SATELLITE COMMUNICATIONS

SYLLABUS

UNIT-I:

Communication Satellite: Orbit and Description: A brief History of Satellite Communication, Satellite Frequency bands, Satellite Systems, Applications, Orbital Period and Velocity, Effects of Orbital inclination, Azimuth and Elevation, Coverage and Slant range, Eclipse, Orbital perturbations , Placement of a Satellite in a Geo-Stationary Orbit.

UNIT-II:

Satellite Sub-Systems: Altitude and orbit control system, TT&C Sub-System, Altitude control Sub-System, Power Systems, Communication Subsystems, Satellite antenna Equipment.

Satellite Link: Basic transmission theory, system noise temperature and G/T ratio, Basic Link Analysis, Interference Analysis, Design of satellite links for specified C/N, (with and without frequency Re-use), Link Budget.

UNIT-III:

Propagation effects: Introduction, Atmospheric Absorption, Cloud Attenuation, Tropospheric and Ionospheric Scintillation and Low angle fading, Rain Induced attenuation, rain induced cross polarization interference.

Multiple Access: Frequency Division Multiple Access(FDMA), Intermodulation, Calculation of C/N. Time Division Multiple Access(TDMA), Frame structure, Burst structure, Satellite Switched TDMA Onboard processing, Demand Assignment Multiple Access (DAMA) – Types of Demand Assignment, Characteristics, CDMA Spread Spectrum Transmission and Reception

UNIT-IV:

Earth Station Technology: Transmitters, Receivers, Antennas, Tracking systems, Terrestrial Interface, Power Test methods, Lower Orbit Considerations.

Satellite Navigation & Global Positioning Systems: Radio and Satellite Navigation, GPS Position Location principles, GPS Receivers, GPS C/A code accuracy, Differential GPS.

UNIT-V:

Satellite Packet Communications: Message Transmission by FDMA: M/G/1 Queue, Message Transmission by TDMA, PURE ALOHA-Satellite Packet Switching, Slotted Aloha, Packet Reservation, Tree Algorithm.

TEXT BOOKS:

1. Satellite Communications- Timothy Pratt, Charles Bostian and Jeremy Allnutt, WSE, Wiley Publications, 2nd Edition, 2003, John Wiley & Sons.
2. Satellite Communication Engineering- Wilbur L. Pritchand, Robert A Nelson and Henri G.Suyderhoud, 2nd Edition, Pearson Publications.
3. Digital Satellite Communications-Tri. T.Ha, 2nd Edition, 1990, Mc. Graw Hill.

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1. Satellite Communications- Dennis Roddy, 2nd Edition, 1996, McGraw Hill.
2. Satellite Communications: Design Principles- M. Richharia, 2nd Edition,BS Publications, 2003.
3. Digital Satellite Communications-Tri. T. Ha,2nd Ed.,MGH,1990.
4. Fundamental of Satellite Communications- K. N Raja Rao, PHI, 2004

Introduction

Satellites orbit around the earth. Depending on the application, these orbits can be circular or elliptical. Satellites in circular orbits always keep the same distance to the earth's surface.

How many countries have sent a satellite into space?

More than 75 countries have sent one or more satellite into space as of 2016. Only 10 of those countries have orbital launch capabilities. This means that they can use their own launch vehicles to deploy satellites. The remaining countries have launched their satellites by buying space on the launch vehicles of other countries that are capable of launching their own satellites.

How many countries have a space program?

There are six government space agencies.

1. RFSA or Roscosmos – Russian Federal Space Agency
2. ISRO – Indian Space Research Organisation
3. ESA – European Space Agency
4. CNSA – China National Space Administration
5. JAXA – Japan Aerospace Exploration Agency
6. NASA – National Aeronautics and Space Administration

Classification of Satellites based on Orbit Height

Low Earth Orbit (LEO) Satellites: These satellites are placed 500-1500 km above the surface of the earth. As LEOs circulate on a lower orbit, they exhibit a much shorter period that is about 95 to 120 minutes. LEO systems try to ensure a high elevation for every spot on earth to provide a high quality communication link. Each LEO satellite will only be visible from the earth for around ten minutes. LEOs even provide 2.4 kbps bandwidth for mobile terminals with Omni- directional antennas using low transmit power in the range of 1W. The delay for packets delivered via a LEO is relatively low (approx 10 ms). The delay is comparable to long-distance wired connections (about 5–10 ms). Smaller footprints of LEOs allow for better frequency reuse, similar to the concepts used for cellular networks. LEOs can provide a much higher elevation in Polar Regions and so better global coverage.

