جامعة بغداد

كلية العلوم

قسم الفلك والفضاء

محاضرات مادة الاقمار الصناعية (Satellites II)

المرحلة الرابعة / الفصل الثاني

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Satellites II **المفردات**

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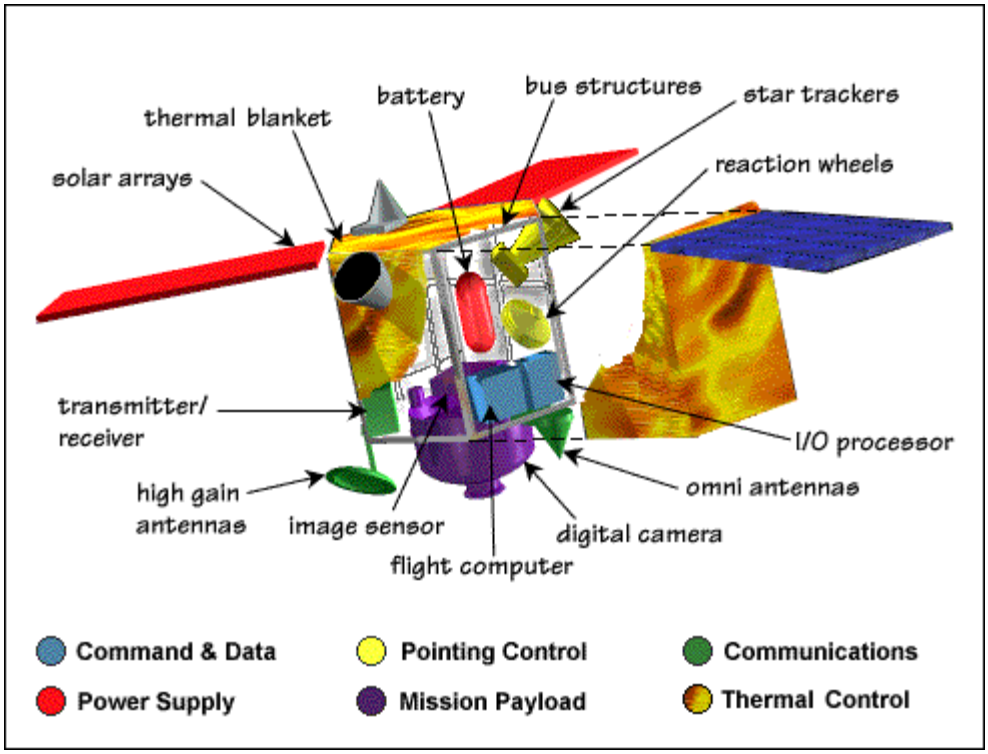
1. Satellite Hardware

Satellite Subsystems

Irrespective of the intended application, be it a communications satellite or a weather satellite or even an Earth observation satellite, different subsystems comprising a typical satellite Include the following:

1. Mechanical structure
2. Propulsion
3. Thermal control
4. Power supply
5. Tracking, telemetry and command
6. Attitude and orbit control
7. Payload
8. Antennas



1. Mechanical structure

The *structural subsystem* provides the framework for mounting other subsystems of the satellite and an interface between the satellite and the launch vehicle.

2. Propulsion

The *propulsion subsystem is* used to provide the thrusts required to impart the necessary velocity changes to execute all the manoeuvres during the lifetime of the satellite. This would include major manoeuvres required to move the satellite from its transfer orbit to the geostationary orbit in the case of geostationary satellites and also the smaller manoeuvres needed throughout the lifespan of the satellite, such as those required for station keeping.

3. Thermal control

The *thermal control subsystem* is essential to maintain the satellite platform within its operating temperature limits for the type of equipment on board the satellite. It also ensures a reasonable temperature distribution throughout the satellite structure, which is essential to retain dimensional stability and maintain the alignment of certain critical equipments.

4. Power supply

The primary function of the *power supply subsystem* is to collect the solar energy, transform it to electrical power with the help of arrays of solar cells, and distribute electrical power to other components and subsystems of the satellite. In addition, the satellite also has batteries, which provide standby electrical power during eclipse periods, during other emergencies and during the launch phase of the satellite when the solar arrays are not yet functional.

5. Tracking, telemetry and command

The *telemetry, tracking, and command* (IT&*C) subsystem* monitors and controls the satellite right from the lift-off stage to the end of its operational life in space. The tracking part of the subsystem determines the position of the spacecraft and follows its travel using angle, range, and velocity information. The telemetry part gathers information on the health of various subsystems of the satellite encodes this information and then transmits it. The command element receives and executes remote control commands to effect changes to the platform functions, configuration, position, and velocity.

6. Attitude and orbit control

The *attitude and orbit control subsystem* performs two primary functions. It controls the orbital path, which is required to ensure that the satellite is in the correct location in space to provide the intended services. It also provides attitude control, which is essential to prevent the satellite from tumbling in space and also to ensure that the antennae remain pointed at a fixed point on the Earth's surface.

7. Payload

The payload subsystem is that part of the satellite that carries the desired instrumentation required for performing its intended function and is therefore the most important subsystem of any satellite. The nature of the payload on any satellite depends upon its mission. The basic payload in the case of a communication satellite is the transponder, which acts as a receiver, amplifier, and transmitter. In the case of a weather forecasting satellite, a radiometer is the most important payload. High-resolution cameras, multispectral scanners, and thematic mappers are the main payloads on board a remote sensing satellite. Scientific

Satellites have a variety of payloads depending upon the mission. These include telescopes, spectrographs, plasma detectors, magnetometers, spectrometers and so on.

8. Antennas

Antennas are used for both receiving signals from ground stations as well as for transmitting signals towards them. There is a variety of antennas available for use on board a satellite. The final choice depends mainly upon the frequency of operation and required gain. Typical antenna types used on satellites include home antennas, center-fed and offset-fed parabolic reflectors and lens antennas.

Mechanical Structure

The mechanical structure weighs between 7 and 10 % of the total· mass of the satellite at launch. It performs three main functions namely:

1. It links the satellite to the launcher and thus acts as an interface between the two.
2. It acts as a support for all the electronic equipment's carried by the satellite.
3. It serves as a protective screen against energetic radiation, dust and micrometeorites in space.

The structure of Intelsat-5 weighs 140kg only, though the total mass of the satellite is greater than 1000 kg.

Example: (مثال للاطلاع معلومات قمر صناعي)

Table1 shows the information's of Intelsat-5 telecommunications satellite.

Mission type	Communication
Operator	Intelsat
COSPAR ID	1982
SATCAT no.	13595

Mission duration	17 years
Spacecraft properties	
Bus	Intelsat-V bus
Manufacturer	Ford Aerospace
Launch mass	1,928.2 kilograms
BOL mass	1,012 kilograms
Start of mission	
Launch date	September 28, 1982,
Rocket	Atlas SLV-3D Centaur-D1AR
Launch site	Cape Canaveral LC-36B (USA)
End of mission	
Disposal	Decommissioned
Deactivated	August, 1999
Orbital parameters	
Reference system	Geocentric
Regime	Geostationary
Longitude	147.4° W
Semi-major axis	42,695 kilometers
Perigee altitude	36,223.4 kilometers
Apogee altitude	36,426.3 kilometers
Inclination	15.4 degrees
Period	1,463.3 minutes
Epoch	April 23, 2017
Transponders	
Band	21 C-band

2. Satellite Launch

Fundamental issues such as laws governing motion of artificial satellites around Earth different orbital parameters, types. of orbits and their suitability for a given application, etc., related to orbital dynamics were addressed in the previous chapter. The next obvious step is to understand the launch requirements to acquire the desired orbit. This should then lead to various in-orbit operations such as orbit stabilization, orbit correction and station keeping that is necessary for keeping the satellite in the desired orbit.

expendable launch vehicles

1. Types of Launch Sequence

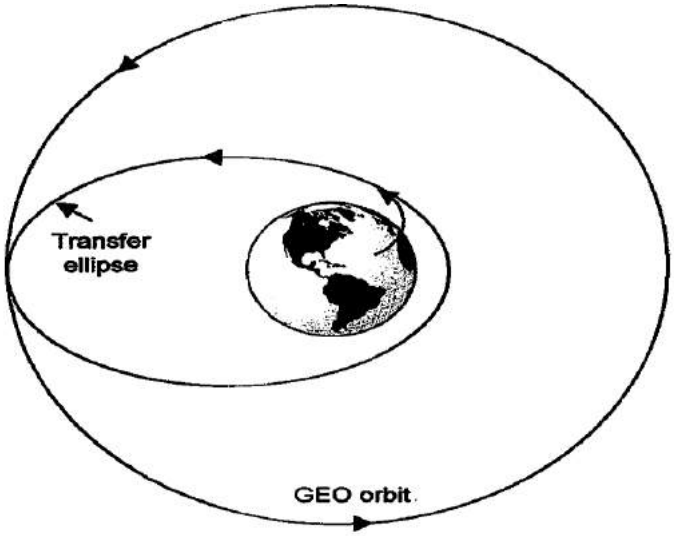
There are two broad categories of launch sequence,
one that is employed by expendable launch vehicles such as Ariane of the European Space Agency and Atlas Centaur and Thor Delta of the United States.
and the other that is employed by a re-usable launch vehicle such as the Space Shuttle of the United States and the recent development Buran of Russia .. Irrespective of whether a satellite is launched by a re-usable launch vehicle like the Space Shuttle or an expendable vehicle like Ariane, the satellite heading for a geostationary orbit is first placed in a transfer orbit.

Space Shuttle

The transfer orbit is elliptical in shape with its perigee at an altitude between 200 km and 300 km and its apogee at the geostationary altitude.

In some cases, the launch vehicle injects the satellite directly into a transfer orbit of this type. Following this, an apogee maneuver circularizes the orbit at the geostationary altitude.

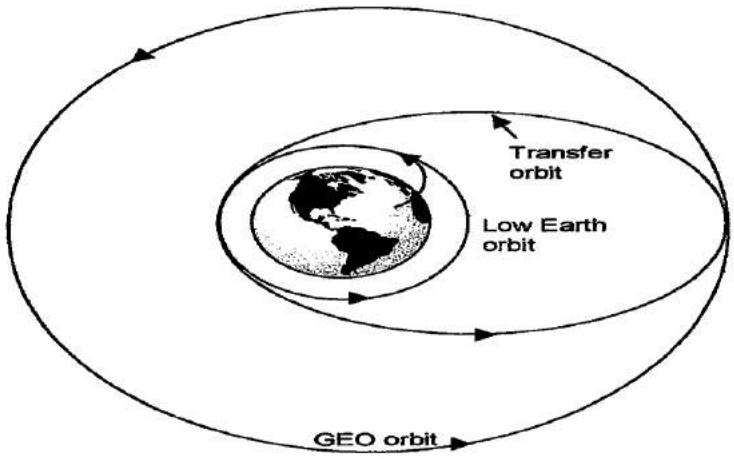
The last step is then-to correct the orbit for its inclination. This type of launch sequence is illustrated in Fige 1.



