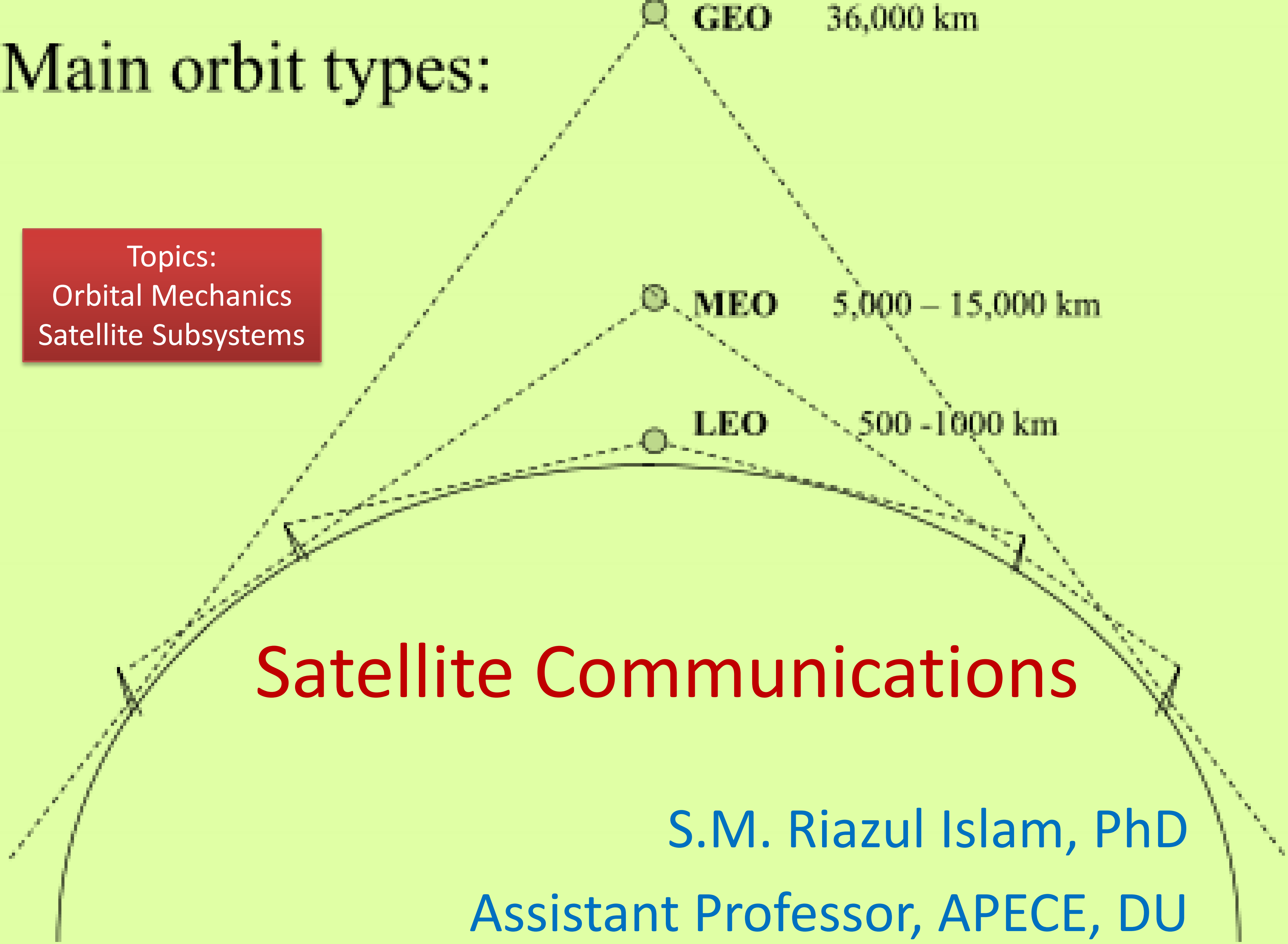

Topics:
Orbital Mechanics
Satellite Subsystems

Satellite Communications

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History

Important Milestones (before 1950)

Putting the concepts together

- 1600 Tycho Brache's experimental observations on planetary motion.
- 1609-1619 Kepler's laws on planetary motion
- 1926 First liquid propellant rocket lauched by R.H. Goddard in the US.
- 1927 First transatlantic radio link communication
- 1942 First successful launch of a V-2 rocket in Germany.
- 1945 Arthur Clarke publishes his ideas on geostationary satellites for worldwide communications (GEO concept).

Important Milestones (1950's)

Putting the pieces together

- 1956 - Trans-Atlantic cable opened (about 12 telephone channels – ope rator).
- 1957 First man-made satellite launched by former USSR (Sputnik, LEO).
- 1958 First US satellite launched (SCORE). First voice communication est ablished via satellite (LEO, lasted 35 days in orbit after batteries failed).

History

Important Milestones (1960's)

First satellite communications

- 1960 First passive communication satellite launched into space (Large balloons, Echo I and II).
- 1962: First non-government active communication satellite launched Telstar I (MEO).
- 1963: First satellite launched into geostationary orbit Syncom 1 (comms. failed).
- 1964: International Telecomm. Satellite Organization (INTELSAT) created.
- 1965 First communications satellite launched into geostationary orbit for commercial use Early Bird (re-named INTELSAT 1).

History

Important Milestones (1970's)

GEO applications development

- 1972 First domestic satellite system operational (Canada). INTERSPUT NIK founded.
- 1975 First successful direct broadcast experiment (one year duration; U SA-India).
- 1977 A plan for direct-to-home satellite broadcasting assigned by the IT U in regions 1 and 3 (most of the world except the Americas).
- 1979 International Mobile Satellite Organization (Inmarsat) established.

History

Important Milestones (1980's)

<u>GEO applications expanded</u>

- 1981 First reusable launch vehicle flight.
- 1982 International maritime communications made operational.
- 1983 ITU direct broadcast plan extended to region 2.
- 1984 First direct-to-home broadcast system operational (Japan).
- 1987 Successful trials of land-mobile communications (Inmarsat).
- 1989-90 Global mobile communication service extended to land mobile and aeronautical use (Inmarsat)

History

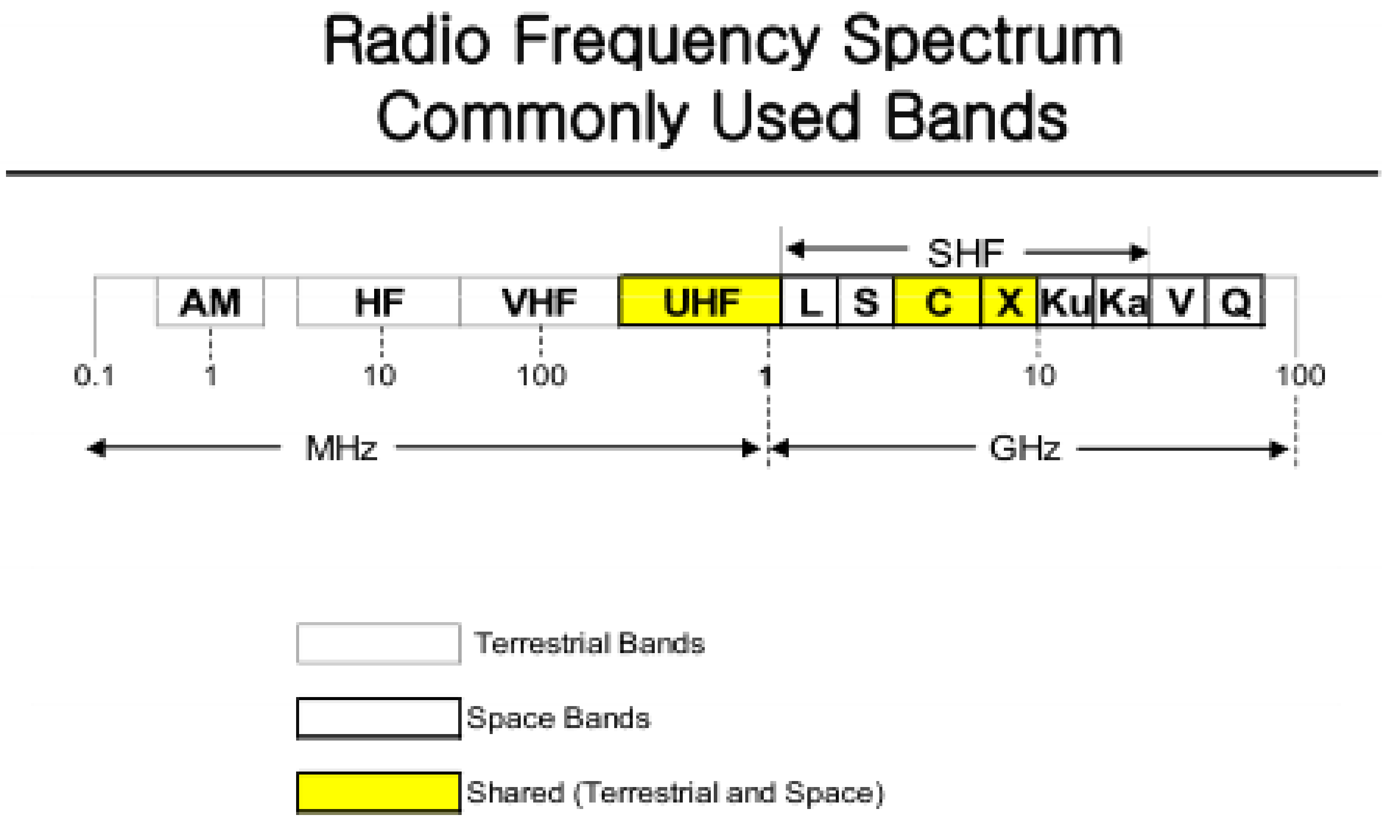
- 1990-95:

  - Several organizations propose the use of non-geostationary (NGSO) s atellite systems for mobile communications.

  - Continuing growth of VSATs around the world.
  - Spectrum allocation for non-GEO systems.
  - Continuing growth of direct broadcast systems. DirectTV created.
- 1997:

  - Launch of first batch of LEO for hand-held terminals (Iridium).
  - Voice service telephone-sized desktop and paging service pocket size mobile terminals launched (Inmarsat).
- 1998: Iridium initiates services.
- 1999: Globalstar Initiates Service.
- 2000: ICO initiates Service. Iridium fails and system is sold to Boeing.

Spectrum



Radio Frequency Spectrum
Commonly Used Bands
SHF
AM
HF
VHF
UHF
L
S
C
X
Ku
Ka
V
Q
0.1
1
10
100
1
10
100
MHz
GHz
Terrestrial Bands
Space Bands
Shared (Terrestrial and Space)

Spectrum

Insights on Frequency Selection:

(Part 1: Lower frequencies, stronger links)

- LEO satellites need lower RF frequencies:
  - Omni-directional antennas on handsets have low gain - typically G = 0 db = 1
  - Flux density F in W/m^2 at the earth's surface in any beam is indep endent of frequency
  - Received power is F x A watts , where A is effective area of anten na in square meters
  - For an omni-directional antenna $A = G \lambda^2 / 4 \pi = \lambda^2 / 4 \pi$
  - At **450 MHz, A = 353 cm^2,** at **20 GHz, A = 0.18 cm^2**
  - Difference is 33 dB - so don't use 20 GHz with an omni!

Spectrum

Insights on Frequency Selection:

(Part 2: Higher frequencies, higher capacity)

- GEO satellites need more RF frequencies
  - High speed data links on GEO satellites need about 0.8 Hz of RF bandwidth per bit/sec.
  - A 155 Mbps data link requires 125 MHz bandwidth
  - Available RF bandwidth:

C band	500 MHz	(All GEO slots occupied)
Ku band	750 MHz	(Most GEO slots occupied)
Ka band	2000 MHz	(proliferating)

Applications

Initial application of GEO Satellites: Telephony

1965	Early Bird	34 kg	240 telephone circuits
1968	Intelsat III	152 kg	1500 circuits
1986	Intelsat VI	1,800 kg	33,000 circuits
2000	Large GEO	3000 kg	8 - 15 kW power 1,200 kg payload

Applications

Current GEO Satellite Applications:

- **Broadcasting** - mainly TV at present
  - DirecTV, PrimeStar, etc.
- **Point to Multi-point** communications
  - **VSAT**, Video distribution for Cable TV
- **Mobile Services**
  - Motient (former American Mobile Satellite), IN MARSAT, etc.

Applications

Satellite Navigation:
GPS and GLONASS

- **GPS** is a medium earth orbit (MEO) satellite system
  - GPS satellites broadcast pulse trains with very accura te time signals
  - A receiver able to "see" four GPS satellites can calcul ate its position within 30 m anywhere in world
  - 24 satellites in clusters of four, 12 hour orbital period
- "You never need be lost again"
  - Every automobile and cellular phone will eventually have a GPS location read-out

Orbital Mechanics

Equations of Orbit:

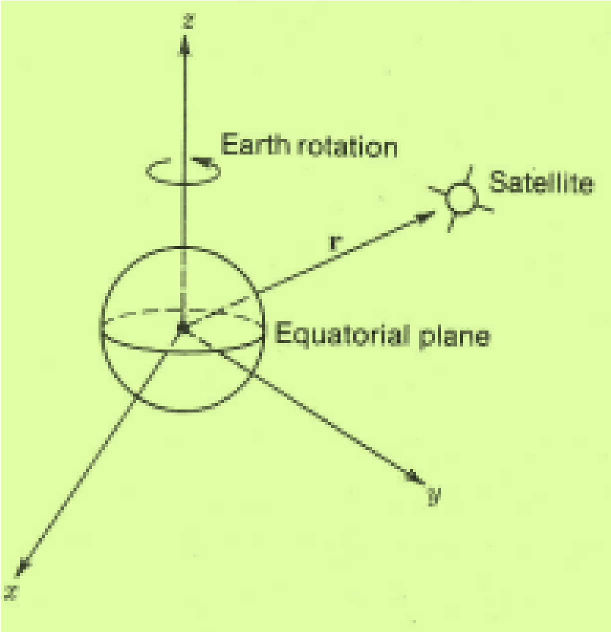


Figure 2.1 The first coordinate system used to describe the earth and a satellite. The x, y, and z axes are fixed by the earth's equatorial plane and north pole. The vector **r** locates the moving satellite with respect to the center of the earth.