Medium Earth Orbit (MEO) Satellites: MEOs can be positioned somewhere between LEOs and GEOs, both in terms of their orbit and due to their advantages and disadvantages. Using orbits around 10,000 km, the system only requires a dozen satellites which is more than a GEO system, but much less than a LEO system. These satellites move more slowly relative to the earth's rotation allowing a simpler system design (satellite periods are about six hours). Depending on the inclination, a MEO can cover larger populations, so requiring fewer handovers. Due to the larger distance to the earth, channel delay increases to about 70–80 ms. the satellites need higher transmit power and special antennas for smaller footprints. There are five types of MEOs which are as given below:

- **Sun-Synchronous Orbit Satellites:** These satellites rise and set with the sun. Their orbit is defined in such a way that they are always facing the sun and hence they never go through an eclipse. For these satellites, the surface illumination angle will be nearly the same every time.
- **Hohmann Transfer Orbit:** This is an intermediate orbit having a highly elliptical shape. It is used by GEO satellites to reach their final destination orbits. This orbit is connected to the LEO orbit at the point of perigee forming a tangent and is connected to the GEO orbit at the point of apogee again forming a tangent.
- **Progragde Orbit:** This orbit is with an inclination of less than 90°. Its direction is the same as the direction as the rotation of the primary (planet) as shown in Fig 1.1.
- **Retrograde Orbit:** This orbit is with an inclination of more than 90°. Its direction is counter to the direction of rotation of the planet as shown in Fig 1.1. Only few satellites are launched into retrograde orbit because the quantity of fuel required to launch them is much greater than for a prograde orbit. This is because when the rocket starts out on the ground, it already has an eastward component of velocity equal to the rotational velocity of the planet at its launch latitude.
- **Polar Orbits:** This orbit passes above or nearly above both poles (north and South Pole) of the planet on each of its revolutions. Therefore it has an inclination of (or very close to) 90 degrees. These orbits are highly inclined in shape.

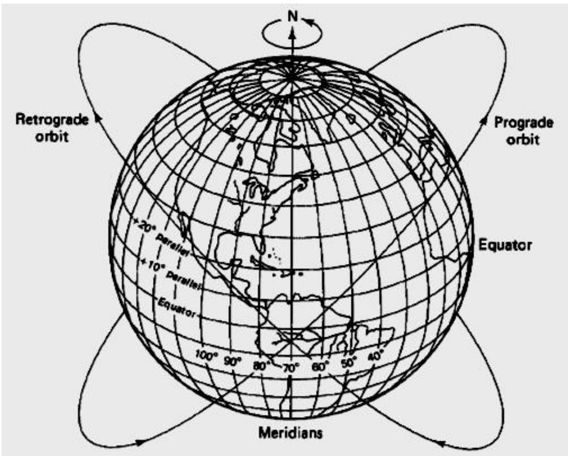

Fig1.1: Prograde and Retrograde Orbits

GEO (Geo Synchronous Earth Orbit): The satellites present in the geostationary orbit are called geostationary satellite. The geostationary orbit is one in which the satellite appears stationary relative to the earth. It lies in equatorial plane and inclination is '0'. The satellite must orbit the earth in the same direction as the earth spin. The orbit is circular.

Disadvantages of GEO: Northern or southern regions of the Earth (poles) have more problems receiving these satellites due to the low elevation above a latitude of 60°, i.e., larger antennas are needed in this case. Shading of the signals is seen in cities due to high buildings and the low elevation further away from the equator limit transmission quality. The transmit power needed is relatively high which causes problems for battery powered devices. These satellites cannot be used for small mobile phones. The biggest problem for voice and also data communication is the high latency as without having any handovers, the signal has to at least travel 72,000 km. Due to the large footprint, either frequencies cannot be reused or the GEO satellite needs special antennas focusing on a smaller footprint. Transferring a GEO into orbit is very expensive.

Environmental Design Conditions

Different environmental conditions are encountered by a satellite during its mission. Some of them are mentioned below.

(a) Zero Gravity

In geostationary earth orbit, effect of earth's gravity is negligible thus making the "zero gravity" effect.

Disadvantage: This causes a problem for liquids to flow. The major issue of fuel is encountered. Thus an external provision has to be made to force the liquids to flow.

Advantage: Absence of gravity leads to operation of deployment mechanism used for stowing antennas and solar panels during the launch.

(b) Atmospheric Pressure and Temperature:

At geostationary earth orbit, atmospheric pressure is very low, thus making the thermal conditions negligible which further leads to the increase in friction between surfaces.

