would be limited by satellite message storage capability; (3) Replies to the message would require an interrogation or polling system from the MES expecting a reply; (4) Depending on the PEO constellation and MES position, a reply could take some hours.

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

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

#### 2.3.4.2. High PEO

The High PEO constellation would consist of three satellites separated by  $120^\circ$  in the same circular orbit of 12,000 km altitude, geometrically similar to the GEO and as orbit similar to MEO configuration. This orbit provides continuous coverage to all polar regions above  $59^\circ$  latitude. Thus, six satellites (in two orbital planes of three satellites each) would provide continuous and real global coverage if that were required, which GEO cannot obtain.

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

#### 2.3.5. Hybrid Satellite Orbits (HSO)

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

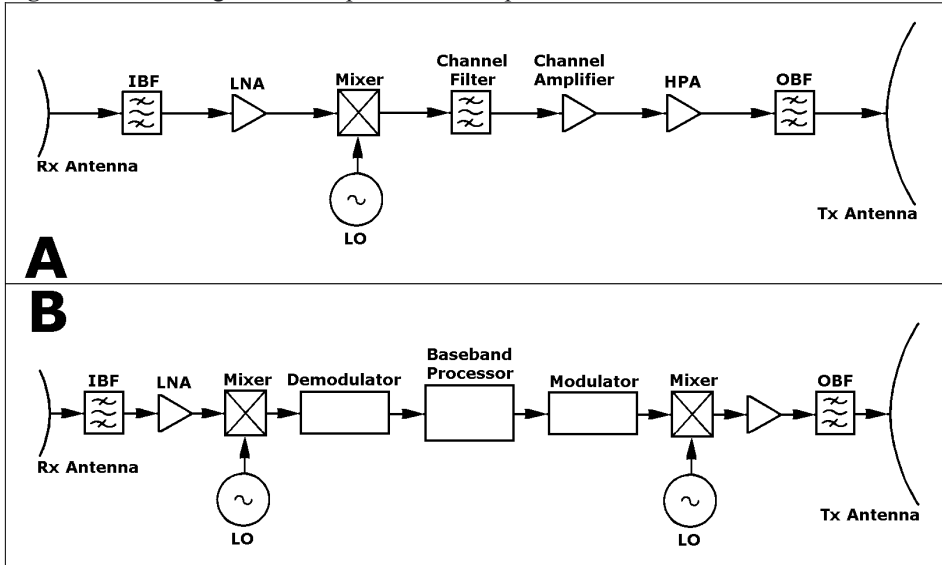
**1) Combination of GEO and HEO Constellations** – The development of a MSS which would provide reliable communications with MES, such as vessels, road vehicles, trains and aircraft, rural areas and remote terminals. This MSC system, called Marathon, includes five GEO Arcos-type satellites and four Mayak-type satellites in a HEO, as well as a ground segment that is composed of base stations and terminals installed at fixed or mobile users premises. Therefore, the combination of GEO and Non-GEO satellite constellations makes it possible to render GMSC services, including those at high latitudes and in the both polar areas, this is especially important for Russia, with its vast northern Eurasian territories and to provide the most reliable satellite communication between the territories of the western and Eastern Hemispheres. This hybrid constellation can be useful as well as for the Alaska, Greenland and northern territory of Canada.

**2) Combination of GEO and PEO Constellations** – This current combination of orbits has been developed by the efforts of the Cospas-Sarsat organization, with the assistance of IMO, Inmarsat and other international and regional contributors. At the other words, the Cospas-Sarsat space segment is a combination of three GEO operational satellites of the subsystem called GEOSAR and four PEO operational satellites of the subsystem called LEOSAR, with spare spacecraft for all participants. The GEOSAR employs one satellite type of INSAT-2A and two GOES type GOES-E and GOES-W, while the LEOSAR configuration has two satellites supplied by Cospas and two by Sarsat. Otherwise, the GEOSAR project in the future has to include the European MSC and two Russian Luch-M spacecraft. This system is responsible for providing distress alert and to help SAR forces on-scene determinations for maritime, land and aeronautical applications.