Orbital Mechanics

A satellite of mass m is located at a vector distance $\mathbf{r}$ from the center of the earth. The gravitational force $\mathbf{F}$ on the satellite is given by

$$\mathbf{F} = -\frac{GM_E m\hat{\mathbf{r}}}{r^2} \tag{2.1}$$

where M_E is the mass of the earth and $G = 6.672 \times 10^{-11}$ Nm/kg^2 [1] is the universal gravitational constant. The unit vector in the $\mathbf{r}$ direction is represented by $\hat{\mathbf{r}}$. The product $GM_E = 3.9861352 \times 10^5$ km^3/s^2 is called Kepler's constant, symbol μ. Writing Newton's second law

$$\mathbf{F} = m\frac{d^2r}{dt^2}\hat{\mathbf{r}} \tag{2.2}$$

and equating the inertial force on the satellite to the gravitational force, we have

$$-\frac{\mu\hat{\mathbf{r}}}{r^2} = \frac{d^2r}{dt^2}\hat{\mathbf{r}} \tag{2.3}$$

This is a second-order vector linear differential equation, and its solution will involve six undetermined constants called the *orbital elements*. Writing

$$\mathbf{r} = r\hat{\mathbf{r}} \tag{2.4}$$

and rearranging we have

$$\frac{1}{r}\frac{d^2\mathbf{r}}{dt^2} + \frac{\mu\mathbf{r}}{r^3} = 0 \tag{2.5}$$

Orbital Mechanics

$$\mathbf{r} \times \frac{d^2\mathbf{r}}{dt^2} = 0 \tag{2.6}$$

Invoking the rule for finding the derivative of a product, it follows that

$$\frac{d}{dt}\left[\mathbf{r} \times \frac{d\mathbf{r}}{dt}\right] = \frac{d\mathbf{r}}{dt} \times \frac{d\mathbf{r}}{dt} + \mathbf{r} \times \frac{d^2\mathbf{r}}{dt^2} \tag{2.7}$$

The cross product of any vector with itself is zero; hence, (Eq. 2.7) may be rewritten

$$\frac{d}{dt}\left[\mathbf{r} \times \frac{d\mathbf{r}}{dt}\right] = 0 \tag{2.8}$$

This is equivalent to

$$\mathbf{r} \times \frac{d\mathbf{r}}{dt} = \mathbf{h} \tag{2.9}$$

where the constant, $\mathbf{h}$, is the orbital angular momentum of the satellite. But the orbital angular momentum can be a constant only if the orbit lies in a plane. Hence, the problem of satellite motion in three dimensions reduces to the problem of motion in a plane. Note that the orientation of this plane is as yet undetermined.

Orbital Mechanics

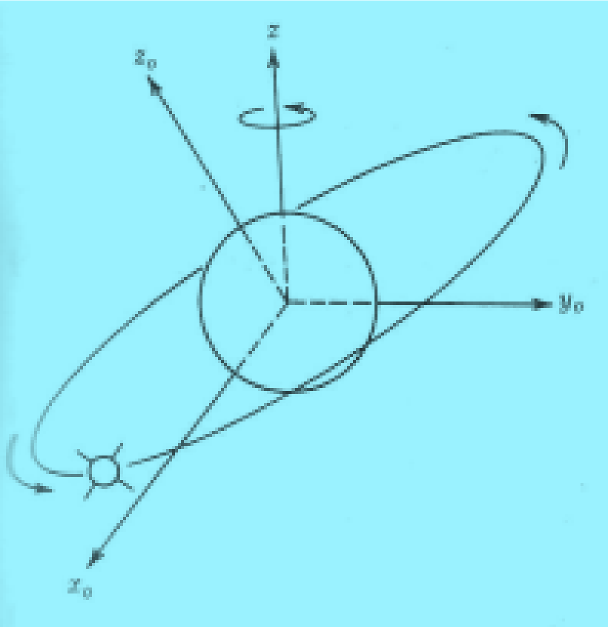


Figure 2.2 The orbital plane coordinate system. The x_o and y_o axes lie in the orbital plane and the z_o axis is perpendicular to it.

Orbital Mechanics

$$\hat{\mathbf{x}}_o\left(\frac{d^2x_o}{dt^2}\right) + \hat{\mathbf{y}}_o\left(\frac{d^2y_o}{dt^2}\right) + \frac{\mu(x_o\hat{\mathbf{x}}_o + y_o\hat{\mathbf{y}}_o)}{(x_o^2 + y_o^2)} = 0 \tag{2.10}$$

Equation (2.10) is easier to solve if it is expressed in the polar coordinate system (r_o, ϕ_o) of Figure 2.3 in the orbital plane. Using the transformations

$$x_o = r_o \cos \phi_o \tag{2.11a}$$

$$y_o = r_o \sin \phi_o \tag{2.11b}$$

$$\hat{\mathbf{x}}_o = \hat{\mathbf{r}}_o \cos \phi_o - \hat{\boldsymbol{\phi}}_o \sin \phi_o \tag{2.11c}$$

$$\hat{\mathbf{y}}_o = \hat{\boldsymbol{\phi}}_o \cos \phi_o + \hat{\mathbf{r}}_o \sin \phi_o \tag{2.11d}$$

and equating the $\hat{\mathbf{r}}_o$ components of Eq. (2.10) yields

$$\frac{d^2r_o}{dt^2} - r_o\left(\frac{d\phi_o}{dt}\right)^2 = -\frac{\mu}{r_o} \tag{2.12}$$

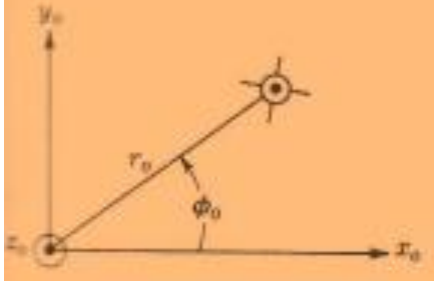


Figure 2.3 Polar coordinates in the orbital plane. In this drawing the orbital plane coincides with the plane of the paper.

Orbital Mechanics

Likewise equating the $\hat{\boldsymbol{\phi}}_o$ components we have

$$r_o\left(\frac{d^2\phi_o}{dt^2}\right) + 2\left(\frac{dr_o}{dt}\right)\left(\frac{d\phi_o}{dt}\right) = 0 \tag{2.13}$$

Following closely the approach presented in reference 2 we may rewrite Eq. (2.13) as

$$\frac{1}{r_o}\frac{d}{dt}\left(r_o^2\frac{d\phi_o}{dt}\right) = 0 \tag{2.14}$$

This is equivalent to

$$r_o^2\frac{d\phi_o}{dt} = \text{a constant} = |\mathbf{h}| = h \tag{2.15}$$

where h is the magnitude of the angular momentum vector in Eq. (2.9). Writing

$$r_o\left(\frac{d\phi_o}{dt}\right)^2 = \frac{h^2}{r_o^3} \tag{2.16}$$

and substituting this expression into Eq. (2.12), we have

$$\frac{d^2r_o}{dt^2} - \frac{h^2}{r_o^3} = -\frac{\mu}{r_o} \tag{2.17}$$

Orbital Mechanics

In order to find the equation relating r_o and ϕ_o (i.e., the equation of the orbit), we must eliminate time t from Eq. (2.17). To do this we define a new variable u by

$$u = \frac{1}{r_o} \tag{2.18}$$

so that

$$\frac{dr_o}{d\phi_o} = -\frac{1}{u^2}\frac{du}{d\phi_o} \tag{2.19}$$

and using the relationship

$$\frac{dr_o}{dt} = \left(\frac{dr_o}{d\phi_o}\right)\left(\frac{d\phi_o}{dt}\right) = \left(\frac{dr_o}{d\phi_o}\right)\left(\frac{h}{r_o^2}\right) = -h\frac{du}{d\phi_o} \tag{2.20}$$

to transform d^2r_o/dt^2 in Eq. (2.17) to

$$\frac{d^2r_o}{dt^2} = -h^2u^2\left(\frac{d^2u}{d\phi_o^2}\right) \tag{2.21}$$

We may rewrite Eq. (2.17) as

$$\frac{d^2u}{d\phi_o^2} + u = \frac{\mu}{h^2} \tag{2.22}$$

Orbital Mechanics

According to reference 2, the solution to this differential equation is

$$u = \frac{\mu}{h^2} + C\cos(\phi_o - \theta_o) \tag{2.23}$$

where C and θ_o are constants to be determined from the boundary conditions. If we express Eq. (2.23) in terms of r_o, we have

$$\begin{aligned} r_o &= \frac{1}{\mu/h^2 + C\cos(\phi_o - \theta_o)} \\ &= \frac{\left(\frac{h^2}{\mu}\right)}{1 + \left(\frac{h^2}{\mu}\right) C\cos(\phi_o - \theta_o)} \\ &= \frac{p}{1 + e\cos(\phi_o - \theta_o)} \end{aligned} \tag{2.24}$$

For $e < 1$ this is the equation of an ellipse whose semilatus rectum p is given by

$$p = \frac{h^2}{\mu} \tag{2.25}$$

and whose eccentricity e is h^2C/μ. That the orbit is an ellipse is Kepler's first law of planetary motion. Under the limiting condition $e = 0$, the orbit is a circle with the earth at its center.

Orbital Mechanics

Describing the Orbit

The quantity θ_o in Eq. (2.24) serves to orient the ellipse with respect to the orbital plane axes x_o and y_o. Now that we know the orbit is an ellipse, we can always choose x_o and y_o so that θ_o is zero. For the rest of this discussion we will assume that this has been done, making the equation of the orbit

$$r_o = \frac{p}{1 + e\cos\phi_o} \tag{2.26}$$

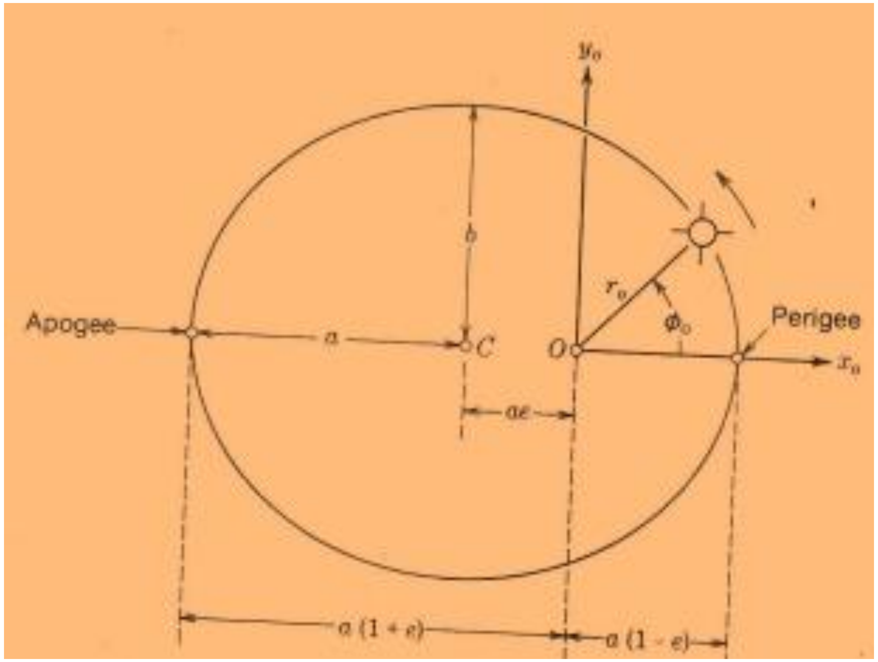
The path of the satellite in the orbital plane is shown in Figure 2.4. The lengths a and b of the semimajor and semiminor axis are given by reference 2 as

$$a = \frac{p}{1 - e^2} \tag{2.27}$$

$$b = a(1 - e^2)^{\frac{1}{2}} \tag{2.28}$$

The point in the orbit where the satellite is closest to the earth is called the *perigee* and the point where the satellite is farthest from the earth is called the *apogee*. To make θ_o equal to zero, we have chosen x_o axis so that both the apogee and perigee lie along it.

Orbital Mechanics



$$dA = 0.5 r_o^2 \left(\frac{d\phi_o}{dt}\right) dt = 0.5\, h\, dt$$

$$T = \frac{2\pi a^{\frac{3}{2}}}{\mu^{\frac{1}{2}}}$$

$$T^2 = \frac{4\pi^2 a^3}{\mu}$$

Radius of Geosynchronous orbit ?

T=86400 s➔a=42,241.558 km
--42,242 km

Figure 2.4 The orbit as it appears in the orbital plane. The point O is the center of the earth and the point C is the center of the ellipse. The two centers do not coincide unless the eccentricity e is zero. The dimensions a and b are the semimajor and semiminor axes of the orbital ellipse.