Thus additional lubricants are required to keep the satellite parts in motion.

Due to the presence of electronic components inside the satellite, pressure us the satellite is higher making the functioning of the inner components of the satellite more manageable.

Sun's heat also affects the external components of the satellite.

© Space Particles:

Besides planets, natural and artificial satellites, many other particles like cosmic rays, protons, electrons, meteoroids and manmade space debris exists in space. These particles collide with the satellites causing permanent damage to it and sometimes degrading the solar cells.

Space debris, also known as orbital debris, space junk and space waste, is the collection of objects in orbit around Earth that were created by humans but no longer serve any useful purpose. These objects consist of everything from spent rocket stages and defunct satellites to explosion and collision fragments.

The debris can include slag and dust from solid rocket motors, surface degradation products such as paint flakes, clusters of small needles, and objects released due to the impact of micrometeoroids or fairly small debris onto spacecraft. As the orbits of these objects often overlap the trajectories of spacecraft, debris is a potential collision risk.

The vast majority of the estimated tens of millions of pieces of space debris are small particles, like paint flakes and solid rocket fuel slag. Impacts of these particles cause erosive damage, similar to sandblasting. The majority of this damage can be mitigated through the use of a technique originally developed to protect spacecraft from micrometeorites, by adding a thin layer of metal foil outside of the main spacecraft body.

Impacts take place at such high velocities that the debris is vaporized when it collides with the foil, and the resulting plasma spreads out quickly enough that it does not cause serious damage to the inner wall. However, not all parts of a spacecraft may be protected in this manner, i.e. solar panels and optical devices (such as telescopes, or star trackers), and these components are subject to constant wear by debris and micrometeorites.

The present means for spacecraft shielding, such as those used for the manned modules of the International Space Station are only capable of protecting against debris with diameters below about 1 cm. The only remaining means of protection would be to maneuver the spacecraft in order to avoid a collision. This, however, requires that the orbit of the respective object be precisely known.

If a collision with larger debris does occur, many of the resulting fragments from the damaged spacecraft will also be in the 1 kilogram mass range, and these objects become an additional collision risk.

As the chance of collision is a function of the number of objects in space, there is a critical density where the creation of new debris occurs faster than the various natural forces that remove these objects from orbit. Beyond this point a runaway chain reaction can occur that quickly reduces all objects in orbit to debris in a period of years or months.

(d) Magnetic Fields:
Due to the magnetic field of earth, charged particles which are trapped in the surrounding region of the earth get deflected.

This effect is more seen in the layers around the equator where the magnetic power of the earth is of maximum effect. This region is called the Van Allen’s Belt.

Even though satellites in geostationary earth orbit are not really affected by the earth’s magnetic field, they have to pass through the Van Allen’s belt during orbit raising (launching).

The electric charges present in this belt affect the electronic components against radiation. To overcome this effect, large coils are used by satellites.

Communication Design Considerations
For telecommunication satellite, the main design considerations are:

- i) Type of service to be provided
- ii) Communication capacity
- iii) Coverage area
- iv) Technological limitations

Depending upon the type of service to be provided by the satellite, certain basic specifications are laid down.

For domestic fixed satellite services, the main parameters are EIRP (Equivalent Isotropically Radiated Power) per carrier, number of carriers and the assigned coverage area.

For direct broadcast satellites, the number of television channels and coverage area is specified.

Based on these parameters, satellites are designed to fulfill the areas needs and at the same time it should be made in the specified cost fulfilling all the technical constraints.

While developing a satellite, the earth station's previous experience and in-house capabilities are also taken into account.

Often, for same set of requirements, different types of configurations are often proposed.

Communication Satellites

- Satellites are specifically made for telecommunication purpose. They are used for mobile applications such as communication to ships, vehicles, planes, hand-held terminals and for TV and radio broadcasting.
- They are responsible for providing these services to an assigned region (area) on the earth. The power and bandwidth of these satellites depend upon the preferred size of the footprint, complexity of the traffic control protocol schemes and the cost of ground stations.
- A satellite works most efficiently when the transmissions are focused with a desired area. When the area is focused, then the emissions don't go outside that designated area and thus minimizing the interference to the other systems. This leads more efficient spectrum usage.
- Satellite's antenna patterns play an important role and must be designed to best cover the designated geographical area (which is generally irregular in shape). Satellites should be designed by keeping in mind its usability for short and long term effects throughout its life time.
- The earth station should be in a position to control the satellite if it drifts from its orbit it is subjected to any kind of drag from the external forces.

Lifetime

The useful lifetime of a geostationary satellite is determined by the highest tolerable deviation in inclination and orbit location together with reliability of satellite's critical sub-system.