Fige1 .Possible geostationary satellite launch sequences

In the second case, the satellite is first injected into a low Earth circular orbit. In the second step, the low Earth circular orbit is

transfer into an elliptical transfer orbit with a perigee manoeuvre. Circularization of the transfer orbit and then correction of the orbit inclination follow this. This type of sequence is illustrated in Fige **2**



Fige2 Possible geostationary satellite launch sequence

In the following paragraphs is a discussion concerning some typical launch sequences observed in the case of various launch vehicles used to deploy geostationary satellites from some of the prominent launch sites all over the world. The cases presented here include

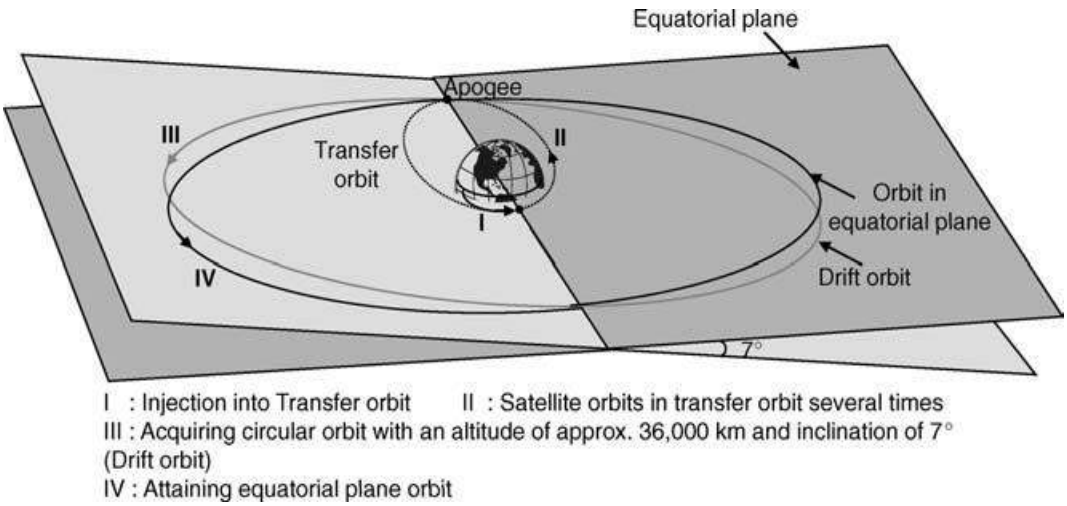
The launch of geostationary satellites from Kourou in French Guiana and Cape Canaveral in the United States, both situated towards the eastern coast of America, and Baikonur in Russia. A typical Space Shuttle launch will also be described. The two types of launch sequence briefly

1. Launch from Kourou

A typical Ariane launch of a geostationary satellite from Kourou in French Guiana will be illustrated. different steps involved in the entire process are:

1. The launch vehicle takes the satellite to a point that is intended to be the perigee of the transfer orbit, at a height of about 200 km above the surface of the Earth. The satellite along with its apogee boost motor is injected before the launch vehicle crosses the equatorial plane, as shown in Fige 3. The injection velocity is

such that the injected satellite attains an eccentric elliptical orbit with an apogee at about 36 000 km. The orbit is inclined at about 7°, which is expected, as the latitude of the launch site is 5.2°



Fige 3. Typical launch of a geostationary satellite from Kourou

2. In the second step, after the satellite has completed several revolutions in the transfer orbit, the apogee boost motor is fired during the passage of the satellite at the apogee point. The resulting thrust gradually circularizes the orbit. The orbit now is a circular
orbit with an altitude of 36000 km.
3. Further thrust is applied at the apogee point to bring the inclination to 0°, thus making the orbit a true circular and equatorial orbit.
4. The last step is to attain the correct longitude and attitude. This is also achieved by applying thrust either tangential by or normal to the orbit.

2. Launch from Cape Canaveral

Different steps involved in the process of the launch of a geostationary satellite from Cape Canaveral are:
1. The launch vehicle takes the satellite to a point that is intended to be the perigee of the transfer orbit, at a height of about 300 km above the surface of the Earth, and injects the satellite first into a circular orbit called the parking orbit. The orbit is inclined at an

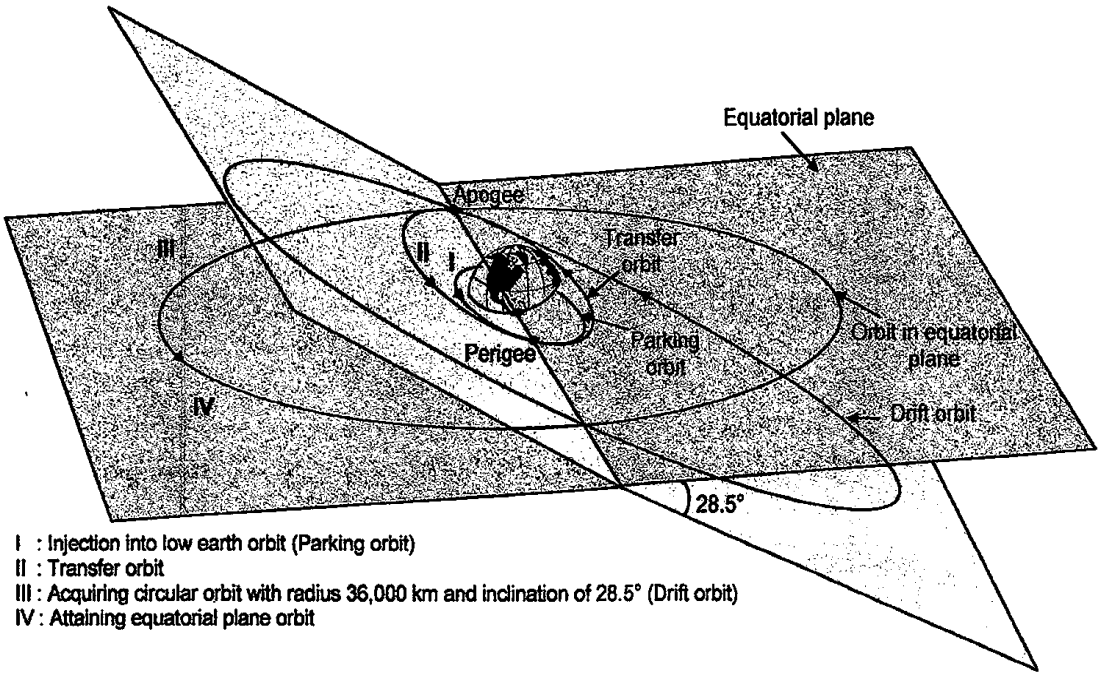
angle of *28,50* with the equatorial plane, as shown in Fige4. Reasons for this inclination angle have been explained earlier.

2. In the second step, a perigee maneuver associated with the firing of a perigee boost motor transforms the circular parking orbit to an eccentric elliptical transfer orbit with perigee and apogee distances of 300 k in and 36000km respectively.

3. In the third step, an apogee maneuver similar to the one used in the case of the Kourou launch circularizes the transfer orbit.

Till now, the orbit inclination is 28.5°. In another apogee maneuver, the orbit inclination is brought to 0°. The thrust required in this maneuvers is obviously much larger than that required in the case of inclination correction in the Kourou launch.

4. In the fourth and last step, several small maneuvers are used to put the satellite in the desired longitudinal position.

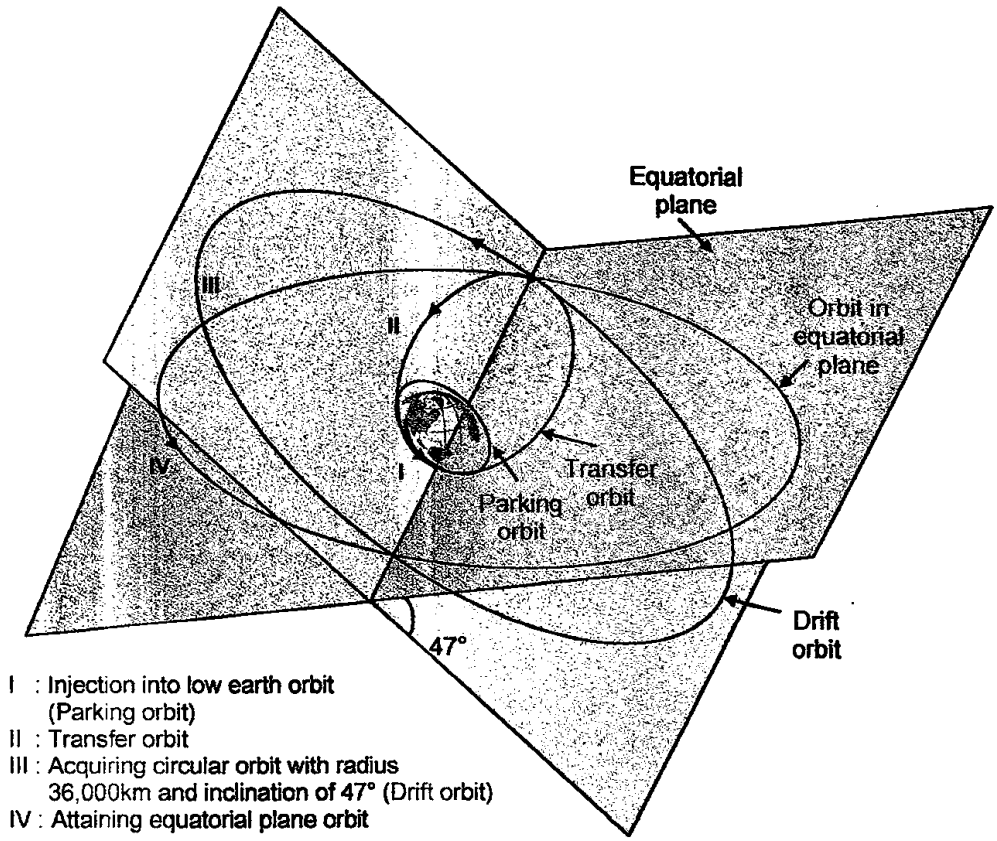


Fige 4. Typical launch of a geostationary satellite from Cape Canaveral

3. Launch from Baikonur

The launch procedure for a geostationary launch from Baikonur is similar to the one described in case of the launch from Cape Canaveral. Different steps involved in the process of the launch of a geostationary satellite from Baikonur are:

1. The launch vehicle injects the satellite in a circular orbit with an altitude of 200 km and "an inclination of 51°.
2. In the second step, during the first passage of the satellite through the intended perigee, a maneuver puts the satellite in the transfer orbit with an apogee of a little above 36000km. The orbit inclination is now 47°.
3. In the third step, the transfer orbit is circularized and the inclination corrected at the descending node.
4. In the fourth and last step, the satellite drifts to its final longitudinal position. Different steps as shown in Fige 5.



Fige 5.Typical launch of a geostationary satellite from Baikonur

H.W:

Q1: Explain the types of satellite launch sequence

Q2 Explain the lunch sequence satellite from Cape Canaveral in USA.