Orbital Mechanics

Locating the Sat. in the Orbit

Consider now the problem of locating the satellite in its orbit. The equation of the orbit may be rewritten by combining Eqs. (2.26) and (2.27) to obtain

$$r_o = \frac{a(1 - e^2)}{1 + e \cos \phi_o} \tag{2.32}$$

The angle ϕ_o (see Figure 2.4) is measured from the x_o axis and is called the *true anomaly*. Since we defined the positive x_o axis so that it passes through the perigee, ϕ_o measures the angle from the perigee to the instantaneous position of the satellite. The rectangular coordinates of the satellite are given by

$$x_o = r_o \cos \phi_o \tag{2.33}$$

$$y_o = r_o \sin \phi_o \tag{2.34}$$

The orbital period T is the time required for the satellite to complete one revolution and travel 2π rad. The average angular velocity is thus

$$\eta = \frac{2\pi}{T} = \frac{\mu^{\frac{1}{2}}}{a^{\frac{3}{2}}} = \frac{1}{a}\left(\frac{\mu}{a}\right)^{\frac{1}{2}} \tag{2.35}$$

Orbital Mechanics

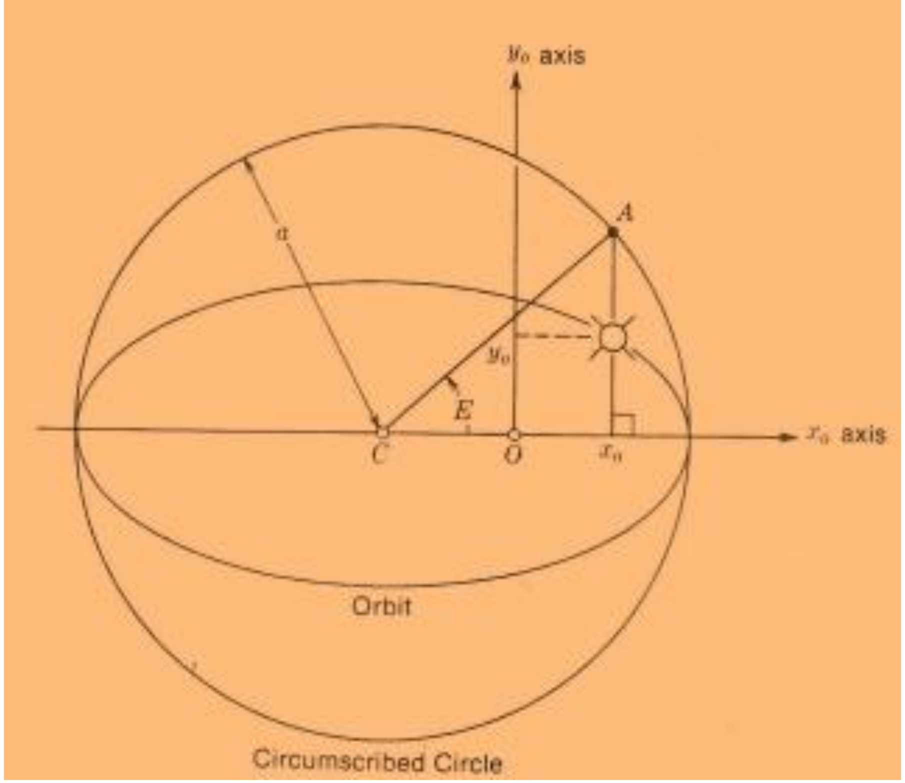


Figure 2.5 The circumscribed circle and the eccentric anomaly E. Point O is the center of the earth and point C is both the center of the orbital ellipse and the center of the circumscribed circle. The satellite location in the orbital plane coordinate system is specified by (x_0, y_0). A vertical line through the satellite intersects the circumscribed circle at point A. The eccentric anomaly E is the angle from the x_o axis to the line joining C and A.

Orbital Mechanics

$$v^2 = \left(\frac{dx_o}{dt}\right)^2 + \left(\frac{dy_o}{dt}\right)^2 = \left(\frac{dr_o}{dt}\right)^2 + r_o^2\left(\frac{d\phi_o}{dt}\right)^2 \tag{2.36}$$

It can be shown [2] that

$$v^2 = \left(\frac{\mu}{a}\right)\left(\frac{2a}{r_0} - 1\right) \tag{2.37}$$

From Eqs. (2.16), (2.15), and (2.27)

$$r_0^2\left(\frac{d\phi_0}{dt}\right)^2 = \frac{h^2}{r_0^2} = \frac{\mu p}{r_0^2} = \frac{\mu a(1 - e^2)}{r_0^2} \tag{2.38}$$

Thus Eq. (2.36) becomes

$$\left(\frac{\mu}{a}\right)\left(\frac{2a}{r_o} - 1\right) = \left(\frac{dr_o}{dt}\right)^2 + \left(\frac{\mu a}{r_o^2}\right)(1 - e^2) \tag{2.39}$$

and

$$\frac{dr_o}{dt} = \left\{\left(\frac{\mu}{ar_o^2}\right)[a^2e^2 - (a - r_o)^2]\right\}^{\frac{1}{2}} \tag{2.40}$$

Solving this equation for dt and multiplying by the mean angular velocity η, we obtain

$$\eta\, dt = \left(\frac{r_o}{a}\right)\frac{dr_o}{[a^2e^2 - (a - r_o)^2]^{\frac{1}{2}}} \tag{2.41}$$

Orbital Mechanics

$$r_o = a(1 - e\cos E) \tag{2.42}$$

Thus

$$a - r_o = ae\cos E \tag{2.43}$$

and when expressed in terms of E, Eq. (2.41) takes on the surprisingly simple form

$$\eta\, dt = (1 - e\cos E)\, dE \tag{2.44}$$

Let t_p be the *time of perigee*. This is simultaneously the time of closest approach to the earth, the time when the satellite is crossing the x_o axis, and the time when E is zero. If we integrate both sides of Eq. (2.44), we then obtain

$$\eta(t - t_p) = E - e\sin E \tag{2.45}$$

The left side of Eq. (2.45) is called the *mean anomaly*, M.

$$M = \eta(t - t_p) = E - e\sin E$$

1. Calculate η by Eq. (2.35)
2. Calculate M by Eq. (2.46)
3. Solve Eq. (2.46) for E
4. Find r_o from E using Eq. (2.43)
5. Solve Eq. (2.32) for ϕ_o
6. Use Eqs. (2.33) and (2.34) to calculate x_o and y_o.

Orbital Mechanics

Locating the Sat. wrt the Earth:

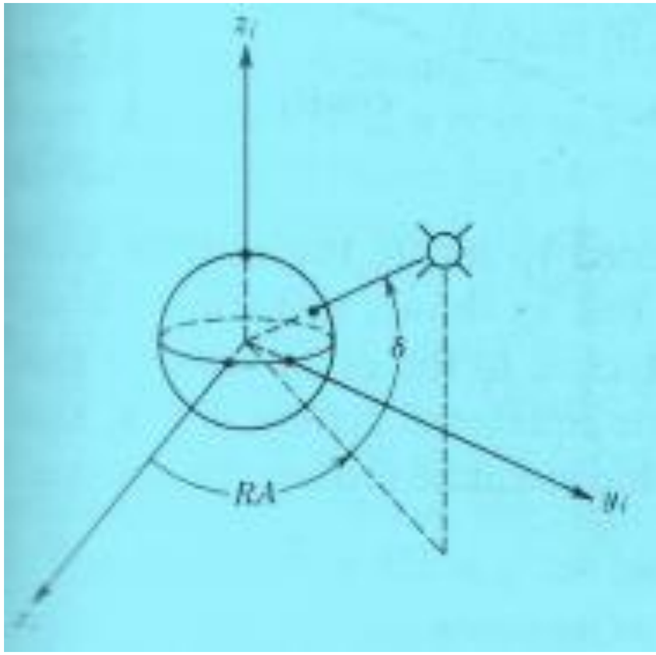


Figure 2.6 The geocentric equatorial system. This differs from the first coordinate system of Figure 2.1 only in that the x_i axis points toward the first point of Aries. An object may be located by its right ascension RA and its declination δ.

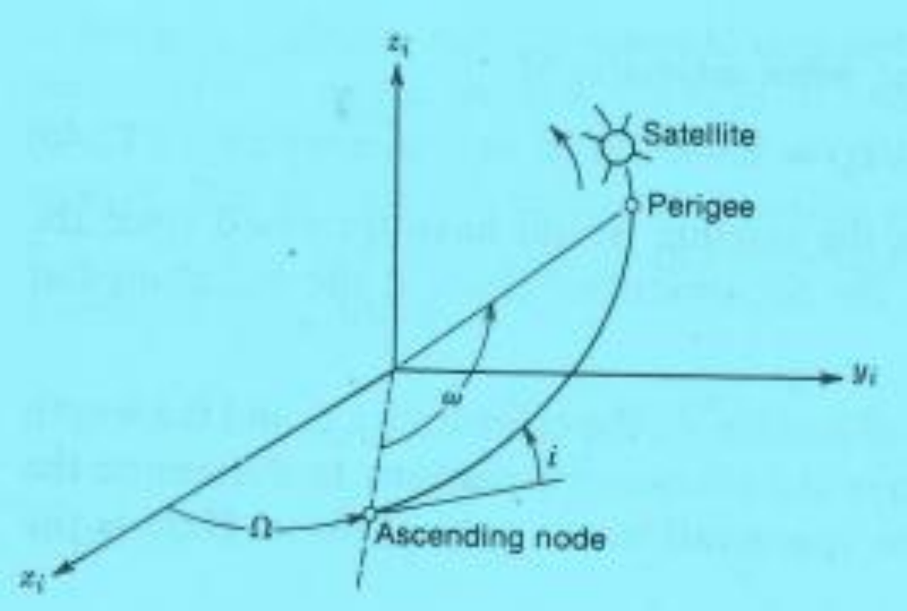
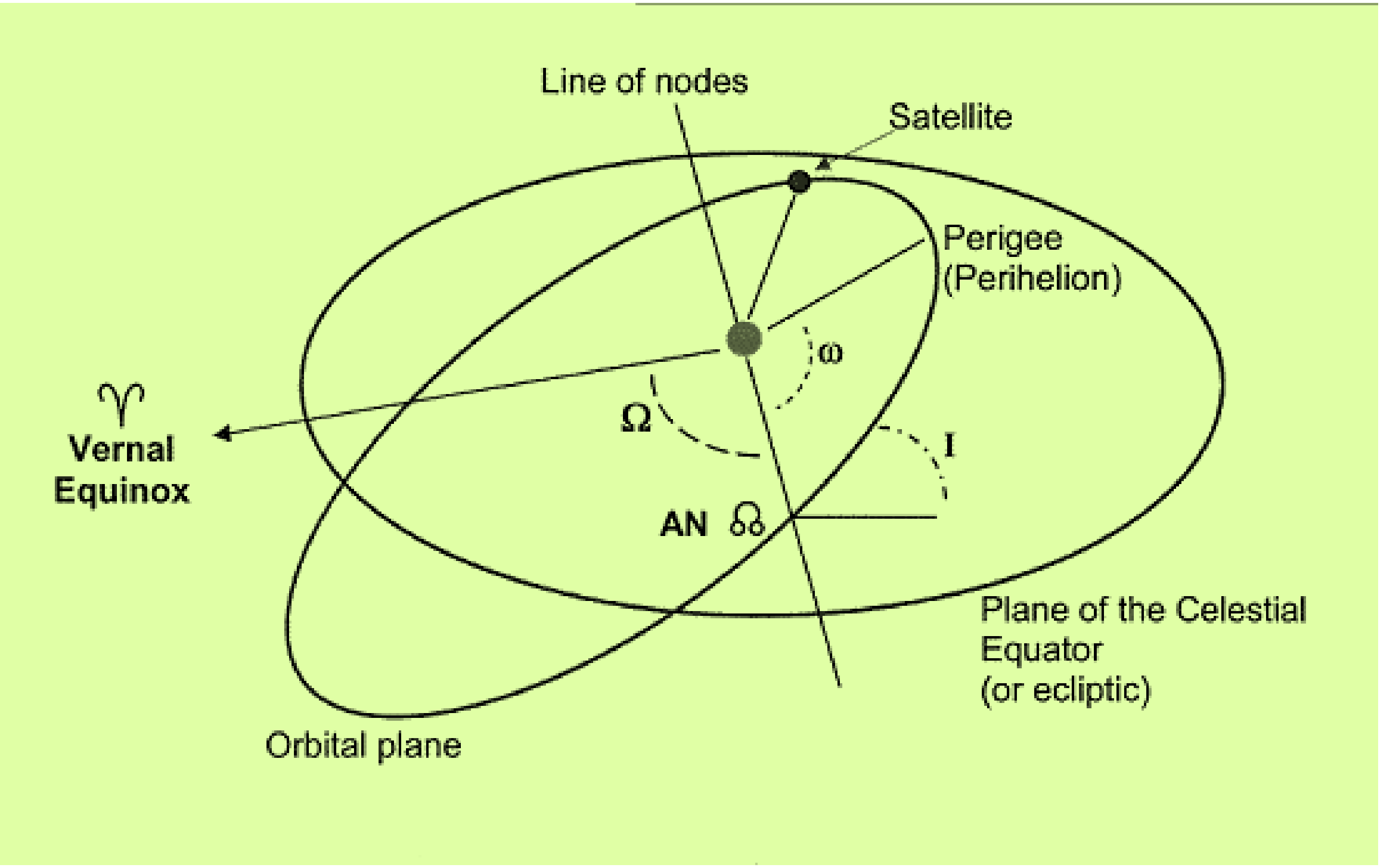
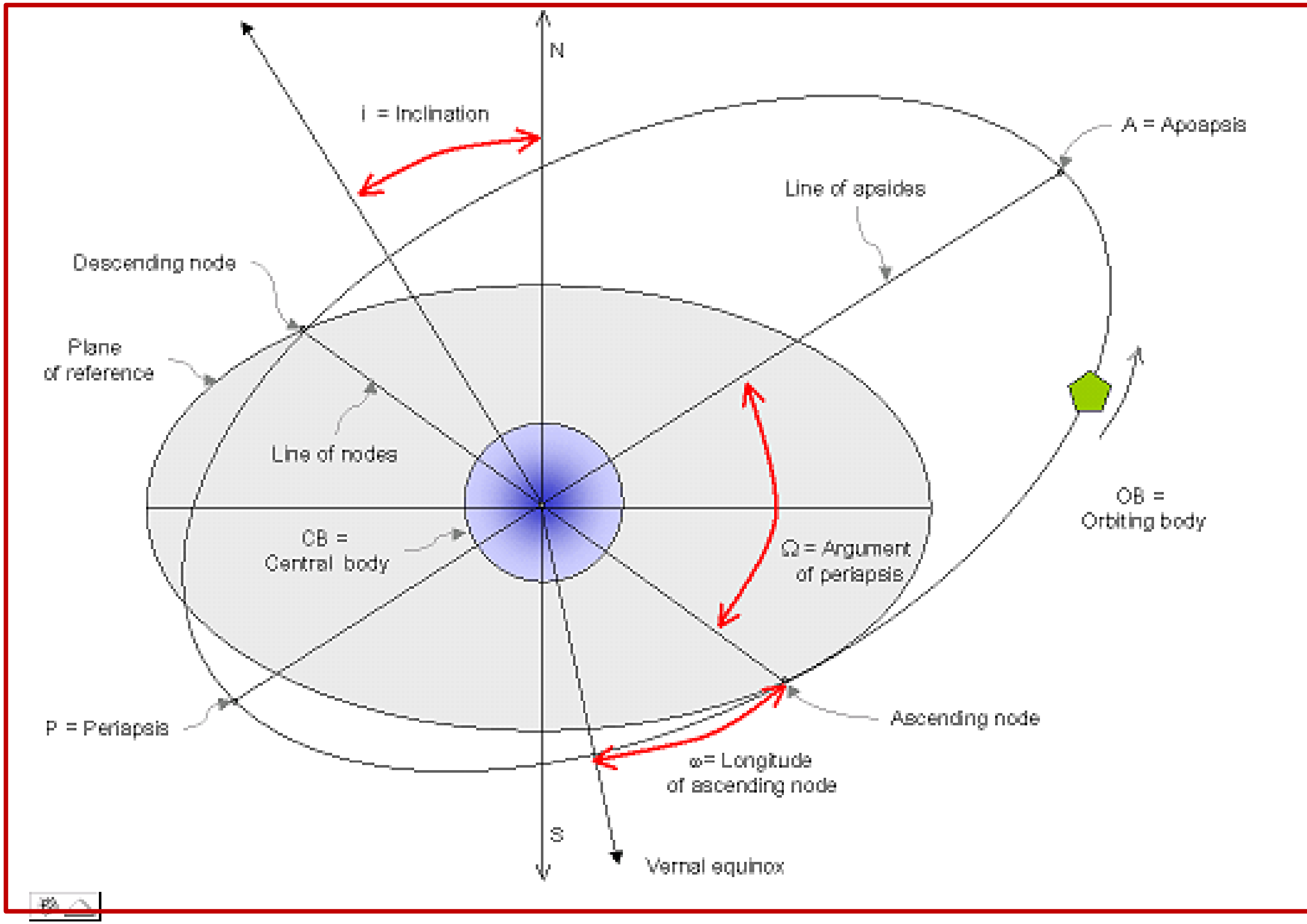