**Figure 2.13.** Configuration of Spacecraft Transponders



Courtesy of Handbook: "Mobile-Satellite Service" by ITU

### 2.4.1. Satellite Payload

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

#### 2.4.1.1. Satellite Repeaters

The function of a repeater is to receive the uplink RF signals from either ground segment service or feeder links, to convert these signals to the appropriate downlink frequency and power and to retransmit them towards the service or feeder links ground segment. Two types of repeater architecture are possible in on-board utilization of communication satellites: Transparent or Bent-pipe, and Regenerative transponders.

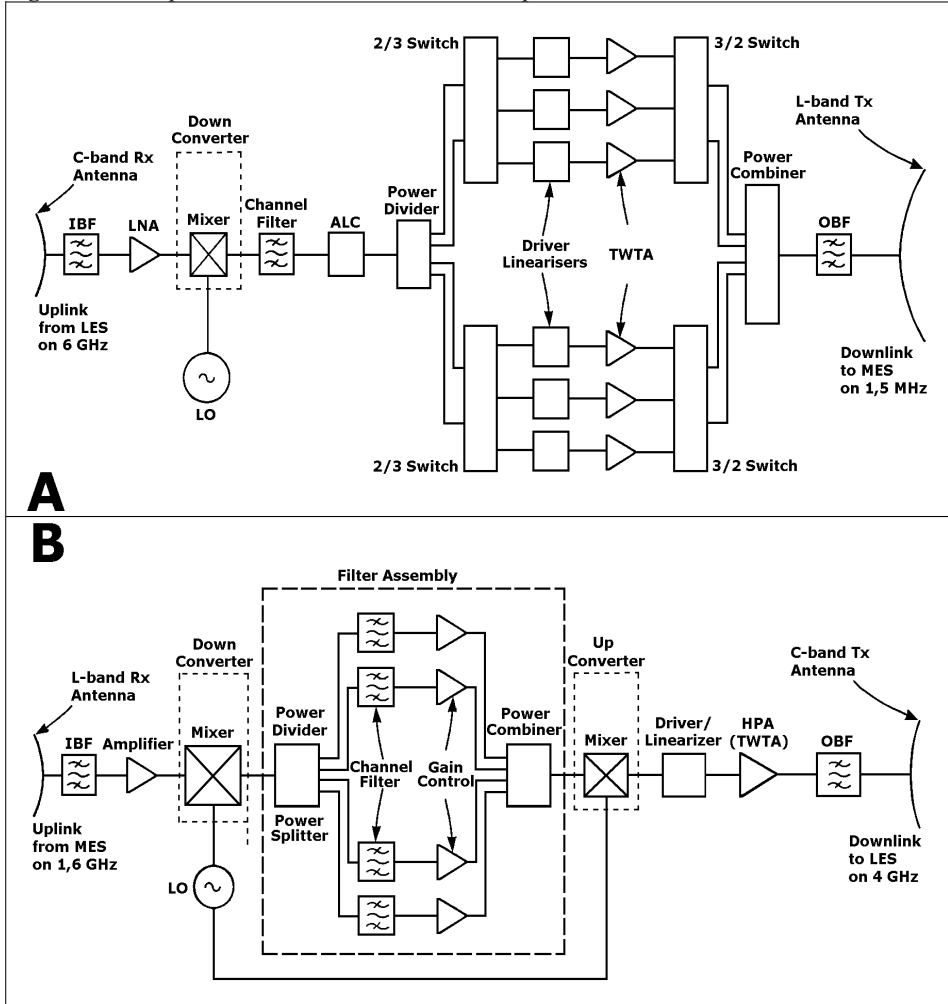
**1. Transparent Transponder** – The basic function of the satellite transponder is to isolate individual carriers or groups of carriers of signals and to boost their power level before they are retransmitted to the ground stations. The carrier frequencies are also altered as the carriers pass through the satellite. Satellite repeaters that process the carrier in this way are typically referred to as transparent or bent-pipe transponders, shown in **Figure 2.13. (A)**. Only the basic RF characteristics of the carrier (amplitude and frequency) are altered by the satellite. However, the detailed signal carrier format, such as the modulation characteristics and the spectral shape, remains completely unchanged. In such a way, 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. Therefore, 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 LES or MES by Rx antenna are filtered in an Input Bandpass Filter (IBF) prior to amplification in a Low Noise Amplifier (LNA). 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 Tx antenna. 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 frequency, using an LO which is controlled to provide a very stable frequency source. This is needed to reduce all noises to facilitate processing of the incoming signal and to enable the downlink frequencies to be established. The channel filter isolates the various communications channels contained in the waveband allowed through by the input filter. Filtering often leads to large power losses, creating a need for extra amplification, usually followed by a main amplifier. In order to attain the required gain of HPA this segment may employ either a Solid-State Power Amplifier (SSPA) or a Traveling Wave Tube Amplifier (TWTa). In a more complex design, in order to achieve higher RF power, it may be possible to combine the output of several amplifiers. To do this the incoming signal must be divided in such a way so as to provide separate identical input to each amplifier, see 6 TWTa presented in **Figure 2.14. (A)**. A power combiner then recombines the RF signals from the amplifiers to produce a single RF output. The output filter removes all unwanted signals from the transmitted downlink returning to the Earth stations. High reliability throughout the lifetime of the satellite is achieved by duplicating critical units in the receiver, such as TWTa, etc.