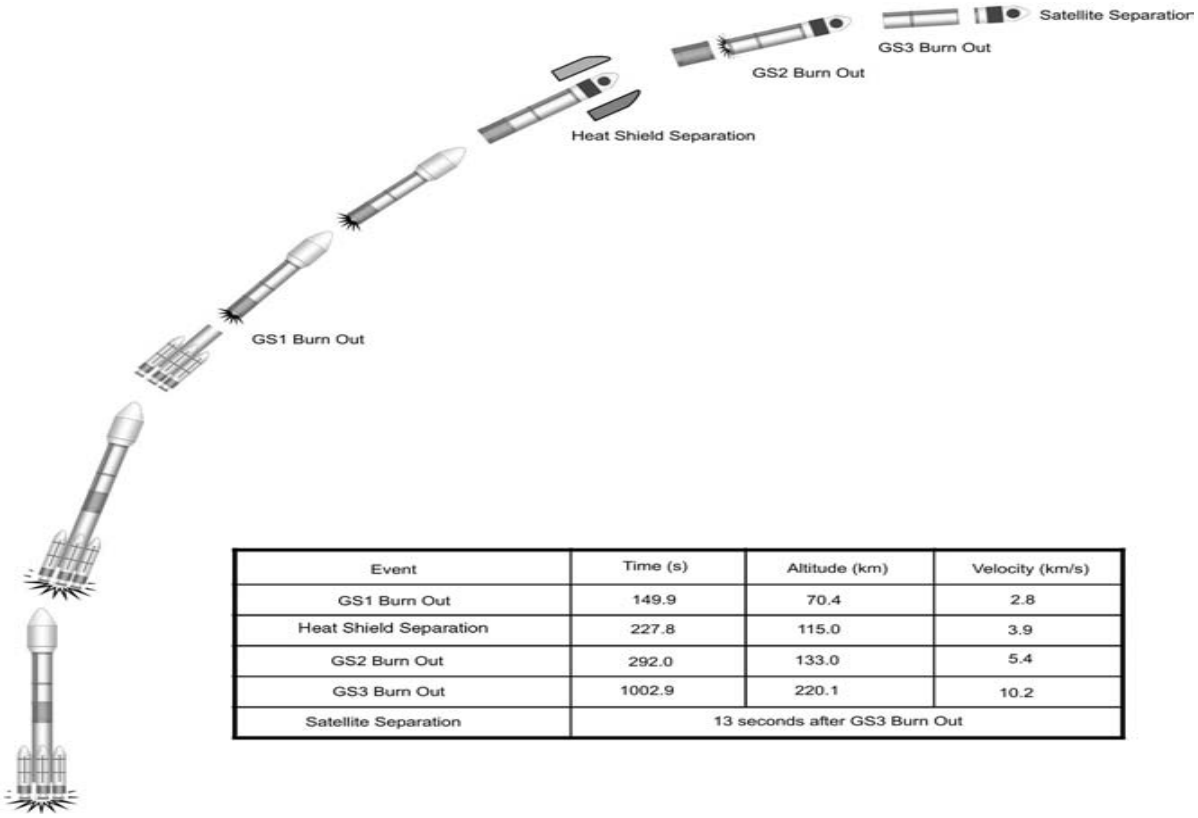
Q3: Explain the lunch sequence satellite from Kourou in French Guiana.

Q4: Explain the lunch sequence satellite from Baikonur.

3. Launch Vehicles

Launch vehicles are used to launch the satellites from the Earth into their desired orbits. In order to launch the satellite into its desired orbit, two parameters that are very important are the velocity vector and the orbital height. Launch vehicles can be further classified as:

Expendable Launch Vehicles (ELV) and Reusable Launch Vehicles (RLV). ELVs are designed to be used only once and their components are not recovered after launch. They mostly comprise of multi stage rockets and the job of each stage is to provide the desired orbital manoeuvre. As the job of the stage is completed, it is expended. The process goes on till the satellite is placed into the desired trajectory. The number of rocket stages can be as many as five. It may be mentioned here that some launch vehicles used to put the satellite into small LEO orbits can comprise of a single rocket stage. In addition to the rocket stages, launch vehicles also comprise of boosters that are used to aid the rockets during main orbital manoeuvres or to provide small orbital corrections. Figure 3.23 shows the launch sequence of GSLV-FO4 launch vehicle developed by Indian Space Research Organization (ISRO). It is a three stage vehicle comprising of a



Event	Time (s)	Altitude (km)	Velocity (km/s)
GS1 Burn Out	149.9	70.4	2.8
Heat Shield Separation	227.8	115.0	3.9
GS2 Burn Out	292.0	133.0	5.4
GS3 Burn Out	1002.9	220.1	10.2
Satellite Separation	13 seconds after GS3 Burn Out		

solid rocket engine based first stage (GS-1) and liquid rocket engine based second and third stages (GS-2 and GS-3).

Figure 3.23 Launch sequence of GSLV-FO4 launch vehicle

Ariane (Europe), Atlas (USA), Delta (USA), GSLV (India), PSLV (India), Long March (China) and Proton (Russia) are some of the launch vehicles being used internationally to launch satellites. The Ariane launch vehicle from the European Space Agency (ESA) has entered the fifth generation with ARIANE-5 series. The ARIANE-5 ECA (Enhanced Capability-A) (Figure 3.24) launch vehicle of the Ariane-5 series has the capacity of launching 12 tonnes of payload to geostationary transfer orbit. The GSLV launch vehicle (Figure 3.25), developed by Indian Space Research Organization (ISRO) can launcha2 to 2.5 tonne satellite into GTO (200 km

× 36 000 km).

Figure 3.24 Ariane 5 ECA launch vehicle

Reusable launch vehicles (RLV) are designed to be recovered intact and used again for subsequent launches. The Space Shuttle (Figure 3.26) from the USA is one example. It is generally used for human spaceflight missions.

Launch vehicles are also classified according to the mass they carry into orbit. As an example the PSLV (Polar Satellite Launch Vehicle),

developed by ISRO, India has the capability of launching a payload of 1000 to 1200 kg into polar sun-synchronous orbit. Another launch vehicle developed by ISRO, GSLV (Geostationary Satellite Launch Vehicle) has the capability to launch a2 to 2.5 tonne payload to GTO (200 km × 36 000 km).

Figure 3.25 GSLV launch vehicle (Courtesy:

Figure 3.26 Space shuttle (Courtesy: NASA)

4. Satellite Stabilization

Commonly employed techniques for satellite attitude control include:

1. Spin stabilization
2. Three-axis or body stabilization

1. Spin Stabilization

In a spin-stabilized satellite, the satellite body is spin at a rate between 30 and 100 rpm about an axis perpendicular to the orbital plane (Figure 3.28). Like a spinning top, the rotating body offers inertial stiffness, which prevents the satellite from drifting from its desired orientation. Spin-stabilized satellites are generally cylindrical in shape. For stability, the satellite should be spun about its major axis, having a maximum moment of inertia. To maintain stability, the moment of inertia about the desired spin axis should at least be 10 % greater than the moment of inertia about

the transverse axis.

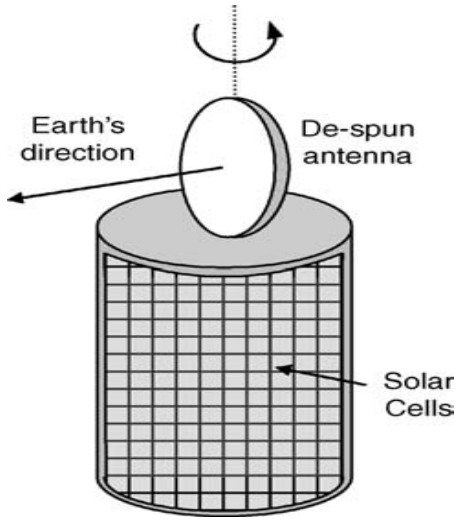


Figure 3.28 Spin stabilized satellite

There are two types of spinning configurations employed in spin-stabilized satellites. These include the simple spinner configuration and the dual spinner configuration. In the simple spin- ner configuration, the satellite payload and other subsystems are placed in the spinning section, while the antenna and the feed are placed in the de-spun platform. The de-spun platform is spun in a direction opposite to that of the spinning satellite body. In the dual spinner configuration, the entire payload along with the antenna and the feed is placed on the de-spun platform and the other subsystems are located on the spinning body. Modern spin-stabilized

satellites almost invariably employ the dual spinner configuration. It may be mentioned here that mounting of the antennae system on the de-spun platform in both the configurations ensures a constant pointing direction of the antennae. In both configurations, solar cells are mounted on the cylindrical body of the satellite. Intelsat-1 to Intelsat-4, Intelsat-6 and TIROS-1 are some of the popular spin-stabilized satellites. Figure 3.29 shows the photograph of Intelsat-4 satellite.

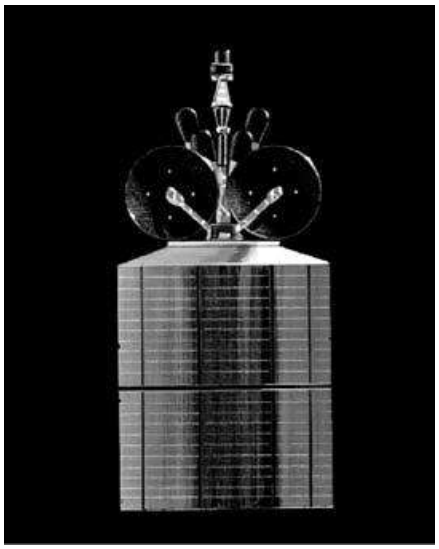


Figure 3.29 Spin-stabilized satellite (Intelsat-4)

2. Three-axis or Body Stabilization

In the case of three-axis stabilization, also known as body stabilization, the stabilization is achieved by controlling the movement of the satellite along the three axes, i.e. yaw, pitch and roll, with respect to a reference (Figure 3.30). The system uses reaction wheels or momentum

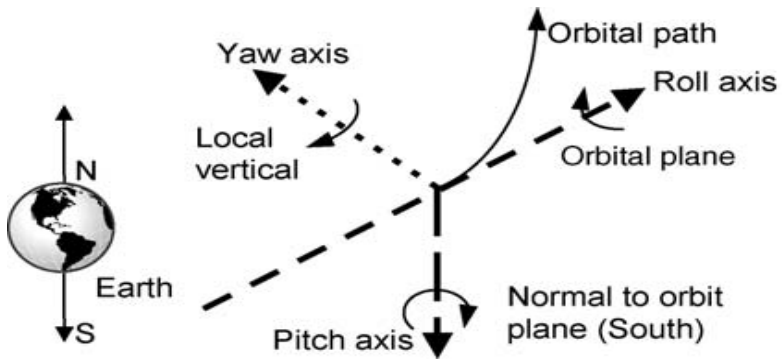


Figure 3.30 Three-axis stabilization

wheels to correct orbit perturbations. The stability of the three-axis system is provided by the active control system, which applies small corrective

forces on the wheels to correct the undesirable changes in the satellite orbit.