A lifetime could be improved by increasing the fuel capacity and by saving fuel by accepting orbital deviation to the maximum extent that is possible. Saving fuel couldn't be implemented to a great level. So for this purpose propulsion is used. The life period of Communication satellites is usually 10-15 years.

Reliability

Reliability is counted by considering the proper working of satellites critical components. Reliability could be improved by making the critical components redundant. Components with a limited lifetime such as travelling wave tube amplifier etc should be made redundant.

Travelling Wave Tube Amplifier (TWTA): travelling wave tube amplifiers have applications in both receiver and transmitter systems, and come in all shapes and sizes, but they all consist of three basic parts-the tube, the tube mount (which includes the beam focussing magnets) and the power supply. The main attraction of these devices is their very high gain (30-60 dB), linear characteristics and 1-2 octave bandwidth. They are quite widely used professionally, but are still rather scarce in amateur circles.

When used as receiver RF amplifiers they are characterized by high gain, low noise figure and wide bandwidth, and are known as low noise amplifiers (LNAS). These usually come with tube, mount and power supply in one integral unit, with no external adjustments to make-just input socket, output socket and mains supply connections. A typical LNA has an octave bandwidth (eg 2-4 GHz), 30 dB gain, 8 dB noise figure, and a saturated power output of 10 mW, within a volume of 2 in X 2 in X 10 in.

Transmitter TWTAs are naturally somewhat bulkier, and often have the power or supplies as a separate unit. Medium-power tubes have outputs of up to about 10 W, while high-power tubes deliver several hundred watts. Such tubes have gains of the order of 30 or 40 dB, and bandwidths of up to an octave

Other critical components are antenna reflectors, beaming assemblers etc.

A reliability model is used to calculate the satellite's reliability. It is defined as "the probability that a given component or system performs its functions as desired within a specific time t.

The failure rate for all components is calculated and they are categorized into the following three categories:

Early high failure rate region: used for manufacturing faults, defects in material etc.

- Low failure: used for random component failure.
- High failure rate: used for components weave-out.

Certainly early failures criteria is eliminated as most of the components are tested before used in the satellite. Random failures are more seen. They could be reduced by using reliable engineering techniques.

The life-spam of component could be increased by improving manufacturing techniques and the type of material used to reduce the number of worn out parts and hence reducing the high failure rate criteria.

It is sent that the failure rate is constant over time and is looking at this reliability can be determined. The system is made of several components, connected in a series, then the overall reliability is determined. By duplicating the less reliable and critical components, the overall reliability of the system could be improved. If any failure occurs in operational unit, then the standby unit takes over to develop a system with redundant components, its redundant elements are considered in parallel.

Parallel redundancy is useful when the reliability of an individual sub-system is high.

Example: consider a system having *i* parallel components in which reliability of each element is independent of others.

If Qi is the unreliability of the $\mathbf{i^{th}}$ parallel element, then the probability that all units will fail is the product of the individual un-reliabilities:

$Qs = Q1\ Q2\ Q3\ \ldots.\ Qi$

When the un-reliability of all elements is equal, then Qs = Qi where Q is the un-reliability of each element.

By doing a complete failure analysis, one could find out which failure occurs more than the rest and such analysis help in finding out the manufacturing defects in the product of a given batch of components or probably a design defect.

This analysis is done to reduce the overall reliability to a value less than that predicted by the above analysis.

Co-related failures could also be reduced by using units from different manufacturers. The design defects are generic to all satellite produced in a series. Generally these defects are detected and corrected to minimize their impact. This is done when a complete design change cannot be implemented.

Even through the reliability can be improved by adding redundant devices and components, the weight of the satellite increases which again becomes a problem. Redundant component also increase the cost of the satellite.

The two major cost components are:

- Cost of equipment together with the switching and failure sensing mechanism used.
- The associated increase in weight of the satellite resulting in an increased launch cost.

Optimization techniques are performed for cost minimization purpose.

Applications of Communication Satellites

- Mobile & Data Communications
- TV Broadcasting
- Navigational Aids (GPS)
- Weather forecasting
- Remote Sensing
- Atmospheric Research

Frequency Allocations for Communication Satellites

Allocation of frequencies to satellite services s a complicated process which requires international coordination and planning. This is done as per the International Telecommunication Union (ITU). To implement this frequency planning, the world is divided into three regions:

- Region1: Europe, Africa and Mongolia
- Region 2: North and South America and Greenland
- Region 3: Asia (excluding region 1 areas), Australia and south-west Pacific.