**2. Regenerative Transponder** – Other satellite system designs go through a more complex 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 MES and handheld user terminals. A typical on-board processing system will implement some or all of the functions that are performed by the ground-based transmitter and/or receiver in a transparent satellite system. Therefore, these functions may include recovery of the original information on board the satellite and the processing of this information into a different carrier format for transmission to the ground stations. In fact, any satellite transponder that recreates the signals carrier in this way is usually referred to as a regenerative transponder, illustrated in **Figure 2.13. (B)**. 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 MES Rx, while the bent-pipe system requires a satellite link design for the entire link, involving both uplink and downlink, but the forward link burst rate is limited by the MES G/T and demodulation performance.

**Figure 2.14.** Spacecraft C/L and L/C-band Transponders

Courtesy of Book: "Never Beyond Reach" by B. Gallagher

Advanced MSS include intersatellite links to establish a direct connection between satellite transponders and in this way to enlarge system coverage and help the reception of signals from other GEO satellite coverage not visible for particular LES and MES terminals. Moreover, an intersatellite link can help to solve the problems associated with some TTN infrastructures or to reduce landline charges. In the same way, intersatellite links can also provide a connection from a satellite to neighboring satellites in a constellation of Non-GEO space segment and so, they are beneficial in reducing the number of satellite hops when Earth coverage of each satellite is limited. Intersatellite links are usually implemented with regenerative satellite transponder systems, mainly due to the flexible connection to intersatellite links.

#### 2.4.1.2. Satellite Transponders for MSC

A transponder for MSC systems cannot be a Mobile Satellite Transponder, which has been determined by some authors; because a transponder cannot be de facto mobile but serves MSS or even FSS. It is, however, an electronic segment made up of repeaters (receiver and transmitter) on board the satellite. The Inmarsat-2 payload consists in two transponders: the C/L-band, shown in **Figure 2.14. (A)** and the L/C-band, shown in **Figure 2.14. (B)**.