Most three-axis stabilized satellites use momentum wheels. The basic control technique used here is to speed up or slow down the momentum wheel depending upon the direction in which the satellite is perturbed. The satellite rotates in a direction opposite to that of speed change of the wheel. For example, an increase in speed of the wheel in the clockwise **direction will** make the satellite to rotate in a counter clockwise direction. The **momentum** wheels rotate in one direction and can be twisted by a gimbal motor to provide the required dynamic force on the satellite.

An alternative approach is to use reaction wheels. Three reaction wheels are used, one for each axis. They can be rotated in either direction depending upon the active correction force. The satellite body is generally box shaped for three-axis stabilized satellites. Antennae are mounted on the Earth-facing side and on the lateral sides adjacent to it. These satellites use flat solar panels mounted above and below the satellite body in such a way that they always point towards the sun, which is an obvious requirement.

Some popular satellites belonging to the category of three-axis stabilized satellites include Intelsat-5, Intelsat-7, Intelsat-8, GOES-8, GOES-9, TIROS-N and the INSAT series of satel- lites. Figure 3.31 is a photograph of the Intelsat-5 satellite.

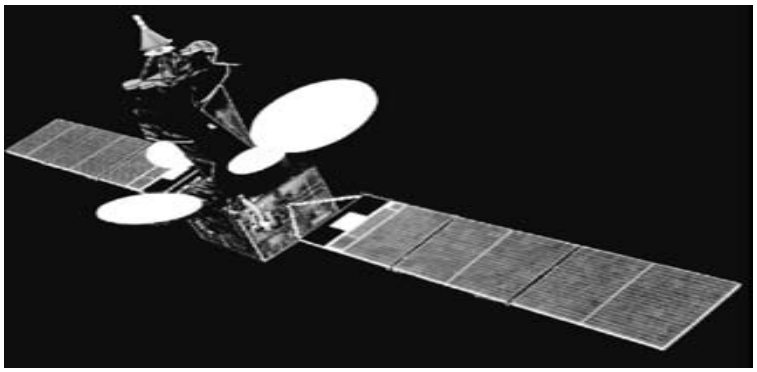


Figure 3.31 Three-axis stabilized satellite (Intelsat-5)

comparison between Spin-stabilized and Three-axis Stabilized Satellites

1. In comparison to spin-stabilized satellites, three-axis stabilized satellites have more power generation capability and more additional mounting area available for complex antennae structures.

2. Spin-stabilized satellites are simpler in design and less expensive than three-axis stabilized satellites.

3. Three-axis stabilized satellites have the disadvantage that the extendible solar array used in these satellites are unable to provide power when the satellite is in the transfer orbit, as the array is still stored inside the satellite during this time.

5. Look Angles of a Satellite

The look angles of a satellite refer to the coordinates to which an Earth station must be pointed in order to communicate with the satellite and are expressed in terms of azimuth and elevation angles. In the case where an Earth station is within the footprint or coverage area of a geostationary satellite, it can communicate with the satellite by simply pointing its antenna towards it.

The process of pointing the Earth station antenna accurately towards the satellite can be accomplished if the azimuth and elevation angles of the Earth station location are known.

Also, the elevation angle, as we shall see in the following paragraphs, affects the slant range, i.e. line of sight distance between the Earth station and the satellite.

In order to determine the look angles of a satellite, its precise location should be known. The location of a satellite is very often determined by the position of the sub-satellite point.

The sub-satellite point is the location on the surface of the Earth that lies directly between the satellite and the centre of the Earth. To an observer on the sub-satellite point, the satellite will appear to be directly overhead (Figure 3.38).

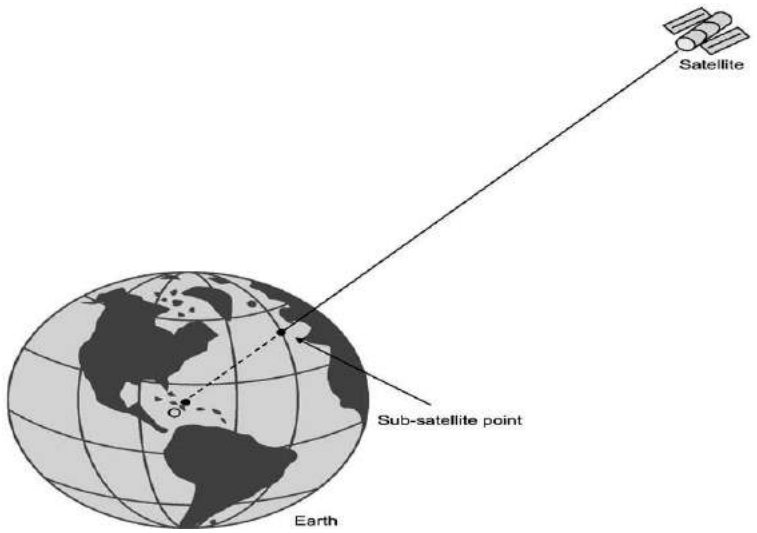


Figure 3.38 Sub-satellite point

6.. Azimuth Angle

The azimuth angle *A* of an Earth station is defined as the angle produced by the
intersection of the local horizontal plane and the plane passing through the Earth station, the satellite and the centre of the Earth with the true north (Figure 3.39). We can visualize that this line of intersection between the two above-mentioned planes would be one of the many possible tangents that can be drawn at the point of location of the Earth station. Depending upon the location of the Earth station and the sub-satellite point, the azimuth angle can be computed as follows:

1. Earth station in the northern hemisphere:

$\boldsymbol{A = 180^\circ - A'}$ when the Earth station is to the west of the satellite (3.19)

$\boldsymbol{A = 180^\circ + A'}$ when the Earth station is to the east of the satellite (3.20)

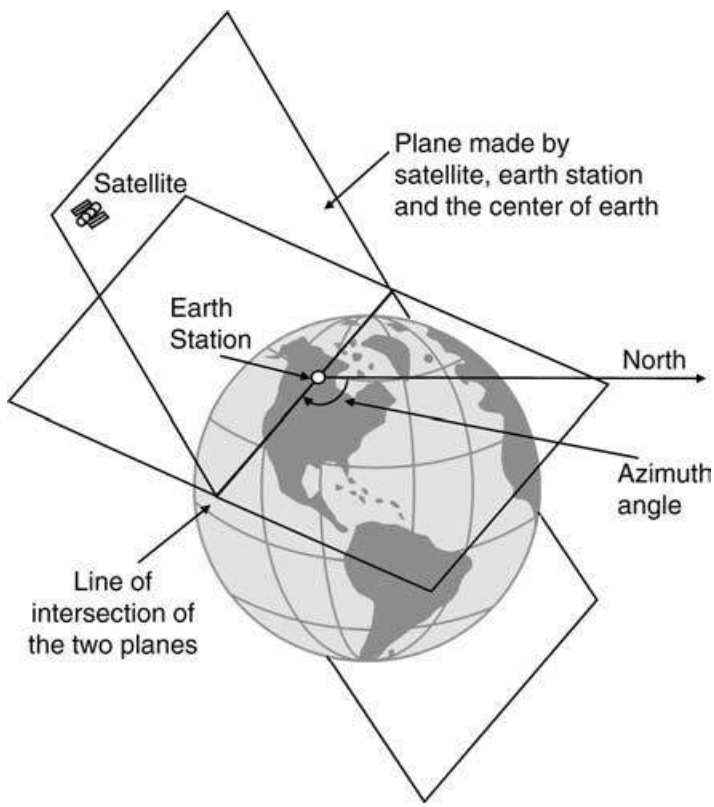


Figure 3.39 Azimuth angle

2. Earth station in the southern hemisphere:

$A = A'$ when the Earth station is to the west of the satellite

$A = 360^\circ - A'$ when the Earth station is to the east of the satellite

where A can be computed from

$$A' = \tan^{-1}\left(\frac{\tan|\theta_s - \theta_L|}{\sin\theta_l}\right) \tag{3.21}$$

where

θ_s = satellite longitude
θ_L = Earth station longitude
θ_l = Earth station latitude

7. Elevation Angle

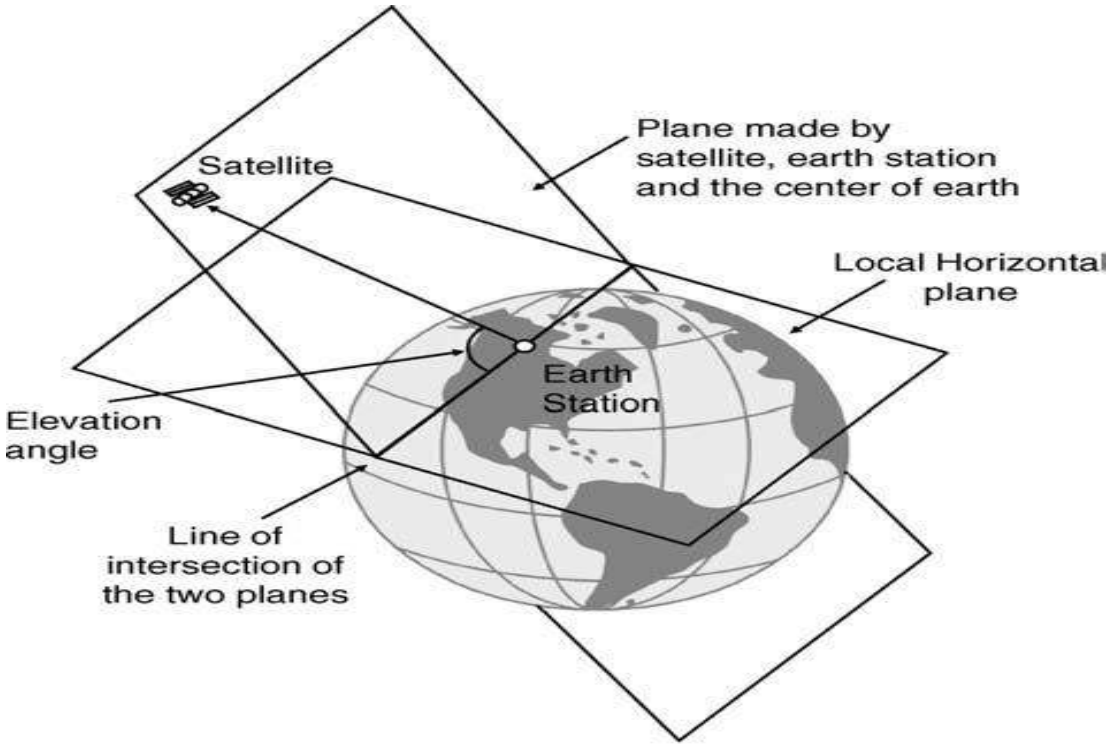
The Earth station elevation angle E is the angle between the line of intersection of the local horizontal plane and the plane passing through the Earth station, the satellite and the centre of the Earth with the line joining the Earth station and the satellite. Figures 3.40 (a) and (b) show the elevation angles for two different satellite and Earth station positions. It can be computed from:

$$E = \tan^{-1}\left[\frac{r - R\ \cos\theta_1\ \cos|\theta_s - \theta_L|}{R\sin\cos^{-1}(\cos\theta_1\cos|\theta_s - \theta_L|)}\right] - \cos^{-1}(\cos\theta_1\cos|\theta_s - \theta_L|) \qquad (3.22)$$

where

r = orbital radius

R = Earth's radius



(a)