Frequency Bands for Satellite Communication:-

BAND	DOWNLINK [MHz]	UPLINK [MHz]
UHF-military	250-270	292-312
C-commercial	3700-4200	5925-6425
X-military	7250-7750	7900-8400
Ku-commercial	11700-12200	14000-14500
Ka-commercial	17700-21200	27500-30000
Ka-military	20200-21200	43500-45500

Kepler's Laws

Kepler's First Law

'The orbit of every satellite is an ellipse with the Planet at one of the two foci'.

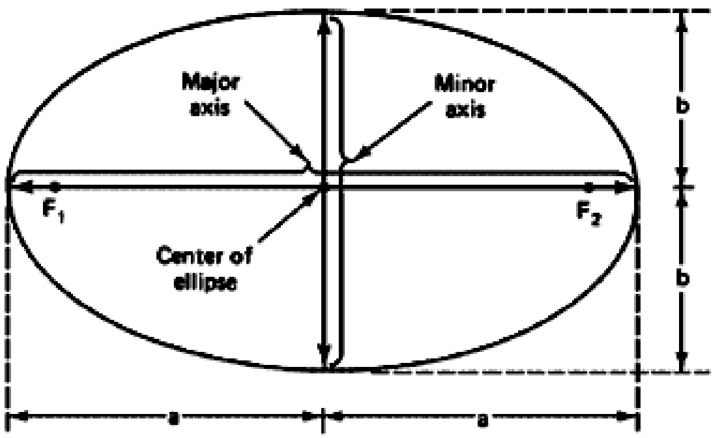

Fig 1.2: Elliptical Orbit shape

An ellipse is a particular class of mathematical shapes that resemble a stretched out circle. The Planet is not at the center of the ellipse but is at one of the focal points.
Ellipses have two focal points neither of which are in the center of the ellipse (except for the one special case of the ellipse being a circle). Circles are a special case of an ellipse that are not stretched out and in which both focal points coincide at the center.

Kepler's Second Law

"A line joining a planet and the Sun sweeps out equal areas during equal intervals of time."

To understand the second law let us suppose a planet takes one day to travel from point A to point B. The lines from the Sun to points A and B, together with the planet orbit, will define an (roughly triangular) area. This same area will be covered every day regardless of where in its orbit the planet is. Now as the first law states that the planet follows an ellipse, the planet is at different distances from the Sun at different parts in its orbit. So the planet has to move faster when it is closer to the Sun so that it sweeps an equal area as shown in Fig 1.3.

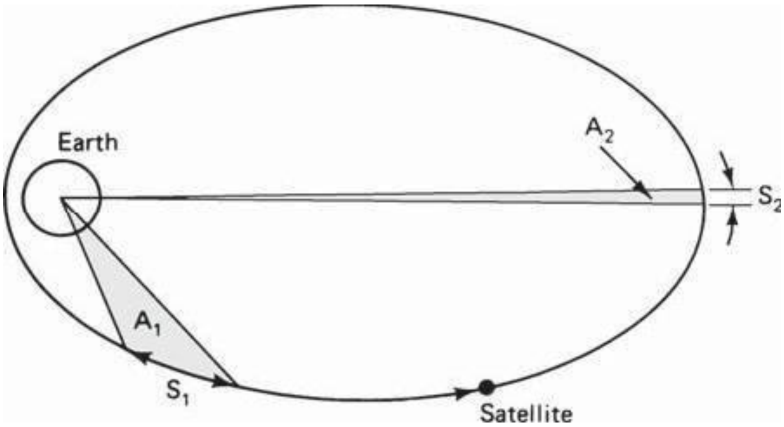

Figure 1.3: Kepler's Second law. The areas $A_1$and A_2 swept out in unit time are equal.

Kepler's second law is equivalent to the fact that the force perpendicular to the radius vector is zero. The "areal velocity" is proportional to angular momentum, and so for the same reasons, Kepler's second law is also in effect a statement of the conservation of angular momentum.

Symbolically:

$$\frac{d}{dt}[\frac{1}{2}r^2\frac{d\theta}{dt}] = 0$$

Where $\frac{1}{2}r^2\frac{d\theta}{dt}$ is the angular velocity of the satellite which is constant

Kepler's Third Law

"The square of the orbital period of a planet is directly proportional to the cube of the semi-major axis of its orbit."