Figure 2.7 Locating the orbit in the geocentric equatorial system. The satellite penetrates the equatorial plane (while moving in the positive z_i direction) at the ascending node. The right ascension of the ascending node is Ω and the inclination i is the angle between the equatorial plane and the orbital plane. Angle ω, measured in the orbital plane, locates the perigee with respect to the equatorial plane.

Orbital Mechanics

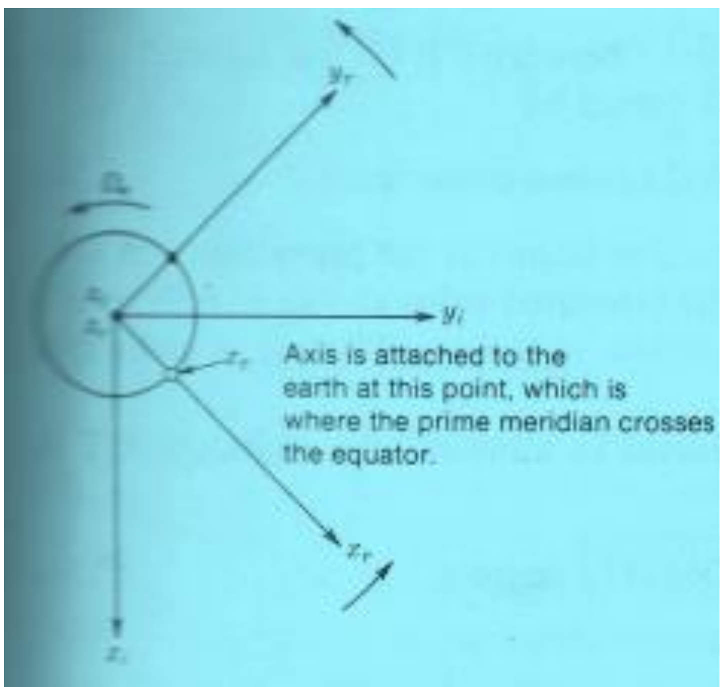


Orbital Mechanics



Orbital Mechanics

$$\begin{bmatrix} x_i \\ y_i \\ z_i \end{bmatrix} = \begin{bmatrix} \cos(\Omega)\cos(\omega) - \sin(\Omega)\cos(i)\sin(\omega) & -\cos(\Omega)\sin(\omega) - \sin(\Omega)\cos(i)\cos(\omega) & \sin(\Omega)\sin(i) \\ \sin(\Omega)\cos(\omega) + \cos(\Omega)\cos(i)\sin(\omega) & -\sin(\Omega)\sin(\omega) + \sin(\Omega)\cos(i)\cos(\omega) & -\cos(\Omega)\sin(i) \\ \sin(i)\sin(\omega) & \sin(i)\cos(\omega) & \cos(i) \end{bmatrix} \begin{bmatrix} x_o \\ y_o \\ z_o \end{bmatrix} \tag{2.47}$$



$$\begin{bmatrix} x_r \\ y_r \\ z_r \end{bmatrix} = \begin{bmatrix} \cos(\Omega_e T_e) & \sin(\Omega_e T_e) & 0 \\ -\sin(\Omega_e T_e) & \cos(\Omega_e T_e) & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} x_i \\ y_i \\ z_i \end{bmatrix}$$

Figure 2.8 The relationship between the rotating coordinate system (x_r, y_r, z_r) and the geocentric equatorial system (x_i, y_i, z_i). The equatorial plane coincides with plane of the paper. The earth rotates counterclockwise with angular velocity Ω_e. The x_r and y_r axes are attached to the earth and rotate with it. The z_i and z_r axes coincide.

Orbital Mechanics

Orbital Elements

From Equation of Motion of: Second-order Linear Differential Equations

satellite communications: eccentricity (e), semimajor axis (a), time of perigee (t_p), right ascension of ascending node (Ω), inclination (i), and argument of perigee (ω). Frequently the mean anomaly (M) at a given time is substituted for t_p.

Orbital Mechanics

Subsatellite Point

North Latitude of SubSat Point

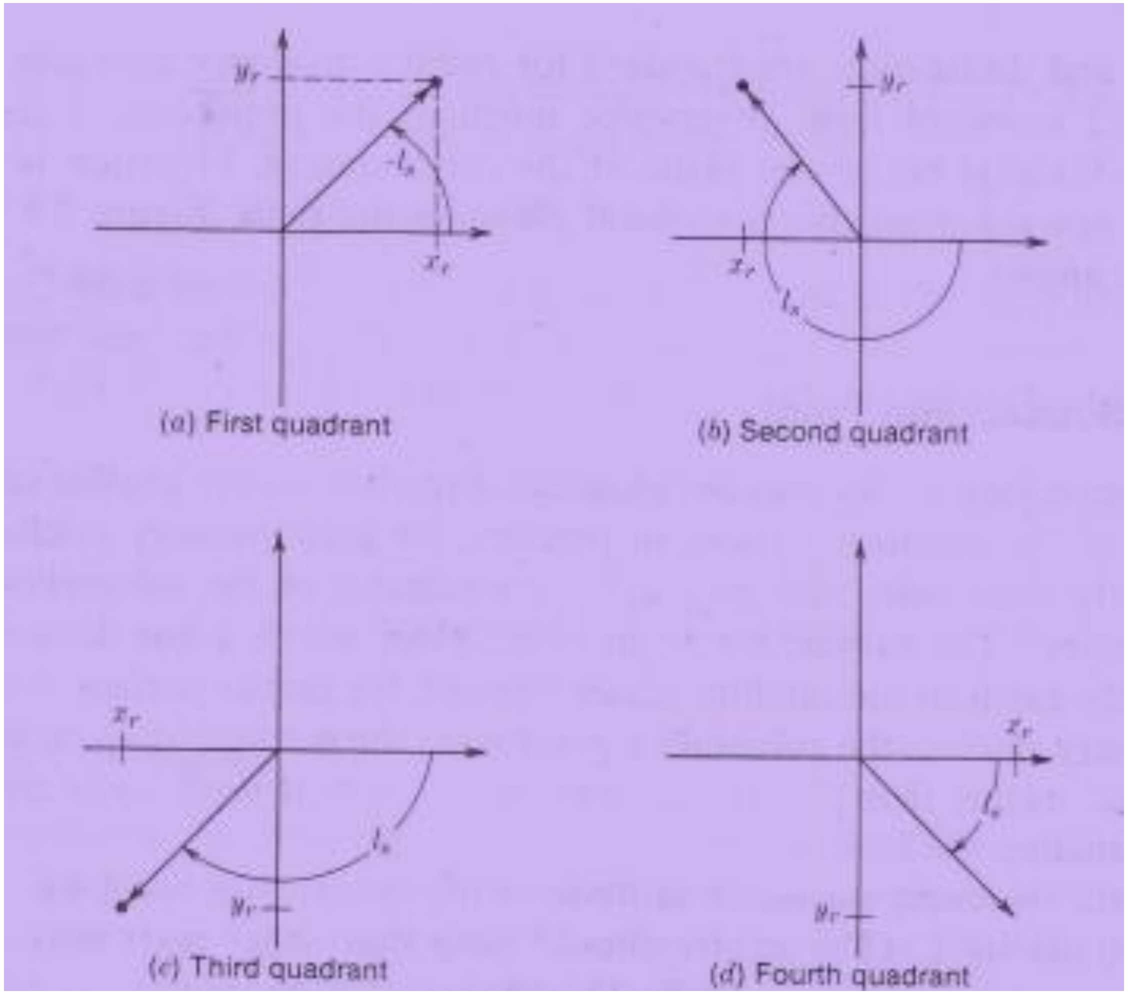
West longitude

$$L_s = 90^\circ - \cos^{-1}\left[\frac{z_r}{(x_r^2 + y_r^2 + z_r^2)^{\frac{1}{2}}}\right] \tag{2.52}$$

The equation used for calculating the subsatellite longitude l_s depends on the quadrant in which the point (x_r, y_r) lies. Its value in degrees west is given by

$$l_s = \begin{cases} -\tan^{-1}\left(\dfrac{y_r}{x_r}\right), & y_r \geq 0 \quad \text{and} \quad x_r \geq 0 \text{ (first quadrant)} \\ 180^\circ + \tan^{-1}\left(\dfrac{y_r}{|x_r|}\right), & y_r \geq 0 \quad \text{and} \quad x_r \leq 0 \text{ (second quadrant)} \\ 90^\circ + \tan^{-1}\left|\dfrac{x_r}{y_r}\right|, & y_r \leq 0 \quad \text{and} \quad x_r \leq 0 \text{ (third quadrant)} \\ \tan^{-1}\left(\dfrac{|y_r|}{x_r}\right), & y_r \leq 0 \quad \text{and} \quad x_r \geq 0 \text{ (fourth quadrant)} \end{cases} \tag{2.53}$$

Orbital Mechanics



Orbital Mechanics

Look Angle Determination

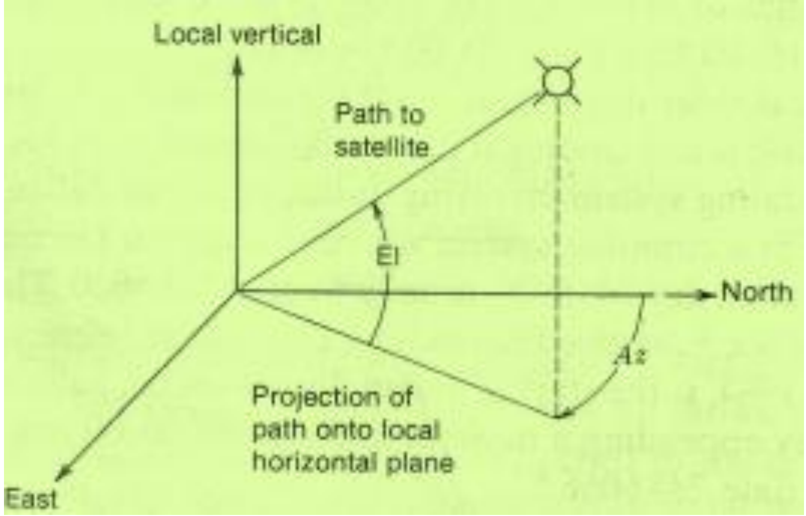


Figure 2.9 The definitions of azimuth (Az) and elevation (El). Elevation is measured upward from local horizontal and azimuth is measured from north eastward to the projection of the satellite path onto the local horizontal plane.

Orbital Mechanics

Elevation Calculation:

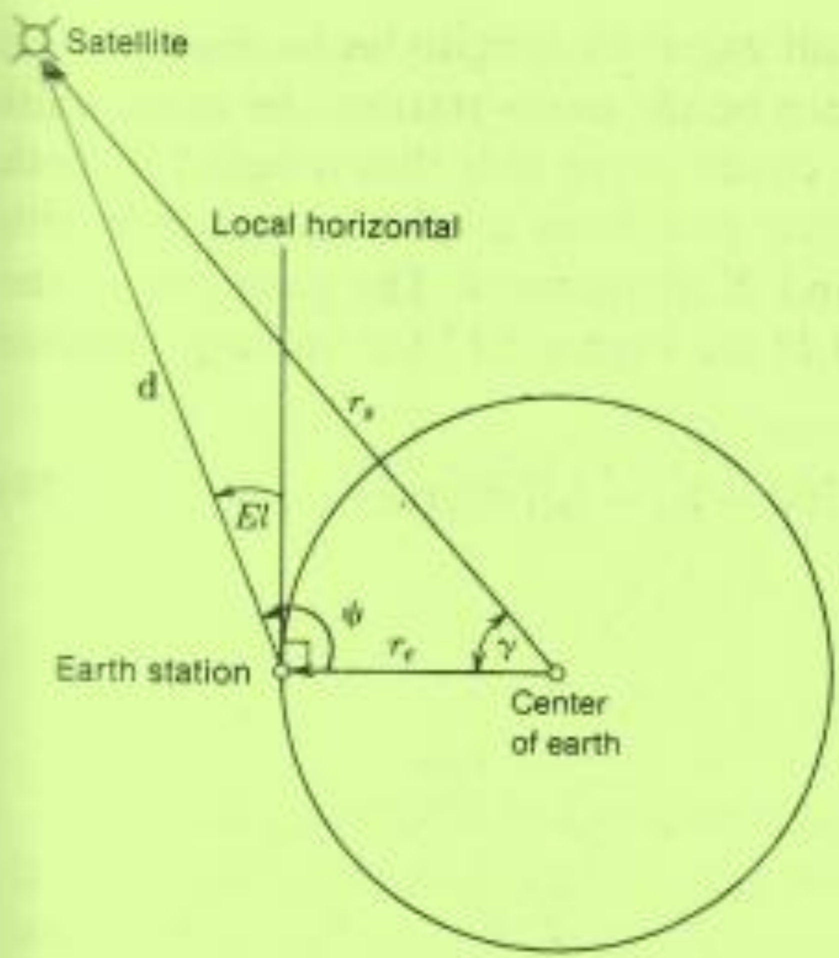


Figure 2.11 The geometry of elevation angle calculation. The plane of the paper is the plane defined by the center of the earth, the satellite, and the earth station. The central angle is γ. The elevation angle *El* is measured up from local horizontal to the satellite path.