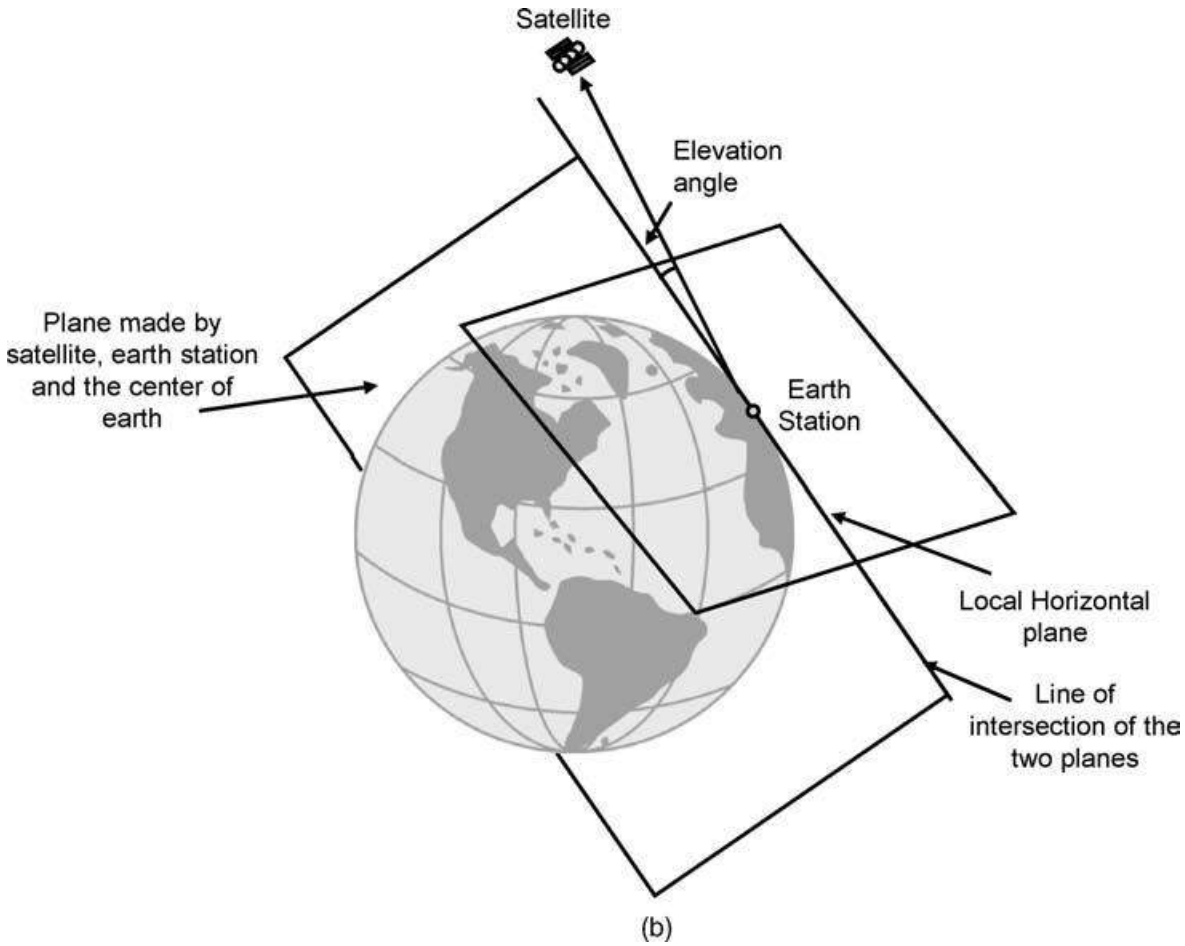


Figure 3.40 Earth station elevation angle

Calculating the Coverage Area for Satellites

حساب منطقة تغطية الاقمار الصناعية

8. Earth Coverage Area (footprint)

Earth coverage is the 'footprint' and is the surface area of the Earth that can be covered by a given satellite. In the discussion to follow, the effect of satellite altitude on Earth coverage provided by the satellite will be examined.

Computing the Slant Range

The elevation angle E, as mentioned earlier, has a direct bearing on the slant range. The smaller the angle of elevation of the Earth station, the larger is the slant range and the coverage angle. Refer to Figure 3.34.

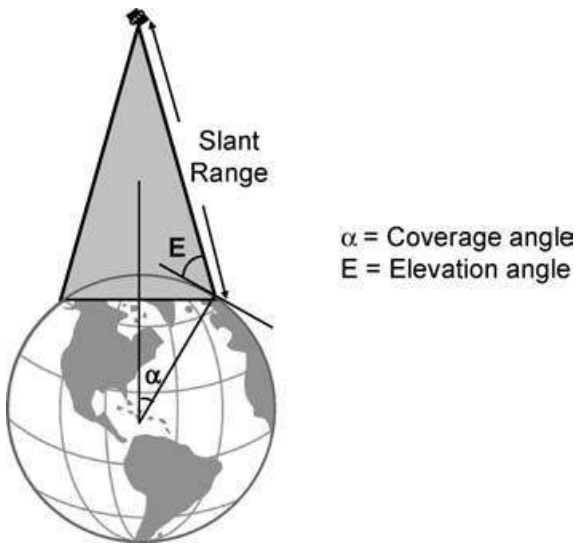


Figure 3.41 Elevation angle, slant range and coverage angle

The slant range can be computed from:

$$\text{Slant range } D = R^2 + (R+H)^2 - 2R(R+H)\ \sin\left[E + \sin^{-1}\left(\frac{R}{R+H}\right)\cos E\right]$$

Where:
R = radius of the Earth= 6378 km
E = angle of elevation
H = height of the satellite above the surface of the Earth

It is evident from the above expression that a zero angle of elevation leads to the maximum coverage angle. A larger slant range means a longer propagation delay time and a greater impairment of signal quality, as it has to travel a greater distance through the Earth's atmosphere.
Refer Figure 3.34. It is evident that the coverage area increases with the height of the satellite above the surface of the Earth. It varies from something like 1.5 % of the Earth's surface for low Earth satellite orbit at 200 km to about 42 % of the Earth's surface area for a satellite at a geostationary height of 36 000 km.

Table 3.1 shows the variation of coverage area as a function on of the satellite altitude. It can be seen from the table that the increase in coverage area with an increase in altitude is steeper in the beginning than it is as the altitude increases beyond 10000 km.
The coverage angle, can be computed from:

$$\text{Coverage angle } 2\alpha = 2\sin^{-1}\left(\frac{R}{R+H}\right)\cos E$$

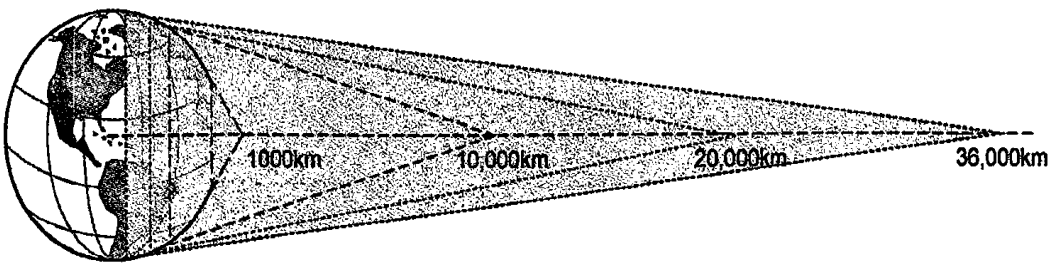


Figure 3.40 Satellite altitude and Earth coverage area

For maximum possible coverage. E = 0°. The expression reduces to

$$\text{Coverage angle } 2\alpha = 2\sin^{-1}\left(\frac{R}{R+H}\right)$$

The coverage angle can be computed to be approximately 1500 for a satellite at 200km and 170 for a satellite at a geostationary height of 36000 km.

Table 3.1 Variation of the coverage area as a function of the satellite altitute

Satellite altitude (km)	Coverage area (% of Earth's surface area)
200	1.5
300	2.0
400	2.5
500	3.0
600	3.5
700	4.5
800	5.5
900	6.0
1 000	7.0
2 000	12.0
4 000	18.5
5 000	21.5
6 000	24.0
7 000	26.0
8 000	27.5
9 000	29.0
10 000	30.0
15 000	35.0
20 000	37.5
25 000	40.0
30 000	41.5
36 000	43.0

Example/1

Determine the theoretical maximum area of the Earth's surface that would be in view from a geostationary satellite orbiting at a height of 35786 km from the surface of the Earth. Also, determine the area in view for a minimum elevation angle of 10 °. (Assume that the radius of the Earth is 6378 km.)

Solution: Refer to Figure 3.44. For the maximum possible coverage angle, the elevation angle E must be 0°. In that case the coverage angle α is given by

$$\text{Coverage angle } \alpha = \sin^{-1}\left[\left(\frac{R}{R+H}\right)\cos E\right] = \sin^{-1}\left(\frac{R}{R+H}\right)$$

where

R = Earth's radius
H = height of the satellite above the Earth's surface

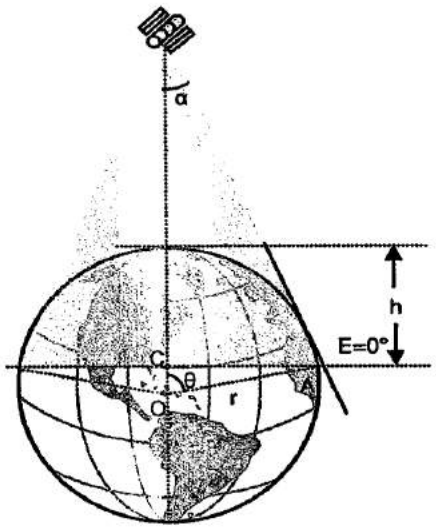


Figure 3.44 Figure for Problem 3.10

Thus

$$\alpha = \sin^{-1}\left(\frac{6378}{42164}\right) = 8.7°$$

This gives

$$\theta = 90° - \alpha - E = 90° - 8.7° = 81.3°$$

In the right-angled triangle OAC, OC = OA × sin 8.7°, so angle OAC = 8.7° and OC = 6378 × 0.1512 = 964.7 km. From the geometry, the covered surface area is given by

$$2\pi R(6378 - 964.7) = 2\pi \times 6378 \times 5413.3 = 216\ 823\ 452\,\text{km}^2$$

For $E = 10°$,

$$\alpha = \sin^{-1}\left[\left(\frac{R}{R+H}\right)\cos E\right] = \sin^{-1}\left[\left(\frac{6378}{42\,164}\right)\cos 10°\right] = 8.56°$$

This gives

$$\theta = 90° - 8.56° - 10° = 71.44°$$

The new value of OC is

$$6378 \sin 18.56° = 2030\,\text{km}$$

$$\text{Covered area} = 2\pi R(6378 - 2030) = 2\pi \times 6378 \times 4348 = 174\,154\,096\,\text{km}^2$$

H.W:

Q1: Draw how the satellites work.