$\mathbf{T^2} \alpha\, \mathbf{a^3}$ Where T is orbital period and a= semi major axis of orbit

The proportionality constant is **μ = 3.986005 x** 10^{14} m^3/sec^2

$\mathbf{a^3 = \mu / T^2}$

Or considering two satellites around the same planet then their orbit periods are related by

$$\frac{T_1^2}{T_2^2} = \frac{a_1^3}{a_2^3}$$

Definitions

Apogee: A point on the satellite's orbit, farthest from the Earth.

Perigee: A point on the satellite's orbit, closest from the Earth.

Line of Apsides: Line joining perigee and apogee through centre of the Earth. It is the major axis of the orbit. One-half of this line's length is the semi-major axis equivalents to satellite's mean distance from the Earth.

Ascending Node: The point where the orbit crosses the equatorial plane going from north to south.

Descending Node: The point where the orbit crosses the equatorial plane going from south to north.

Inclination: The angle between the orbital plane and the Earth's equatorial plane. Its measured at the ascending node from the equator to the orbit, going from East to North. Also, this angle is commonly denoted as **i.**

Line of Nodes: the line joining the ascending and descending nodes through the centre of Earth.

Prograde Orbit: an orbit in which satellite moves in the same direction as the Earth's rotation. Its inclination is always between 0^o to 90^o. Many satellites follow this path as Earth's velocity makes it easier to lunch these satellites.

Retrograde Orbit: an orbit in which satellite moves in the same direction counter to the Earth's rotation.

Argument of Perigee: An angle from the point of perigee measure in the orbital plane at the Earth's centre, in the direction of the satellite motion.

Right ascension of ascending node: The definition of an orbit in space, the position of ascending node is specified. But as the Earth spins, the longitude of ascending node changes and cannot be used for reference. Thus for practical determination of an orbit, the longitude and time of crossing the ascending node is used. For absolute measurement, a fixed reference point in space is required. It could also be defined as "*right ascension of the ascending node; right ascension is the angular position measured eastward along the celestial equator from the vernal equinox vector to the hour circle of the object*".

Mean anomaly: It gives the average value to the angular position of the satellite with reference to the perigee.

True anomaly: It is the angle from point of perigee to the satellite's position, measure at the Earth's centre.

Subsolar Point: It is the point of intersection of line joining centre of earth and centre of sun on the earth surface. It is expressed in terms of longitude and latitude.
Sub satellite Point: It is the point of intersection of line joining centre of earth and the satellite on the earth surface.

Equinoxes: Equinoxes are the only times when the Subsolar point is on the equator, meaning that the Sun is exactly overhead at a point on the equatorial line. At Equinox the Earth's orbital plane and equatorial plane intersect. The subsolar point crosses the equator moving northward at the March equinox (**Vernal Equinox**) and southward at the September equinox(**Autumnal Equinox**).

Right Ascention Angle: (abbreviated RA; symbol α) is the angular distance measured eastward along the celestial equator from the vernal equinox to the hour circle of the point in question

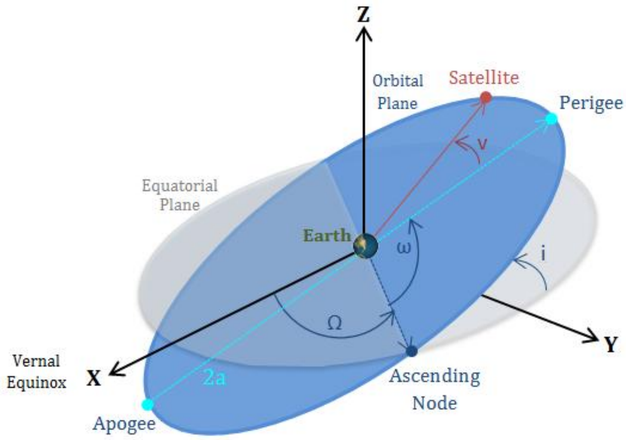

Figure 1.4: Argument of Perigee

Argument of Perigee 'ω': This is the angle measured along the orbit from the ascending node to the perigee in the X-Y plane.

First Point of Aeries: This is the direction of a line from the centre of earth through the centre of the sun at the vernal equinox (around Mar 21^{st})

Orbital Elements: Orbital elements are the parameters required to uniquely identify a specific orbit. There are many different ways to mathematically describe the same orbit, but certain schemes each consisting of a set of six parameters are commonly used in astronomy and orbital mechanics. To specify the coordinates (Inertial) of a satellite at any given time 't', we need to know six quantities known as orbital elements of a satellite.