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

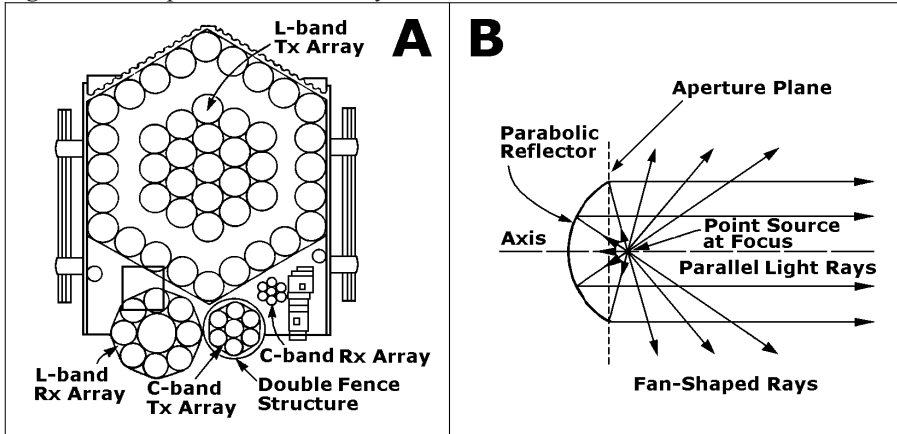
**b) Inmarsat-2 L/C-band Transponder** – This transponder receives uplink signals in the L-band of 1.6 GHz from MES and retransmits downlink signals in the C-band of 3.6 GHz to LES, after frequency conversion and signal amplification by the HPA. The signals received by an L-band antenna are fed to a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

#### 2.4.1.3. Satellite Antenna System for MSC

The antenna array system of Inmarsat-2 satellite for MSC is shown in **Figure 2.15 (A)**. The satellite antenna system mounted on the spacecraft structure, similar to the transponders, is composed of two main integrated elements: the C/L-band and the L/C-band antenna.

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

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

**Figure 2.15.** Spacecraft Antenna Systems

Courtesy of Book: "Never Beyond Reach" by B. Gallagher

#### 2.4.1.4. Characteristics of Satellite Antennas

Both transmit antenna array systems are providing a global (wide) footprint on the Earth's surface. However, narrow circular beams from GEO or Non-GEO can be used to provide spot beam coverage. For instance, from GEO the Earth subtends an angle of  $17.4^\circ$ . Antenna beams  $5.8^\circ$  wide can reuse three frequency bands twice in providing Earth disc coverage. The directional properties of antenna arrays can be exploited to permit RF reuse in space communications, which is similar to several radio stations using the same RF being geographically far apart. Earth coverage by seven spot beams (six spots are set out around one spot in the centre) can be arranged by three pairs of beams: 1 and 4, 2 and 5 and 3 and 6, operating on frequencies  $f_2$ ,  $f_3$  and  $f_4$ , respectively. Mutual interference within pairs is avoided by pointing one beam as far away from the other as possible. Coverage of the centre of the disc is provided by a single beam operating on frequency  $f_1$ . The main advantage with this spot footprint that is specific Earth areas can be covered more accurately than with wide beams. Furthermore, a greater power density per unit area for a given input power can be achieved very well, when compared with that produced by a global circular beam, leading to the use of much smaller receiving MES antennas. The equation that determines received power ( $P_R$ ) is proportional to the power transmitted ( $P_T$ ) separated by a distance ( $R$ ), with gain of transmit antenna ( $G_T$ ) and effective area of receiving antenna ( $A_R$ ) and inverse proportional with  $4\pi$  and square of distance. The relations for  $P_R$  and  $G_T$  are presented as follows:

$$P_R = P_T G_T A_R / 4\pi R^2 \quad (2.42.)$$

$$G_T = 4\pi A_T / \lambda^2$$

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:

$$P_R = P_T G_T G_R \lambda^2 / (4\pi R)^2 \quad \text{or} \quad P_R = P_T G_T G_R / L_P L_K \quad [\text{W}] \quad (2.43.)$$