Q2: How computing the slant range angles.

Q3: Draw the variation of satellite altitude with Earth coverage area.

1. Computing the line-of-Sight Distance between Two Satellites

Refer to Figure 3.35. The line-of-sight distance between two satellites placed in the same circular orbit can be computed from triangle ABC formed by the points of location of two satellites and the centre of the Earth. The line-of-sight distance AB in this case is given by

$$AB = \sqrt{(AC^2 + BC^2 - 2\ AC\ BC\ \cos\theta)} \tag{3.25}$$

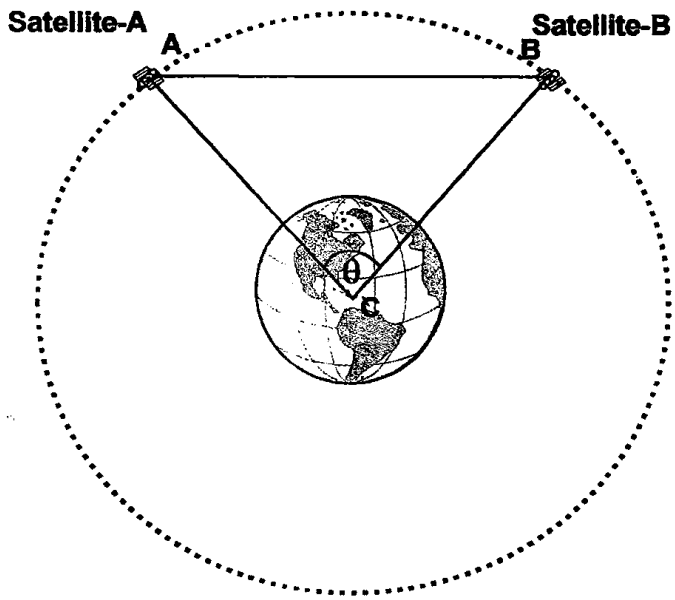


Figure 3.35 Line-of-sight distance between two satellites

Note also that angle θ will be the angular separation of the longitudes of the two satellites. For example, if the two satellites are located at 30°E and 60°E, θ would be equal to 30°. If the two locations are 30°W and 60°E, then in that case θ would be 90°. The maximum line-of-sight distance between these two satellites occurs when the satellites are placed so that the line joining the two becomes tangent to the Earth's surface, as shown in Figure 3.36.

In this the case, the maximum line-of-sight distance (AB) equals OA + OB, which further equals 2OA or 2OB as OA = OB. If R is the radius of the Earth and H is the height of satellites above the surface of the Earth, then

$$OA = AC\ \sin\theta = (R + H)\ \sin\theta \tag{3.26}$$

Now

$$\theta = \cos^{-1}\left(\frac{R}{R+H}\right) \tag{3.27}$$

Therefore

$$\mathrm{OA} = (R+H)\ \sin\left[\cos^{-1}\left(\frac{R}{R+H}\right)\right] \tag{3.28}$$

and

$$\text{Maximum line-of-sight distance} = 2(R+H)\ \sin\left[\cos^{-1}\left(\frac{R}{R+H}\right)\right] \tag{3.29}$$

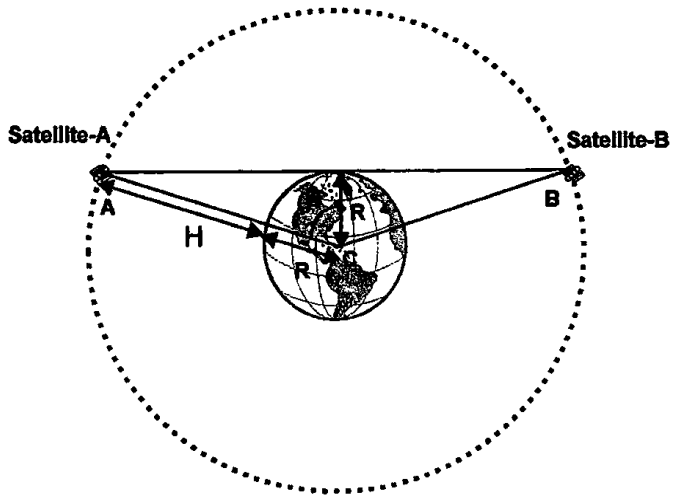


Figure 3.36 Maximum line-of-sight distance between two satellites

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Problem 3.6

A satellite in the Intelsat-VI series is located at 37°W and another belonging to the Intelsat-VII series is located at 74°E (Figure 3.37). If both these satellites are in a circular equatorial geostationary orbit with an orbital radius of 42 164 km, determine the intersatellite distance.

Solution: The intersatellite distance can be computed from

$$\sqrt{(D_1^2 + D_2^2 - 2D_1D_2\cos\theta)}$$

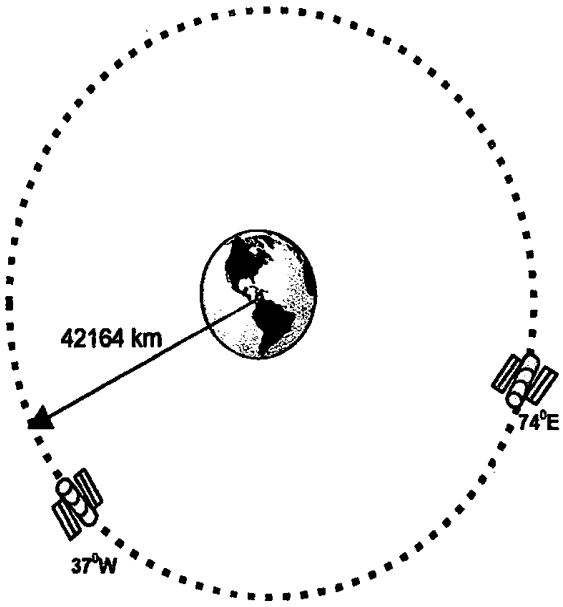

Figure 3.37 Figure for Problem 3.6

where

D_1 = orbital radius of the first satellite
D_2 = orbital radius of the second satellite Now
θ = angle formed by two radii

$$D_1 = D_2 = 42\,164 \text{ km and } \theta = 37° + 74° = 111°$$

$$\begin{aligned}\text{Intersatellite distance} &= \sqrt{\left[(42\,164)^2 + (42\,164)^2 - 2 \times 42\,164 \times 42\,164 \times \cos 111°\right]} \\ &= \sqrt{\left[2 \times (42\,164)^2 (1 - \cos 168.5°\right]} \\ &= 1.414 \times 42\,164 \times 1.165 = 69\,457 \text{ km}\end{aligned}$$

Problem 3.7

An Earth station is located at 30°W longitude and 60°N latitude. Determine the Earth station's azimuth and elevation angles with respect to a geostationary satellite located at 50°W longitude. The orbital radius is 42 164 km. (Assume the radius of the Earth to be 6378 km.)

Solution: Since the Earth station is in the northern hemisphere and is located towards east of the satellite, the azimuth angle A is given by $180° + A'$, where A' can be computed from

$$A' = \tan^{-1}\left(\frac{\tan|\theta_s - \theta_L|}{\sin\theta_l}\right)$$

where

θ_s = satellite longitude = 50°W
θ_L = Earth station longitude = 30°W
θ_l = Earth station latitude = 60°N

Therefore

$$A' = \tan^{-1}\left(\frac{\tan 20°}{\sin 60°}\right) = \tan^{-1}\left(\frac{0.364}{0.866}\right) = \tan^{-1} 0.42 = 22.8°$$

and

$$A = 180 + 22.8 = 202.8°$$

The earth station elevation angle is given by

$$E = \tan^{-1}\left[\frac{r - R\cos\theta_l \cos|\theta_s - \theta_L|}{R\sin\cos^{-1}(\cos\theta_l \cos|\theta_s - \theta_L|)}\right] - \cos^{-1}(\cos\theta_l \cos|\theta_s - \theta_L|)$$

where

r = orbital radius
R = Earth's radius

Substituting the values of various parameters gives

$$\begin{aligned} E &= \tan^{-1}\left[\frac{42\,164 - 6378\ \cos 60° \cos 20°}{6378\sin\cos^{-1}(\cos 60° \cos 20°)}\right] - \cos^{-1}(\cos 60° \cos 20°) \\ &= \tan^{-1}\left[\frac{42\,164 - 2998}{6378\sin(\cos^{-1} 0.47)}\right] - \cos^{-1} 0.47 \\ &= \tan^{-1}\left(\frac{39\,166}{5631}\right) - 62° \\ &= 81.8° - 62° = 19.8° \end{aligned}$$

Therefore,

$$\text{Azimuth} = 202° \qquad \text{and} \qquad \text{Elevation} = 19.8°$$

9. Satellite Ground Tracks

The ground track of an orbiting satellite is the path followed by the sub-satellite point, i.e. the point formed by the projection of the line joining the orbiting satellite with the center of the Earth on the surface of the Earth (Figure 3.48).

If the Earth were not rotating, the ground track would simply be the circumference of the great circle formed by the bisection of the Earth with the orbital plane of the satellite. In reality, however, the Earth does rotate; with the result, that the ground track gets modified from what it would be in the hypothetical case of a non-rotating Earth.

Two factors that influence the ground track due to Earth's rotation include the altitude of the satellite, which in turn determines the satellite's angular velocity, and the latitude at which the satellite is located, which determines the component of the Earth's rotation applicable at that point. In simple words, if we know the ground track that would have been there had the Earth been static, modification to this track at any given point in the satellite orbit would depend on the satellite altitude at that point and also on the latitude of that point.

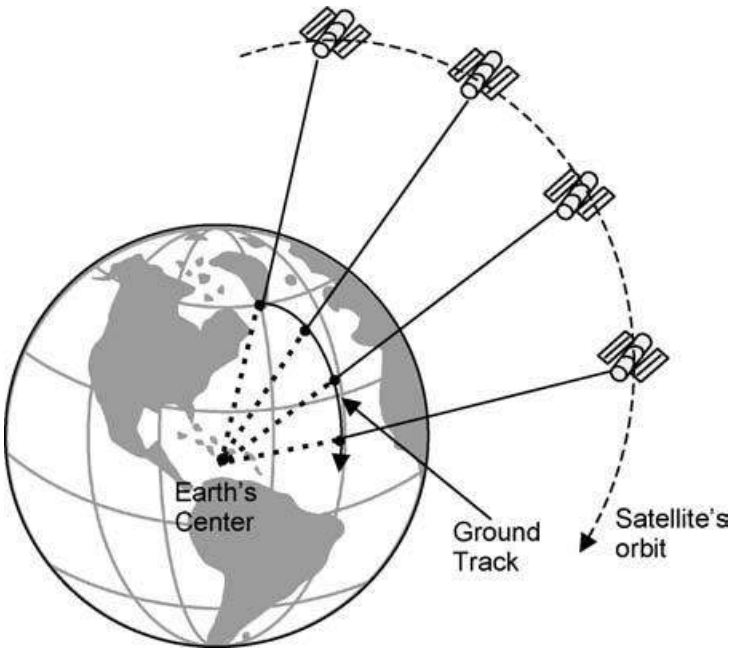

Figure 3.48 Satellite ground tracks

2.1 Effect of Altitude

The higher the altitude of the satellite, the smaller is the angular velocity and the greater will be the displacement of the ground track towards the west due to the Earth's rotation (Figure 3.49).