1. Semi-Major axis (a)
2. Eccentricity (e)
3. Mean anomaly (M): It is the arc length covered by the satellite during the time $t-t_p$
4. Time of Perigee (t_p)
5. Inclination(i)
6. Right ascension of ascending node (Ω)

Universal Time

Universal time coordinate (UTC) is the time used for all civic time keeping purpose. Fundamental unit of UTC is ***mean solar day***. UTC is equivalent to Greenwich Mean Time (GMT) and the Zullu time (Z).
1 Mean Solar Day s divided into 24 hours; 1 Hour into 60 minutes;
1 Minute into 60 seconds.
Thus there are 86,400 seconds in a day.

Example: calculate time in days, hours, minutes and seconds for epoch day 324.95616765

Solution: Mean solar day = 324^{th} + 0.95616765 mean solar day Therefore;
24 x 0.95616765 = 22.948022
60 x 0.948022 = 56.881344
60 x 0.881344 = 52.88064

Thus; Epoch is at 22 hours 56 minutes a 52.88 seconds of the 324^{th} day of the year.

Julian Date

Generally time interval between two events is computed using calendar time or UT. But this notation is not suited for computations where timing of many events has to be computed. **Julian Day Number** (**JDN**) is the integer assigned to a whole solar day in the Julian day count starting from noon Universal Time, with Julian day number 0 assigned to the day starting at noon on January 1, 4713 BC, Proleptic Julian calendar

Thus creating a reference time in which all the events can be referred in decimal days is required. Such a time reference is provided by the Julian zero time reference, which is 12 noon (12:00 UT) on January 1, 4713 (it is a hypothetical starting point).

The decimal parts of a Julian date:
0.1 = 2.4 hours or 144 minutes or 8640 seconds
0.01 = 0.24 hours or 14.4 minutes or 864 seconds
0.001 = 0.024 hours or 1.44 minutes or 86.4 seconds
0.0001 = 0.0024 hours or 0.144 minutes or 8.64 seconds
0.00001 = 0.00024 hours or 0.0144 minutes or 0.864 seconds.

Calculation of JD from Gregorian Calender.

It is easy (with your calculator) to calculate the Julian Day Number of any date given on the Gregorian Calendar. The Julian Day Number so calculated will be for 0 hours, GMT, on that date. Here's how to do it:

1) Express the date as Y M D, where Y is the year, M is the month number (Jan = 1, Feb = 2, etc.), and D is the day in the month.

2) If the month is January or February, subtract 1 from the year to get a new Y, and add 12 to the month to get a new M. (Thus, we are thinking of January and February as being the 13th and 14th month of the previous year).

3) Dropping the fractional part of all results of *all multiplications and divisions*, let
A = Y/100
B = A/4
C = 2-A+B
E = 365.25 x (Y+4716)
F = 30.6001 x (M+1)
JD= C+D+E+ F- 1524.5

This is the Julian Day Number for the beginning of the date in question at 0 hours, Greenwich time. Note that this always gives you a half day extra. That is because the Julian Day begins at *noon*, Greenwich time. This is convenient for astronomers (who until recently only observed at night), but it is confusing.

Example: If the date is 1582 October 15,

Y = 1582
M = 10
D = 15
A = 15
B = 3
C = -10
E = 2300344
F = 336
JD = 2299160.5

To convert a Julian Day Number to a Gregorian date, assume that it is for 0 hours, Greenwich time (so that it ends in 0.5). Do the following calculations, again dropping the fractional part of all multiplications and divisions. *Note: This method will not give dates accurately on the Gregorian Proleptic Calendar, i.e., the calendar you get by extending the Gregorian calendar backwards to years earlier than 1582. using the Gregorian leap year rules. In particular, the method fails if Y<400.* Thanks to a correspondent, Bo Du, for some ideas that have improved this calculation.

Q = JD+0.5
Z = Integer part of Q
W = (Z - 1867216.25)/36524.25
X = W/4
A = Z+1+W-X
B = A+1524
C = (B-122.1)/365.25
D = 365.25xC
E = (B-D)/30.6001
F = 30.6001xE
Day of month = B-D-F+(Q-Z)
Month = E-1 or E-13 (must get number less than or equal to 12)
Year = C-4715 (if Month is January or February) or C-4716 (otherwise)

Example: Check the first calculation by starting with JD = 2299160.5

Q = 2299161
Z = 2299161
W = 11
X = 2
A = 2299171
B = 2300695

C = 6298
D = 2300344
E = 11
F = 336
Day of Month = 15
Month = 10
Year = 1582

Julian Time Computation

To measure time intervals, Julian Century concept is created. A Julian Century (JC) has 36525 mean solar days. The time interval is calculated with respect reference time of 3rd January, 1900 which corresponds to 2,415,020 Julian Days.