Orbital Mechanics

$$\cos(\gamma) = \cos(L_e)\cos(L_s)\cos(l_s - l_e) + \sin(L_e)\sin(L_s) \qquad (2.54)$$

The magnitudes of the vectors joining the center of the earth, the satellite, and the earth station are related by the law of cosines. Thus

$$d = r_s\left[1 + \left(\frac{r_e}{r_s}\right)^2 - 2\left(\frac{r_e}{r_s}\right)\cos(\gamma)\right]^{\frac{1}{2}} \qquad (2.55)$$

Since the local horizontal plane at the earth station is perpendicular to $\mathbf{r}_e$, the elevation angle El is related to the central angle ψ by

$$El = \psi - 90^\circ \qquad (2.56)$$

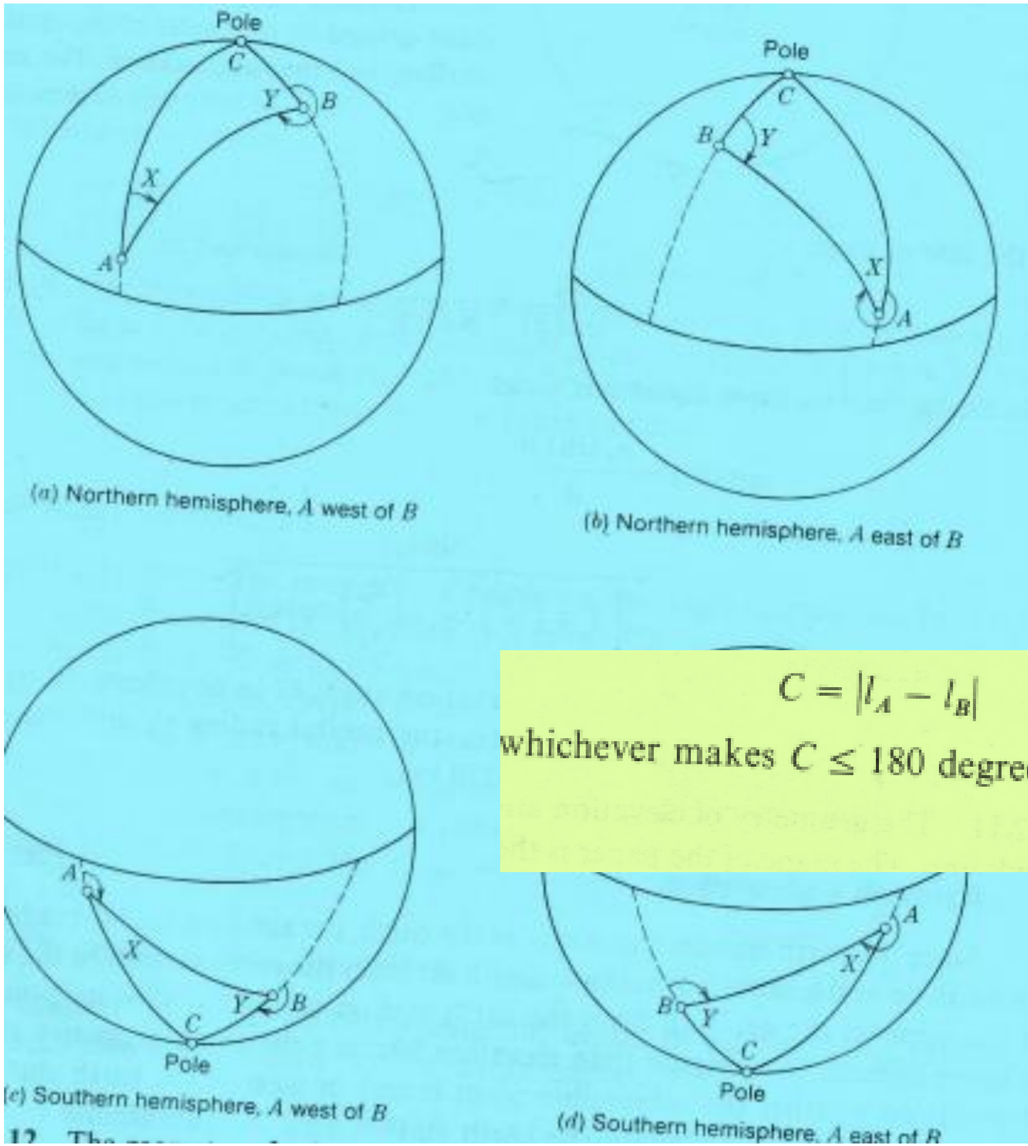
$$\cos(El) = \frac{r_s \sin(\gamma)}{d}$$

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

Equations (2.58) and (2.54) permit the elevation angle El to be calculated from the subsatellite and earth station coordinates, the orbital radius r_s, and the earth's radius r_e. A good value for the last is 6370 km.

Orbital Mechanics

Azimuth Calculation



$$C = |l_A - l_B| \quad \text{or} \quad |360 - |l_A - l_B|| \text{ degrees} \tag{2.59}$$

whichever makes $C \leq 180$ degrees.

Orbital Mechanics

If at least one point is in the northern hemisphere, B must be chosen so that it is closer to the north pole than A, making $L_B > L_A$. Then bearings X and Y may be found from

$$\tan[0.5(Y - X)] = \frac{\cot(0.5C)\sin[0.5(L_B - L_A)]}{\cos[0.5(L_B + L_A)]} \tag{2.60}$$

$$\tan[0.5(Y + X)] = \frac{\cot(0.5C)\cos[0.5(L_B - L_A)]}{\sin[0.5(L_B + L_A)]} \tag{2.61}$$

$$X = 0.5(Y + X) + 0.5(Y - X) \tag{2.62}$$

$$Y = 0.5(Y + X) - 0.5(Y - X) \tag{2.63}$$

If both points are in the southern hemisphere, then point B must be closer to the south pole making $L_B < L_A$ but $|L_B| > |L_A|$. Then Eqs. (2.60) and (2.61) become

$$\tan[0.5(Y - X)] = \frac{\cot(0.5C)\sin[0.5(|L_B| - |L_A|)]}{\cos[0.5(|L_B| + |L_A|)]} \tag{2.64}$$

$$\tan[0.5(Y + X)] = \frac{\cot(0.5C)\cos[0.5(|L_B| - |L_A|)]}{\sin[0.5(|L_B| + |L_A|)]} \tag{2.65}$$

Orbital Mechanics

Special Case: Geo Sat

For most geosynchronous satellites the subsatellite point is at the equator at longitude l_s, and latitude L_s is 0. The geosynchronous radius r_s is 42,242 km. Equation (2.54) for the central angle γ simplifies to

$$\cos(\gamma) = \cos(L_e)\cos(l_s - l_e) \tag{2.66}$$

The distance d from the earth station to the satellite is given by

$$d = 42{,}242[1.02274 - 0.301596\cos(\gamma)]^{1/2}\ \text{km} \tag{2.67}$$

The elevation angle El is given by

$$\cos(El) = \frac{\sin(\gamma)}{[1.02274 - 0.301596\cos(\gamma)]^{1/2}} \tag{2.68}$$

Orbital Mechanics

$$a = |l_s - l_e| \tag{2.69}$$

$$c = |L_e - L_s| \tag{2.70}$$

If we call the half-perimeter of the triangle s,

$$s = 0.5(a + c + \gamma) \tag{2.71}$$

then the angle α at the vertex may be obtained from the equation [6, pp. 46–9ff]

$$\tan^2\left(\frac{\alpha}{2}\right) = \frac{\sin(s-\gamma)\sin(s-c)}{\sin(s)\sin(s-a)} \tag{2.72}$$

The azimuth Az is related to α by the equations that appear in Figure 2.13 and in Table 2.2.

Equations (2.69) through (2.72) may be combined in one form, which, although compact, tends to obscure the derivation.

$$\alpha = 2\tan^{-1}\left\{\frac{\sin(s-\gamma)\sin(s-|L_e|)}{\sin(s)\sin(s-|l_e-l_s|)}\right\}^{\frac{1}{2}} \tag{2.73}$$

Orbital Mechanics

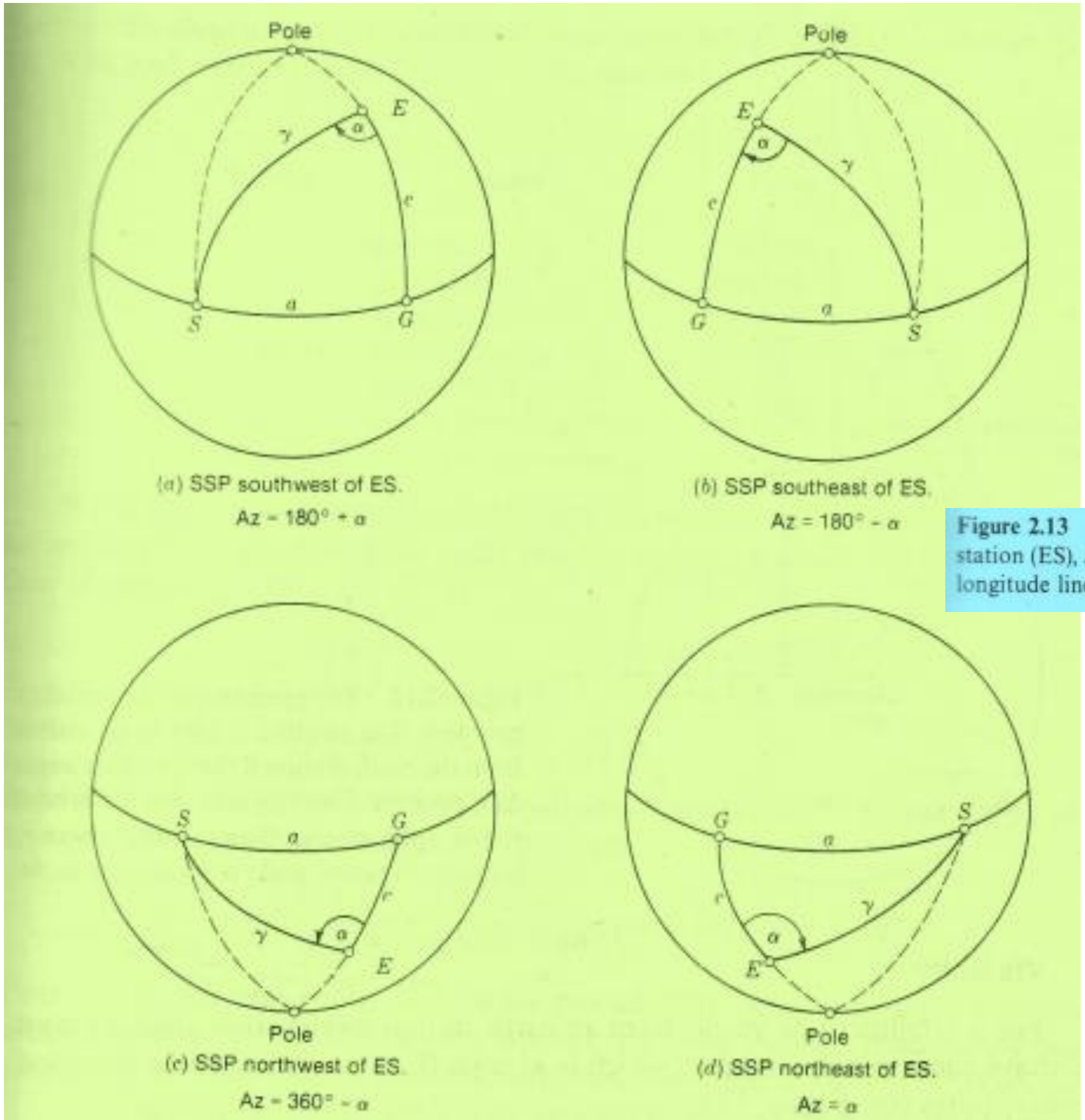


Figure 2.13 The spherical trigonometry of azimuth calculation. In all cases E is the earth station (ES), S is the subsatellite point (SSP), and G is the intersection of the earth station's longitude line with the equator. The arrow indicates the azimuth angle Az.