In the case of circular orbits, the ground track of the satellite at a higher altitude will shift more than that of the satellite in a lower altitude orbit. In the case of eccentric orbits, the shift in the ground track is much less around the perigee point for the same reasons.

2.2 Effect of Latitude

The Earth's relative rotation rate decreases with an increase in latitude, becoming zero at the poles. As a consequence, the shift in the ground track reduces as the latitudes over which the satellite moves increase. In fact, the shift in the ground track is zero at the poles. Another point worth mentioning here is that in the case of a satellite in a prograde orbit, the ground track intersects increasingly westerly meridians. In the case of satellites in a prograde orbit, the ground track intersects increasingly westerly meridians when the satellite's angular speed is less than the rotational rate of Earth and increasingly easterly meridians when it is more than

the Earth's rotation rate

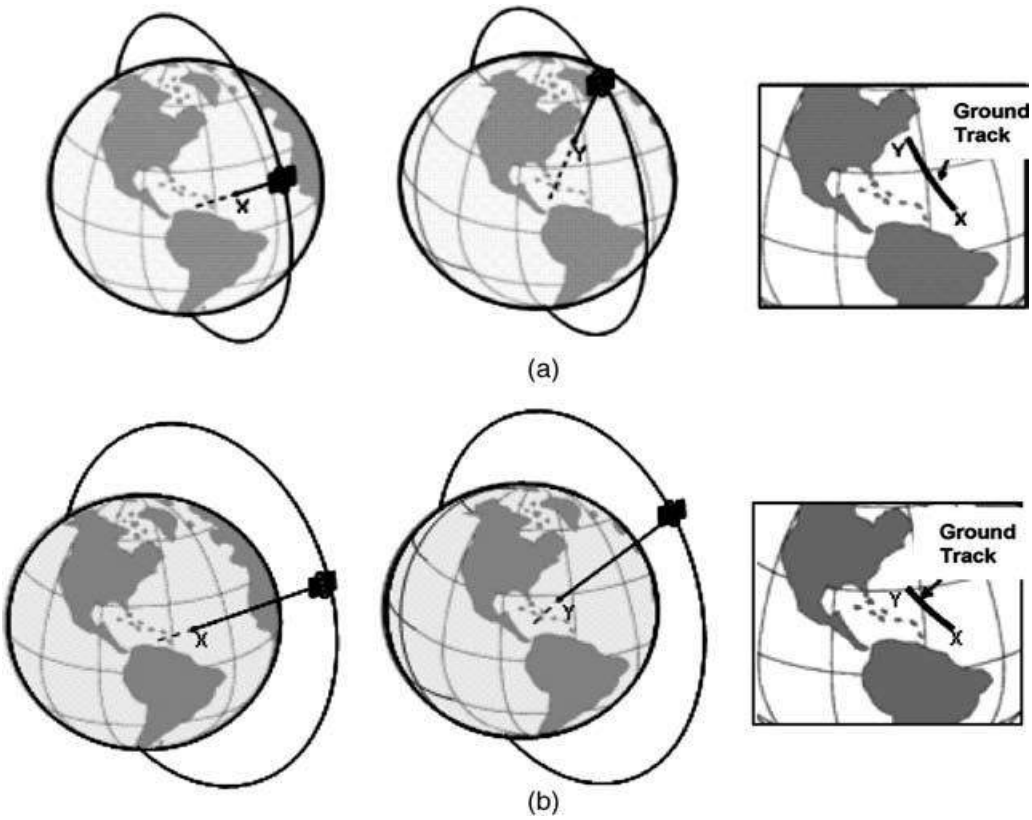

Figure 3.49 Effect of satellite altitude on the ground track

2.3 Orbit Inclination and Latitude Coverage

The northern and southern latitudes of the terrestrial segment covered by the satellite's ground track depend upon the satellite orbit inclination. The zone from the extreme northern latitude to the extreme southern latitude, which is symmetrical about the equator, is called the latitude coverage. Figure 3.50 illustrates the extent of latitude coverage for different inclinations.

It can be seen that the latitude coverage is 100 % only in the case of polar orbits. The higher the orbit inclination, the greater is the latitude coverage. This also explains why an equatorial orbit is not useful for higher latitude regions and also why a highly inclined Molniya orbit is more suitable for the territories of Russia and other republics of the former USSR.

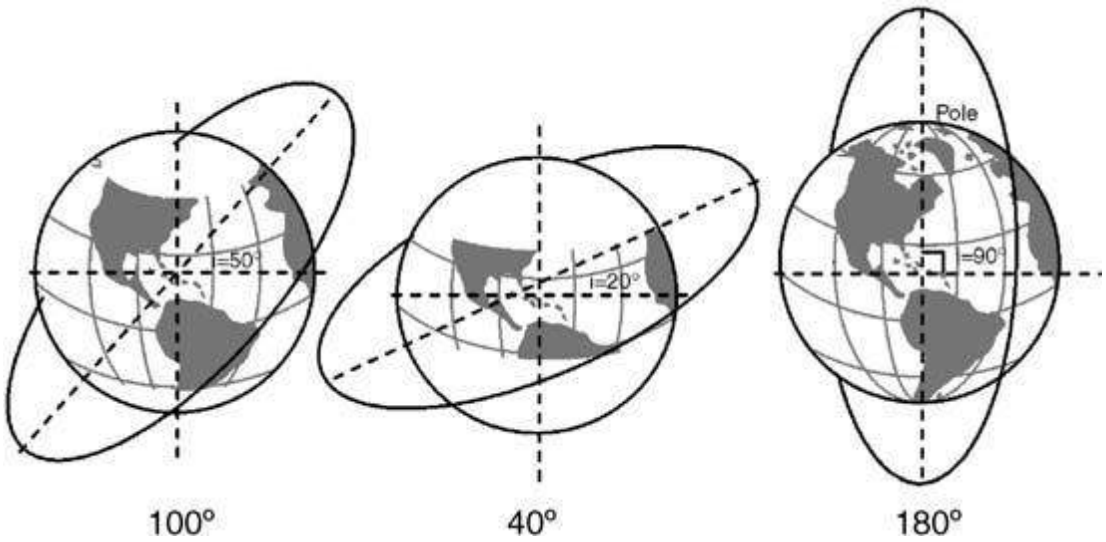

Figure 3.50 Effect of satellite orbital inclination on the latitude coverage

Problem:

A satellite is orbiting the Earth in a low Earth orbit inclined at 30◦ to the equatorial plane. Determine the extreme northern and southern latitudes swept by its ground track.

Solution:

The extreme latitudes covered in northern and southern hemispheres are the same as
orbit inclination.

Therefore, extreme northern latitude covered = 30◦N

Extreme southern latitude covered = 30◦S

In fact, the ground track would sweep all latitudes between 30◦N and 30◦S.

10. Applications of Remote Sensing Satellites:

تطبيقات الاقمار الصناعية في التحسس النائي تقسم حسب مدى الاشعة المستخدم:

1. Optical Remote Sensing Systems

يستخدم الاشعة ضمن المدى المرئي

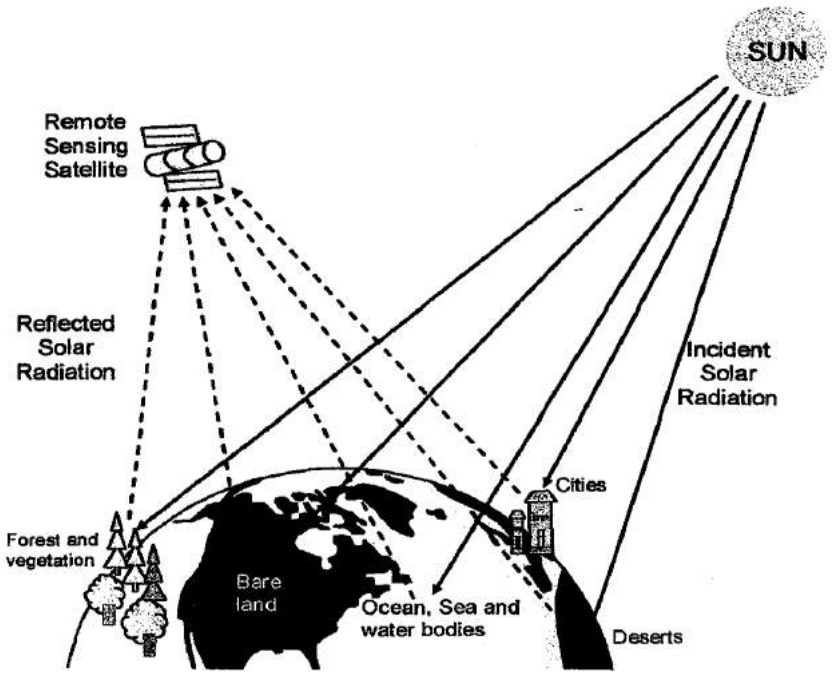

Figure 9.1 Optical remote sensing

2. Thermal Infrared Remote Sensing Systems

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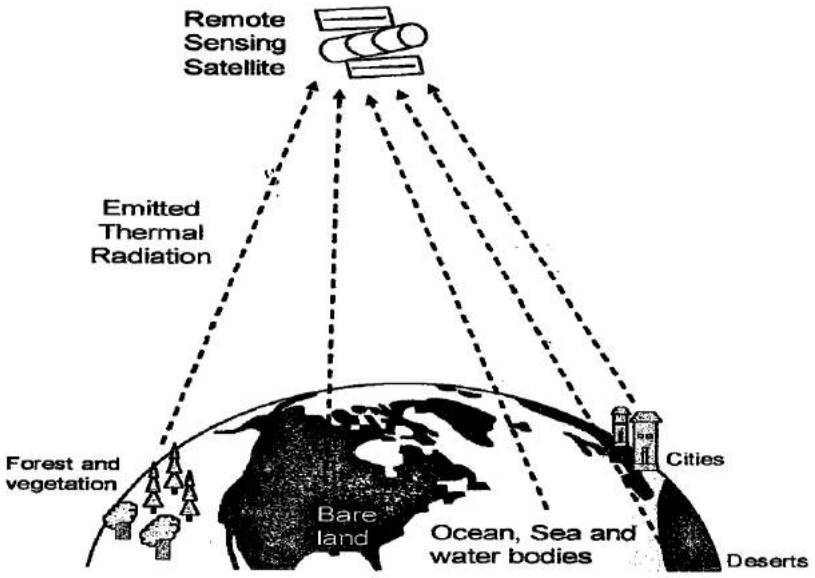