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

$$P_R/N = P_T G_T (G_R/T_R) (1/K L_P L_K) \quad (2.44.)$$

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 LES/MES are minimized. Similarly, on the satellite-to-Earth link, the higher the gain of the satellite transmit antenna, the greater the EIRP for a given transmitter power. Satellites often have parabolic dish antennas, though there are also other types, such as phased arrays. The principal property of a parabolic reflector is its ability to turn light from a point source placed at its focus into a parallel beam, mostly as illustrated in **Figure 2.15 (B)**. 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:

$$G_a = \eta (4\pi A/\lambda^2) = 4\pi A_E/\lambda^2 \quad (2.45.)$$

$$G_p = \eta (\pi D)^2 / \lambda^2$$

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:

$$G_p = \eta (\pi D f/c)^2 = 60.7 (D f)^2 \quad (2.46.)$$

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:

$$G_T = 10 \log G_p. \quad (2.47.)$$

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:

$$F(\delta_o) = s(\delta_o)/s(\delta_o=0) \quad (2.48.)$$

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 **Figure 2.2. (A)** follows the relation:

$$F(\delta_o) = d_o/h = \cos \delta \sqrt{(k^2 - \sin^2 \delta_o)} / 1 - k \quad (2.49.)$$

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

$$F(\delta) = k \cos \delta \quad (2.50.)$$

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:

$$F(\delta) = \Delta L = 20 \log = 20 \log R/(R + h) \cos \delta = 10 \log R/(1 + 2R/h) \text{ [dB]} \quad (2.51.)$$

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:

$$L_P = (4\pi d/\lambda)^2 \quad (2.52.)$$

$$P_R/S = P_T G_T / 4\pi d^2 L_K$$

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

### 2.4.2. Satellite Bus

The satellite bus is usually called a platform and consists in several sections, shown in **Figure 2.12**. The function of the satellite platform is to support the payload operation reliably throughout the mission of primary construction section, such as Structure Platform (SP), Electric Power (EP), Thermal Control (TC), Attitude and Orbit Control (AOC), Telemetry, Tracking and Command (TT&C) and Propulsion Engine.

#### 2.4.2.1. 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.

Most satellites are either cylindrically-shaped and are stabilized by spinning the whole of main body, or box shaped and three-axis body stabilized. Spin stabilized structures have a cylindrical part, which rotates at a speed of 50–100 rpm and the despun stabilized part has mounted antennas always facing to the Earth. The spinning part of the cylinder is covered with solar cells and its spin axis is oriented perpendicularly to the Sun. Body stabilized structures rotate once for every rotation of the Earth, so that the side with mounted antenna will always face the Earth. This platform utilizes a deployed set of solar panels with solar cells mounted on one side of the panel surface relative to the Sun.

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

#### **2.4.2.2. Electric Power (EP)**

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

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



The operational status of batteries including recharge, in-service or reconditioning is remotely controlled by a special ground segment. The mass of a battery constitutes a significant portion of the total satellite mass. Therefore, a useful figure of merit to evaluate the performance of a battery is capacity in W/h per unit weight taken at the end of its life. Until recently virtually all satellites used Ni-Cd (Nickel-Cadmium) batteries because of their high reliability and long lifetime. These batteries provide a low specific energy of about 30 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 hours a day, with about 4 eclipses per day. Otherwise, the Sun can also be sometimes eclipsed by the Moon's shadow, which is less predictable.

#### **2.4.2.3. Thermal Control (TC)**

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. Thermal stress 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. 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. All these extremes 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.

#### 2.4.2.4. 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 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: (1) Measuring the attitude of the satellite by sensors; (2) Comparing the results of measurements with the required values; (3) Calculating the corrections to reduce eventual errors and (4) Introducing these corrections by operating the appropriate torque units.

**a) 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.

**b) 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 LES. When the orbit and position of the satellite have been determined, it is possible to calculate the velocity increments required to keep the N-S and E-W excursion of the satellite within the tolerated limits. The frequency with which N-S correction must be made 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 to 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.

#### 2.4.2.5. 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 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 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 SCC on Earth. 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 LES in the network, it is necessary to estimate the orbital parameters regularly. These parameters can be obtained by tracking the communications satellite from the ground and measuring its angular position and range. Most SCC employ angular and range or range-rate tracking to control satellite orbits.

**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: Satellite transponder and beacon switching, Antenna pointing control, Switch matrix reconfiguration, Controlling direction and speed of solar arrays drive, Battery reconditioning, Thruster firing and Switching heaters of the various systems.

#### 2.4.2.6. 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 satellites 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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