Orbital Mechanics

Visibility

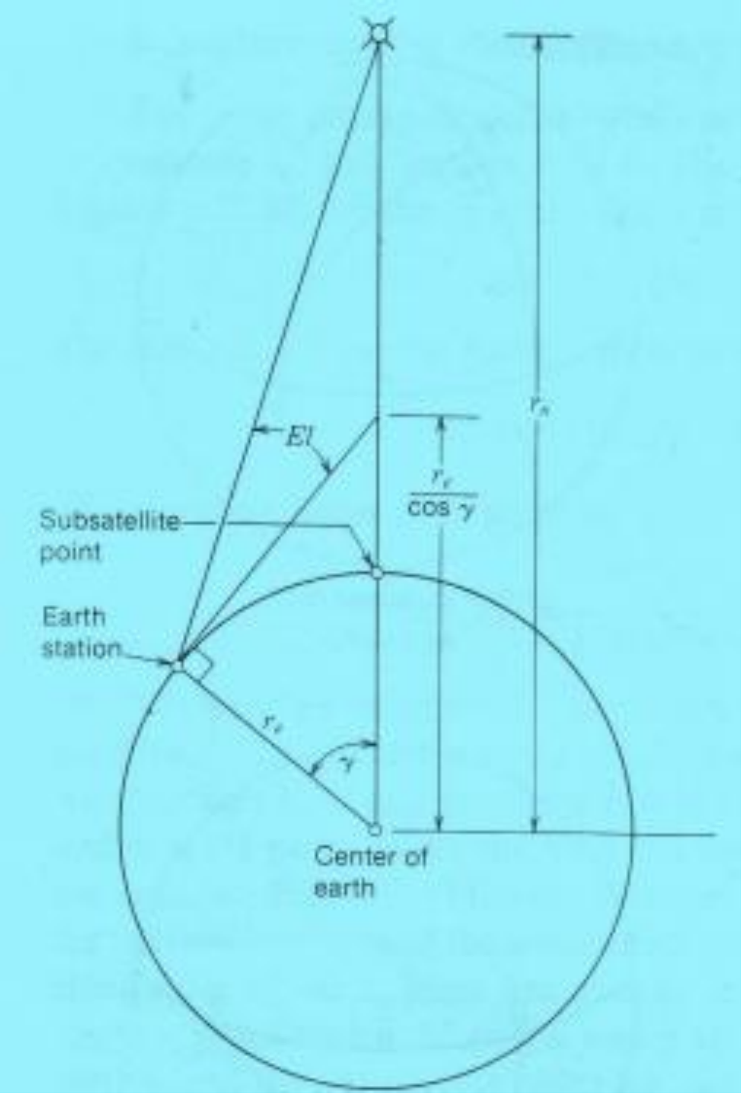


Figure 2.14 The geometry of the visibility problem. The satellite is said to be visible from the earth station if the elevation angle El is positive. This requires that the orbital radius r_s be greater than $r_e/\cos\gamma$ where r_e is the earth radius and γ is the central angle.

Orbital Mechanics

For a satellite to be visible from an earth station its elevation angle *El* must be above some minimum value, which is at least 0°. A positive or zero elevation angle requires (see Figure 2.14).

$$r_s \geq \frac{r_e}{\cos(\gamma)} \tag{2.74}$$

This means that the maximum central angular separation between the earth station and the subsatellite point is limited by

$$\gamma \leq \cos^{-1}\left(\frac{r_e}{r_s}\right) \tag{2.75}$$

For a nominal geosynchronous orbit, the last equation reduces to $\gamma \leq 81.3°$ for visibility. To avoid the propagation problems associated with extremely low elevation angles (see Chapter 8), a smaller angular separation is desirable.

Satellite Subsystems

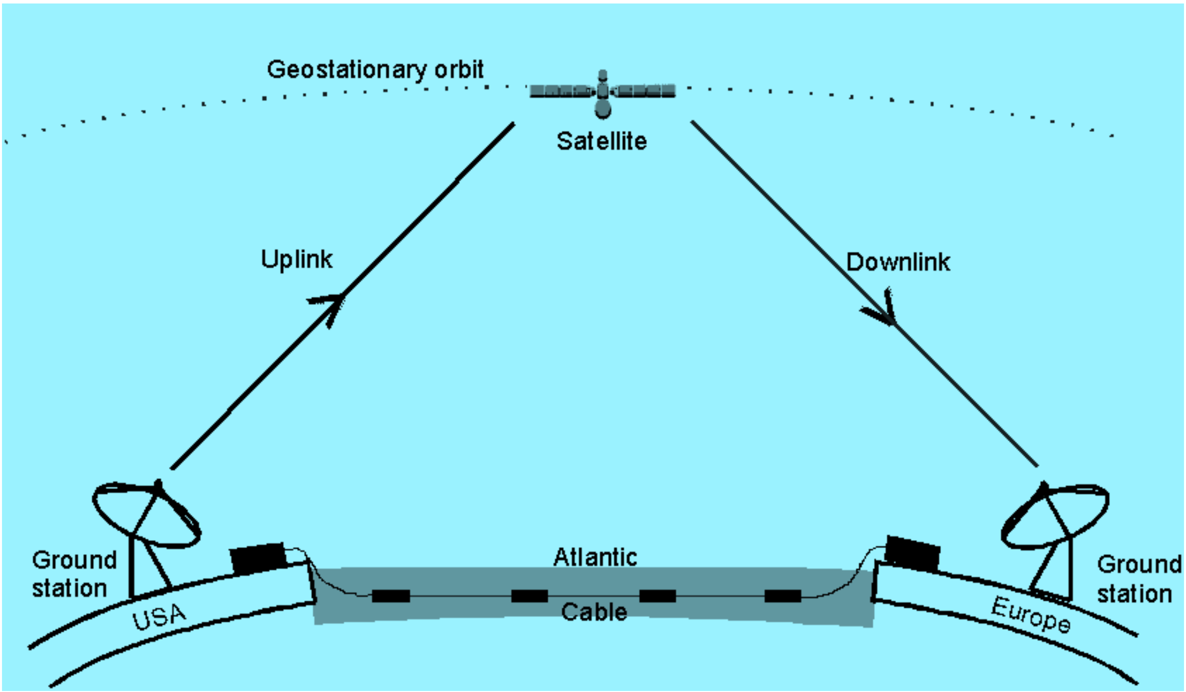
- **AOCS-Altitude and Orbit Control System**
  - Consists of: rocket motors used to move the sat back to correct the orbit when external forces cause it to drift off station
  - And, gas jets or inertial devices that control the altitude of the spacecraft

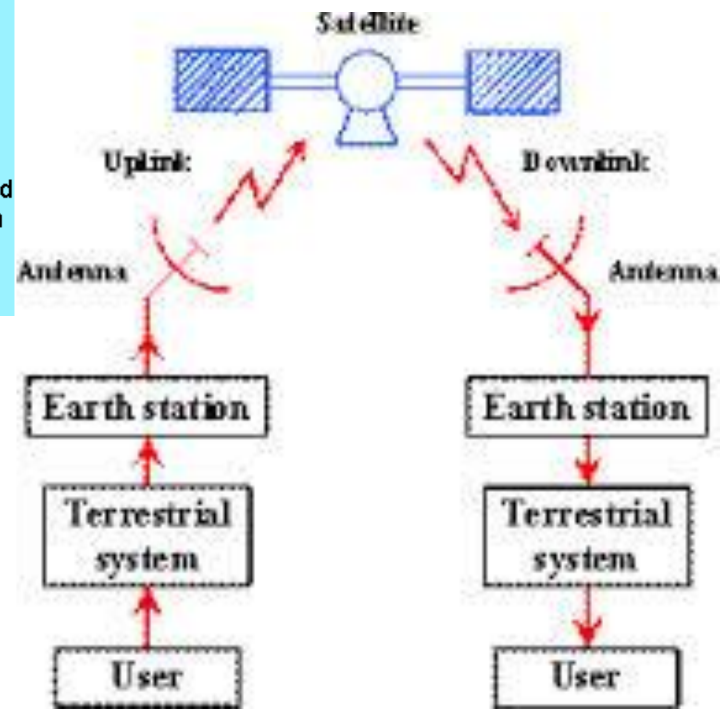
- **TT&C- Telemetry, Tracking & Command**
  - On Sat and earth
  - Data from many sensors on the spacecraft, tracking system on the earth provides info on the range, elevation and azimuth
  - Telemetry data from sat, orbital data from tracking system: control system is used to correct the position and altitude of the spacecraft.

- **Power System:** electrical power from solar cell.
- **Antenna:** Wire, Horn, Reflector and Array type.