Figure 9.3 Thermal remote sensing

3 Microwave Remote Sensing Systems

يستخدم الاشعة الماكروية

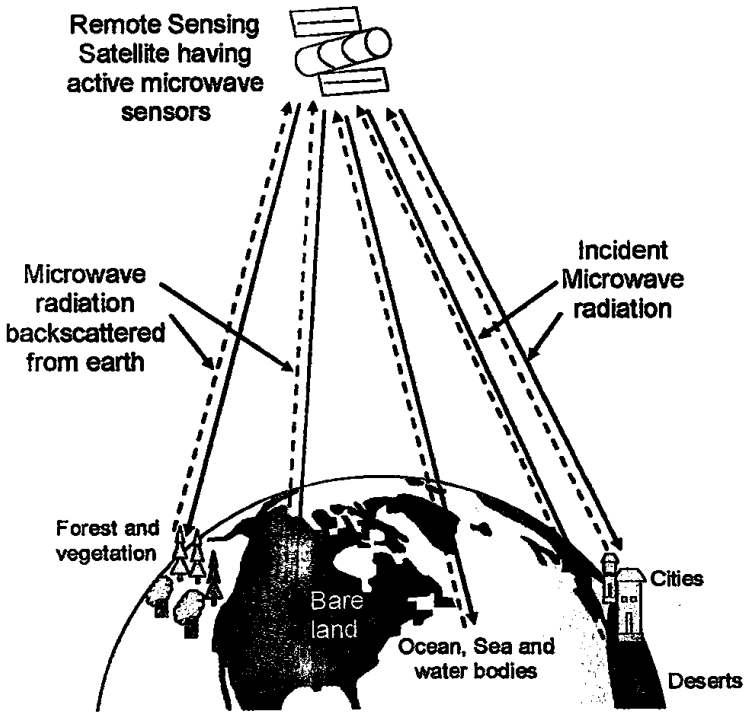


Figure 9.4 Active microwave remote sensing

Applications of Remote Sensing Satellites:

مجالات تطبيقات الاقمار الصناعية في التحسس النائي كثيرة اهمها ما ياتي:

1 Land Cover Classification

2 Land Cover Change Detection

3 Water Quality Monitoring and Management

4. Flood Monitoring

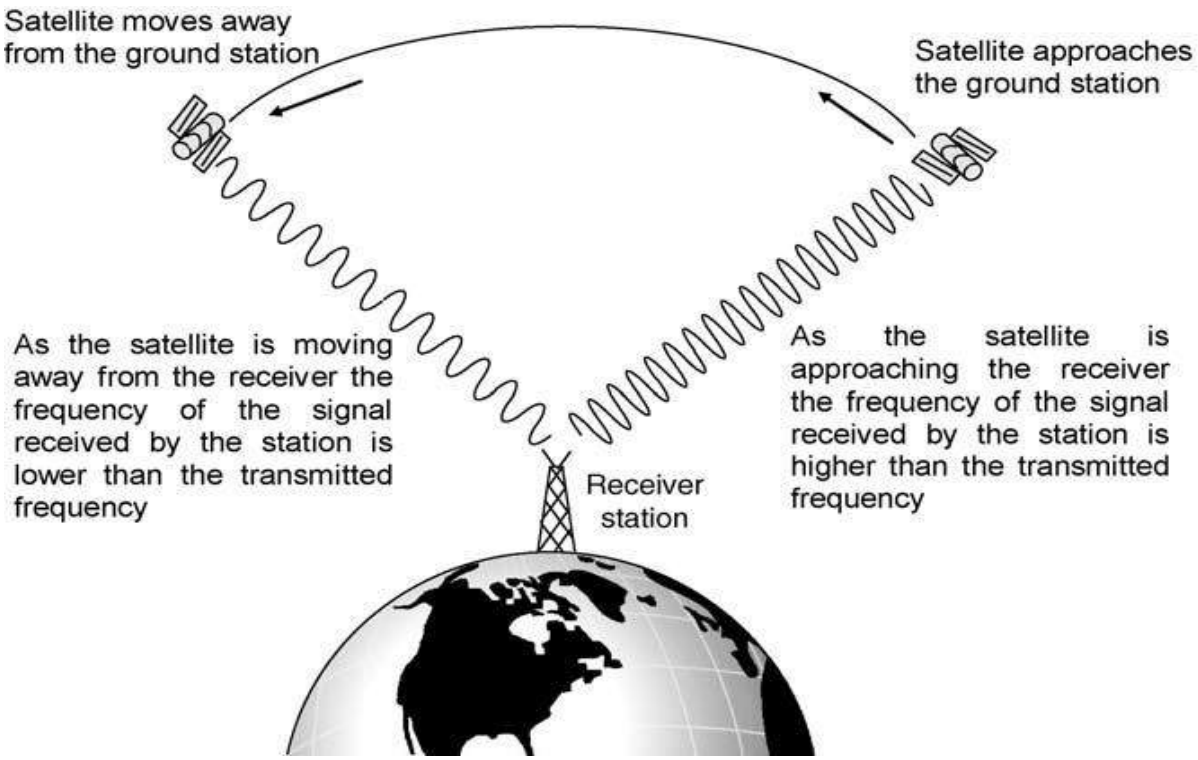
5 Urban Monitoring and Development

6 Measurement of Sea Surface Temperature

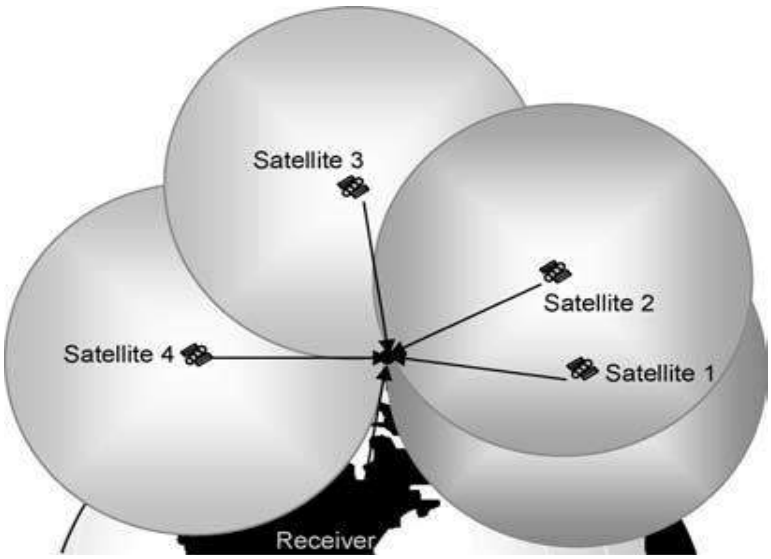
8 Global Monitoring

9 Predicting Disasters:

11. Global Positioning System (GPS)



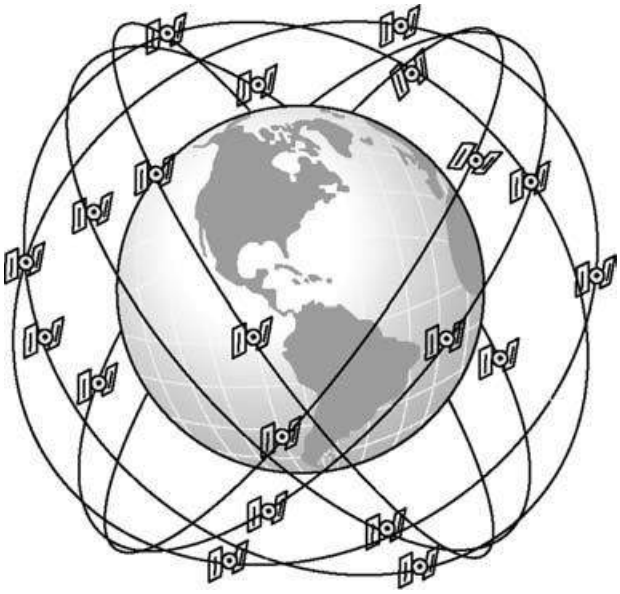
Principle of operation of Doppler Effect based satellite navigation systems



Principle of operation of trileration-based satellite navigation systems

Global Positioning System (GPS) segments

The GPS comprises of three segments, namely the space segment, control segment and user segment. All the three segments work in an integrated manner to ensure proper functioning of the system



1. Space segment of GPS

The space segment comprises of a 28 satellite constellation out of which 24 satellites are active satellites and the remaining four satellites are used as in-orbit spares.

The satellites are placed in six orbital planes, with four satellites in each plane.

Earth orbits (MEO) at an altitude of 20 200 km, inclined at 55◦ to the equator.

The orbital period of each satellite is around 12 hours (11 hours, 58 mins).

2. Control Segment

The control segment of the GPS system comprises aworld wide network of five monitor stations,

four ground antenna stations and a master control station. The monitor stations are located at Hawaii andKwajalein in the Pacific Ocean, Diego Garcia in the Indian Ocean, Ascension Island in the Pacific Ocean and Colorado Springs, Colorado.

There is a master control station (MCS) at Schriever Air Force Base in Colorado that controls the overall GPS network.

The ground antenna stations are located at Diego Garcia in the Indian Ocean, Kwajalein in the Pacific Ocean, Ascension Island in the Pacific Ocean and at Cape Canaveral, USA. Figure 12.11
shows the locations of the stations of the control segment.

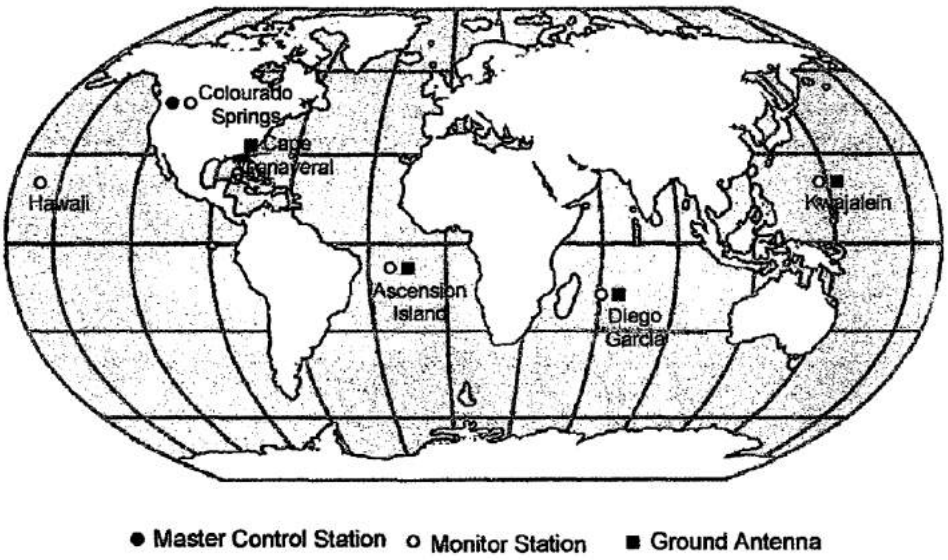


Figure 12.11 Control segment of GPS

3. User Segment

The user segment includes all military and civil GPS receivers intended to provide position, velocity and time information.

These receivers are either hand-held receivers or installed on aircraft, ships, tanks, submarines, cars and trucks. The basic function of these receivers isto detect, decode and process the GPS satellite signals. Some of the receivers have maps of the area stored in their memory.

GPS receiver (Reproduced by permission of © Randy ynum/www.nr6ca.org)

Operation of the GPS navigation system