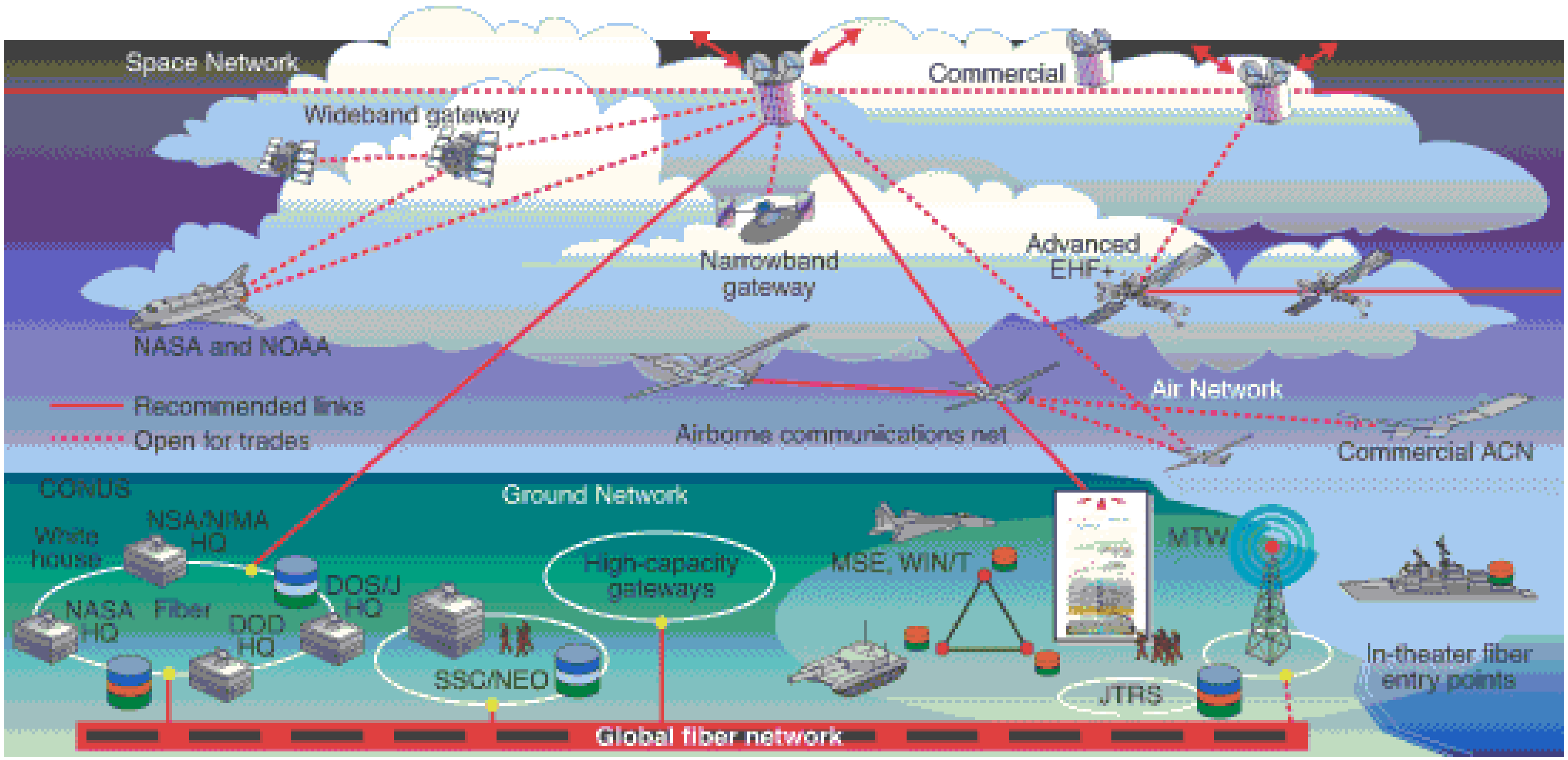
- **Communication subsystems**

Communication Subsystems

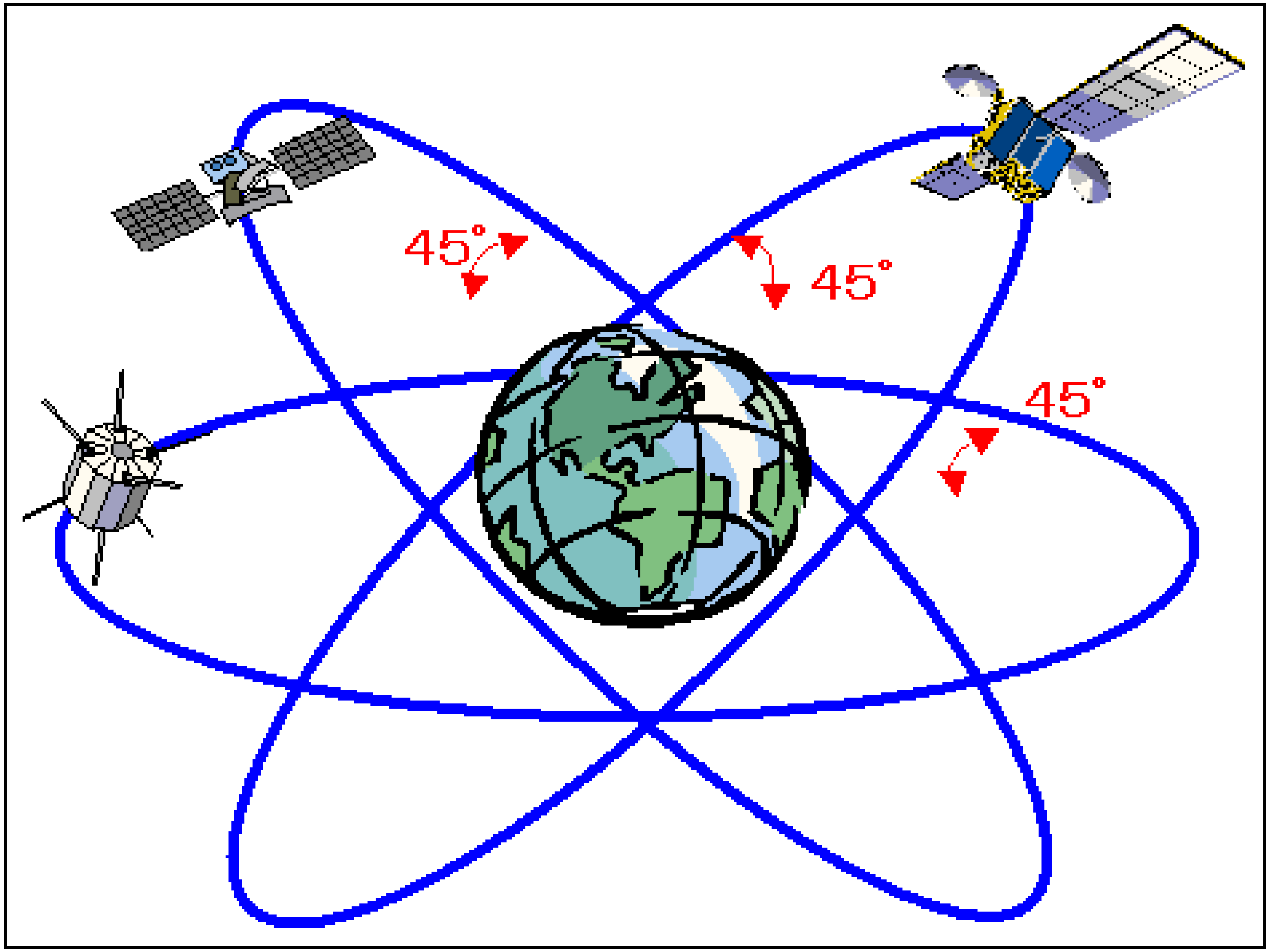




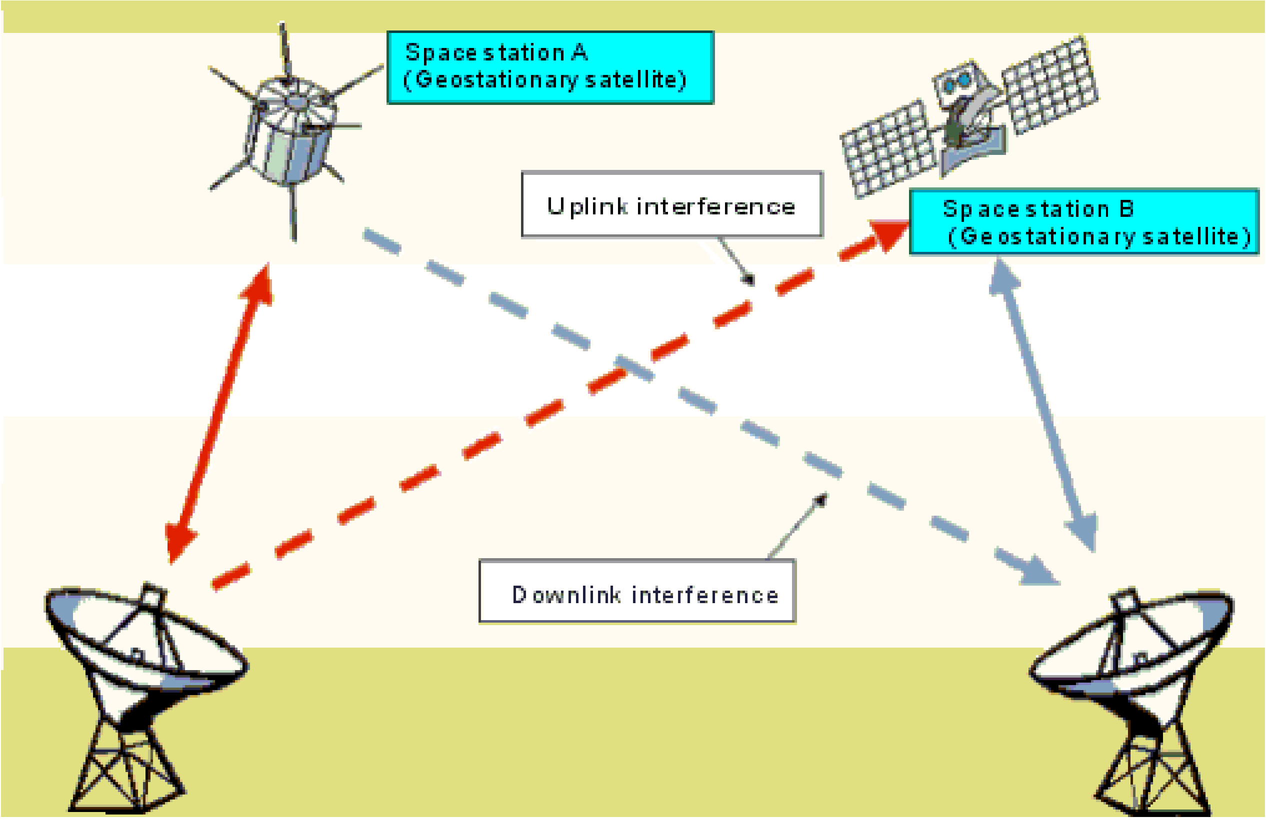
Communication Subsystems



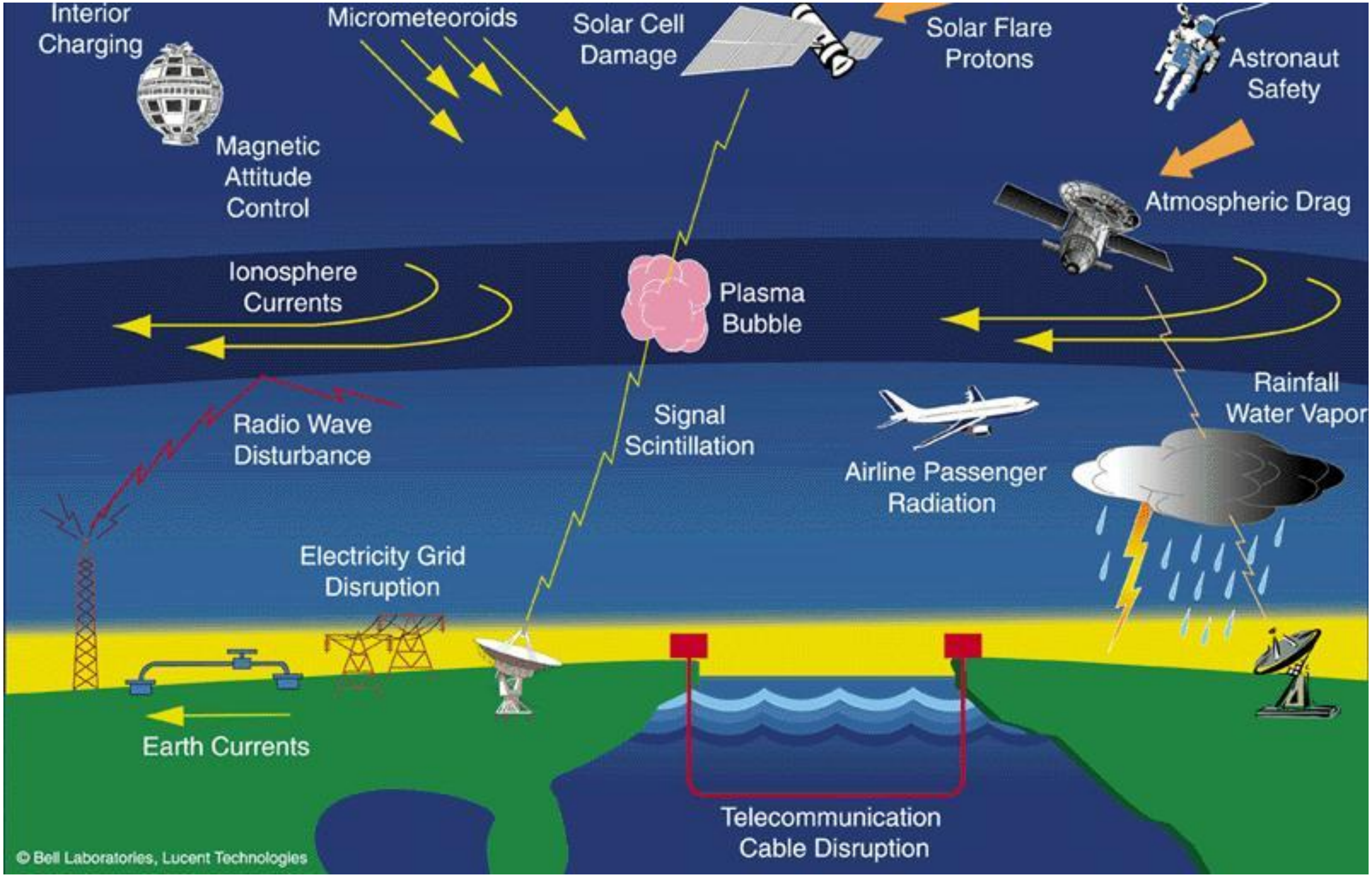
Communication Subsystems



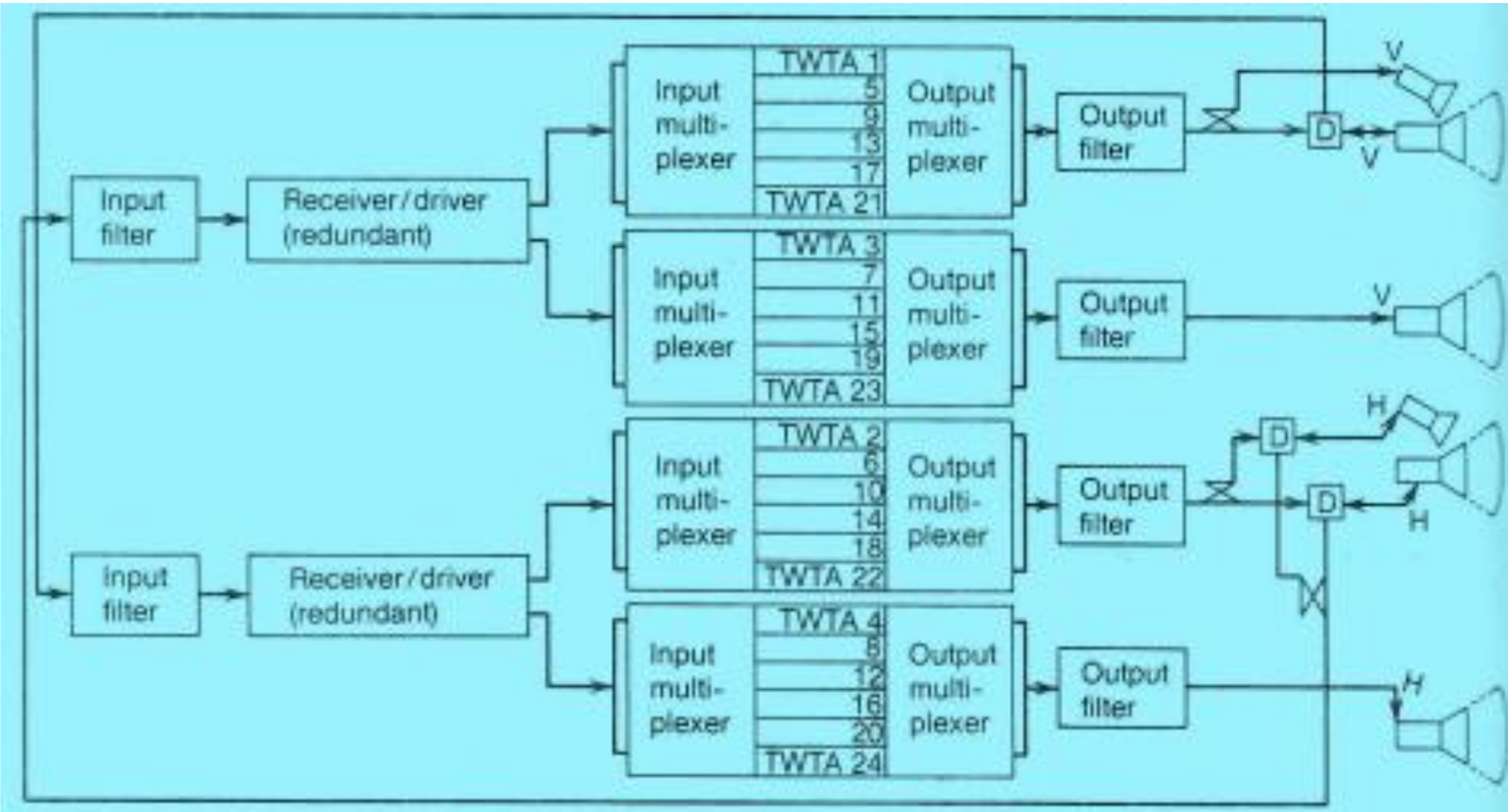
Communication Subsystems



Communication Subsystems



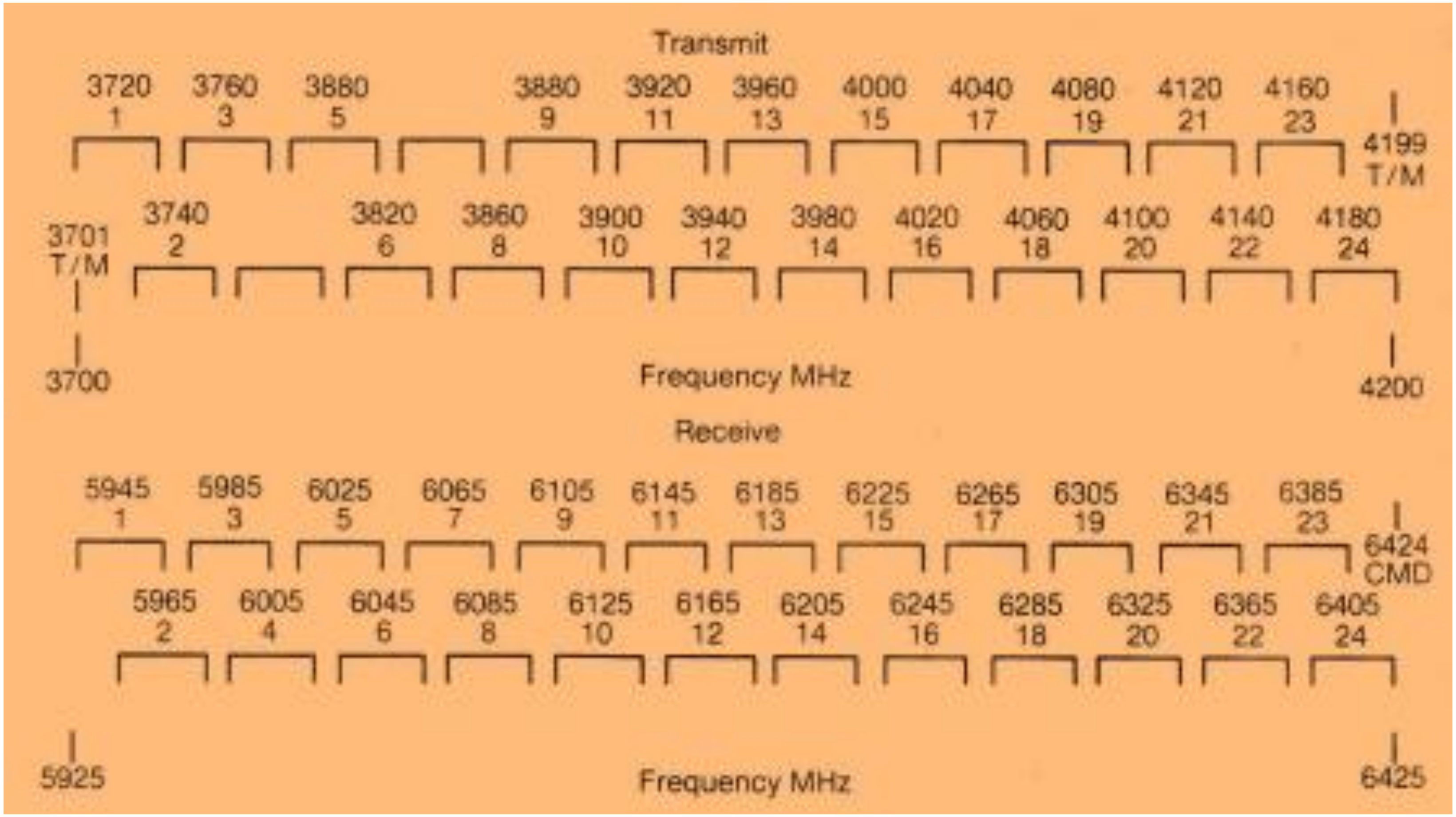
Communication Subsystems



Transponder arrangement

4/6 GHz

Communication Subsystems

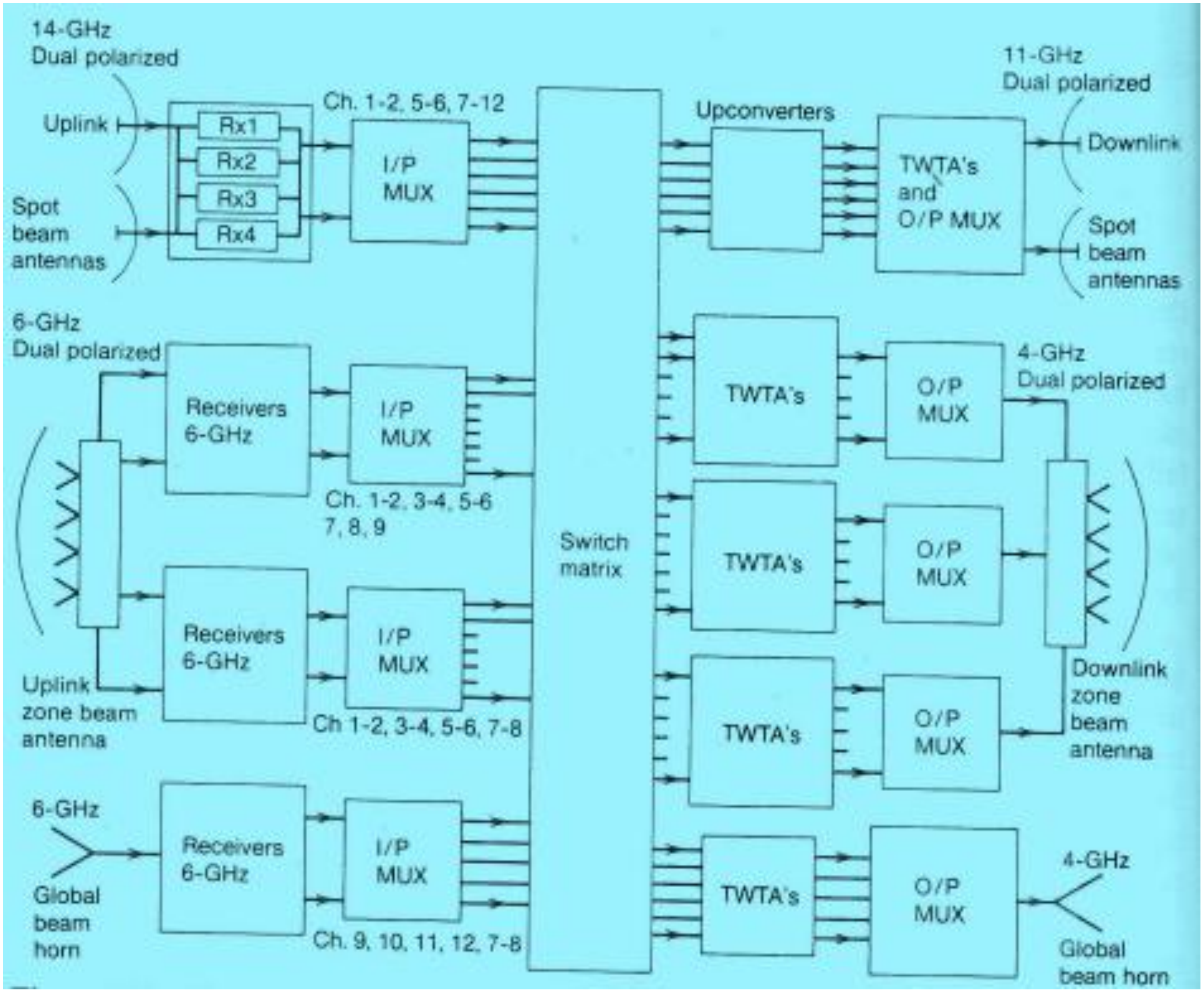


Frequency Plan

4/6 GHz

Communication Subsystems

INTELSAT V Communication System



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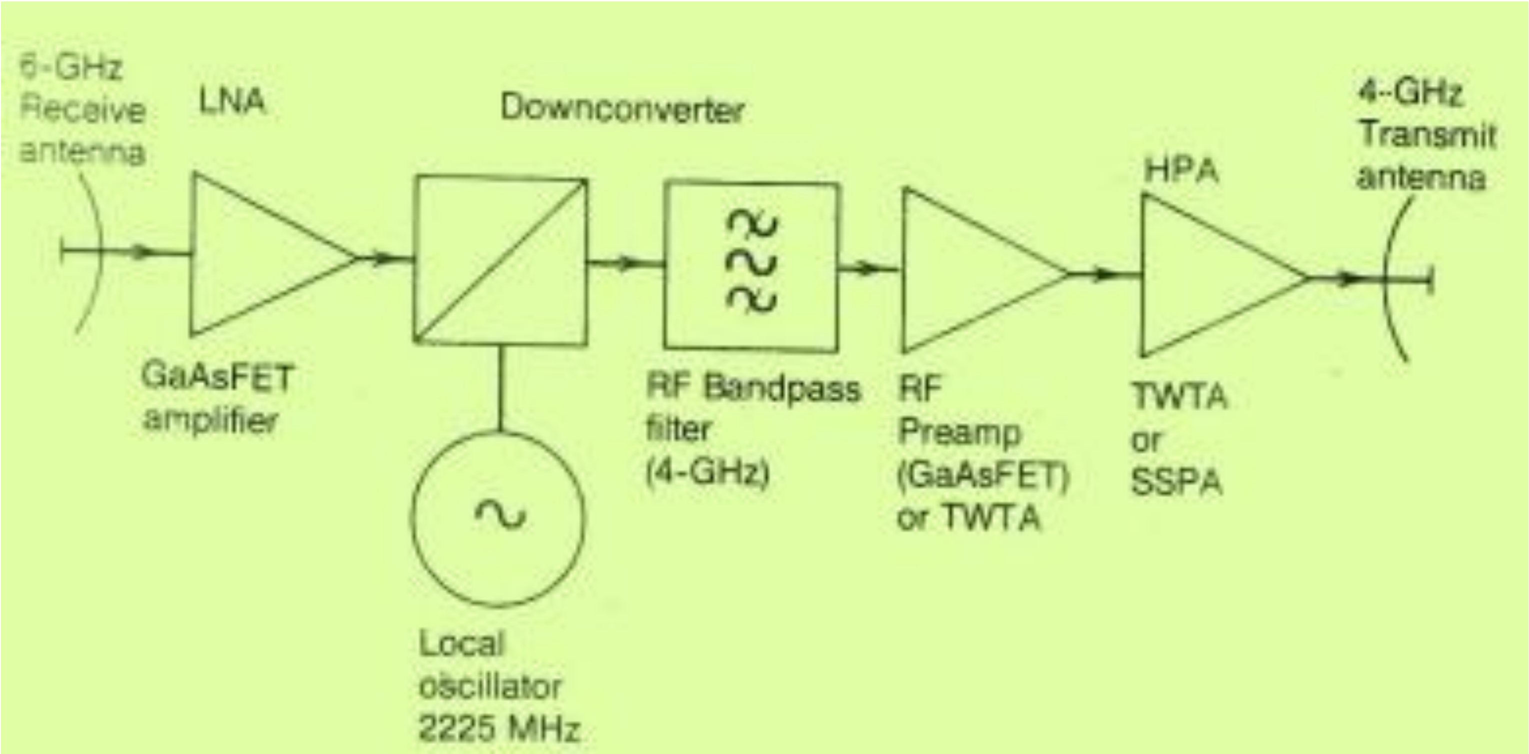
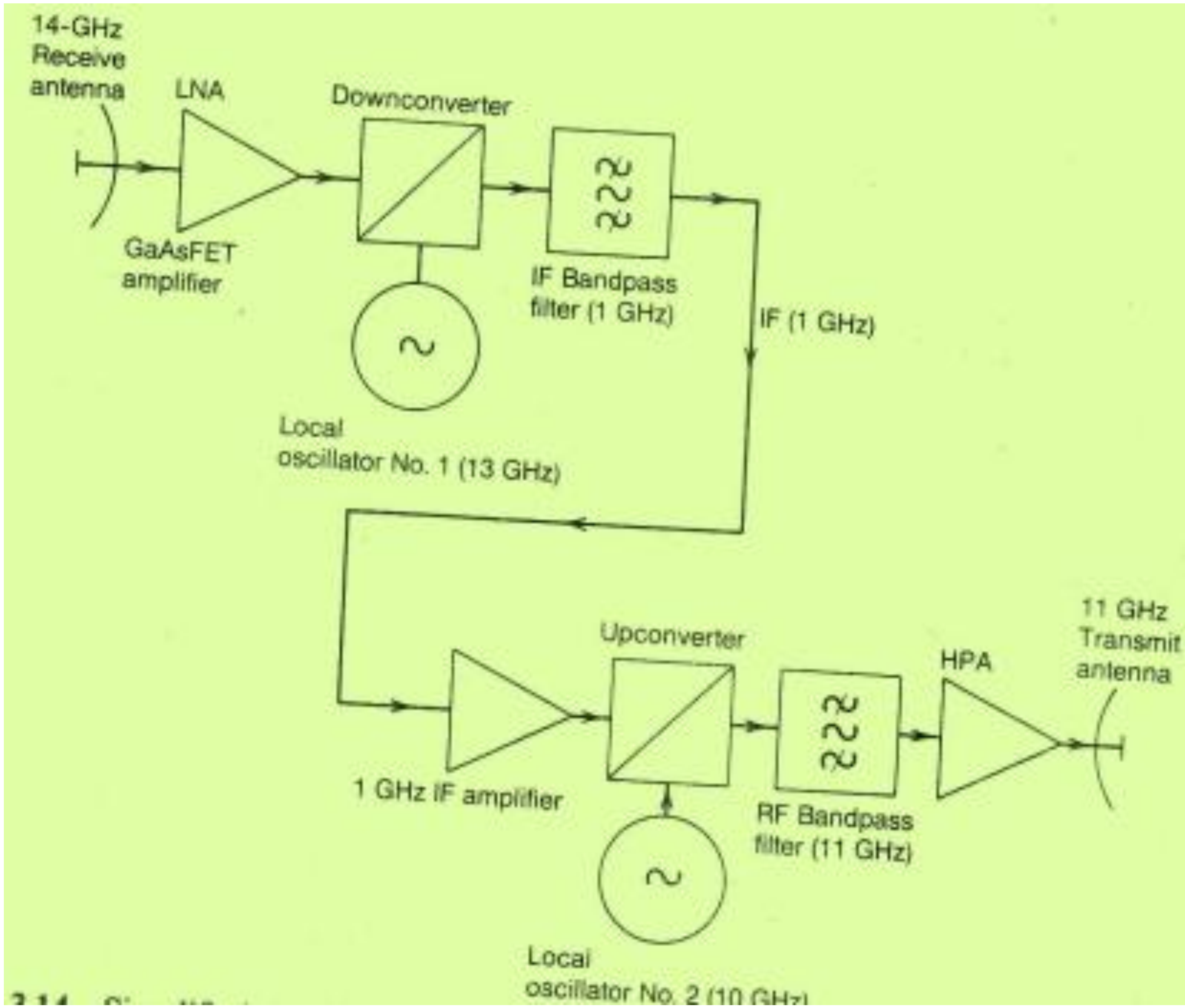


Single Conversion Transponder for 6/4 GHz

Communication Subsystems



Double Conversion Transponder for 14/11 GHz

Communication Subsystems