Denoting reference time as JDref, Julian century as JC and time in question as JD, then interval in JC from the reference time to the time in question is calculated as: T = (JD – JDref)/ JC

The Time from the referenced time to 18th Dec, 13 hours of UT is 1.00963838

Sidereal time

It is a time-keeping system astronomers use to keep track of the direction to point their telescopes to view a given star in the night sky. From a given observation point, a star found at one location in the sky will be found at basically the same location at another night when observed at the same sidereal time. This is similar to how the time kept by a sundial can be used to find the location of the Sun. Just as the Sun and Moon appear to rise in the east and set in the west, so do the stars. Both solar time and sidereal time make use of the regularity of the Earth's rotation about its polar axis. The basic difference between the two is that solar time maintains orientation to the Sun while sidereal time maintains orientation to the stars in the night sky. The exact definition of sidereal time fixes it to the vernal equinox. Precession and nutation, though quite small on a daily basis, prevent sidereal time from being a direct measure of the rotation of the Earth relative to inertial space. Common time on a typical clock measures a slightly longer cycle, accounting not only for the Earth's axial rotation but also for the Earth's annual revolution around the Sun of slightly less than 1 degree per day.

A sidereal day is approximately 23 hours, 56 minutes, 4.091 seconds (23.93447 hours or 0.99726957 mean solar days), corresponding to the time it takes for the Earth to complete one rotation relative to the vernal equinox. The vernal equinox itself precesses very slowly in a west ward direction relative to the fixed stars, completing one revolution every 26,000 years approximately. As a consequence, the misnamed sidereal day, as "sidereal" is derived from the Latin sidus meaning "star", is some 0.008 seconds shorter than the Earth's period of rotation relative to the fixed stars.

Launch of GEO Satellites

It is better to launch rockets closer to the equator because the Earth rotates at a greater speed here than that at either pole. This extra speed at the equator means a rocket needs less thrust (and therefore less fuel) to launch into orbit.

In addition, launching at the equator provides an additional 1,036 mph (1,667 km/h) of speed once the vehicle reaches orbit. This speed bonus means the vehicle needs less fuel, and that freed space can be used to carry more pay load.

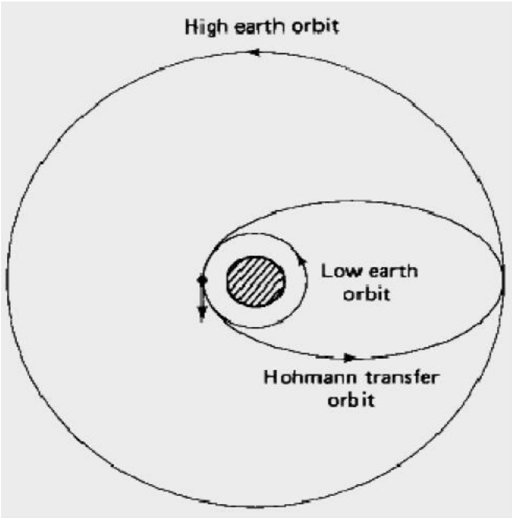

Figure 1.5: Hohmann Transfer Orbit

Hohmann Transfer Orbit: This manoeuvre is named for the German civil engineer who first proposed it, Walter Hohmann, who was born in 1880. He didn't work in rocketry professionally (and wasn't associated with military rocketry), but was a key member of Germany's pioneering Society for Space Travel that included people such as Willy Ley, Hermann, and Werner von Braun. He published his concept of how to transfer between orbits in his 1925 book, The Attainability of Celestial Bodies.)
The transfer orbit is selected to minimize the energy required for the transfer. This orbit forms a tangent to the low attitude orbit at the point of its perigee and tangent to high altitude orbit at the point of its apogee.

Orbital Perturbations

- Theoretically, an orbit described by Kepler is ideal as Earth is considered to be a perfect sphere and the force acting around the Earth is the centrifugal force. This force is supposed to balance the gravitational pull of the earth.

- In reality, other forces also play an important role and affect the motion of the satellite. These forces are the gravitational forces of Sun and Moon along with the atmospheric drag.

- Effect of Sun and Moon is more pronounced on geostationary earth satellites where as the atmospheric drag effect is more pronounced for low earth orbit satellites.

- As the shape of Earth is not a perfect sphere, it causes some variations in the path followed by the satellites around the primary. As the Earth is bulging from the equatorial belt, and keeping in mind that an orbit is not a physical entity, and it is the forces resulting from an oblate Earth which act on the satellite produce a change in the orbital parameters.

- This causes the satellite to drift as a result of regression of the nodes and the latitude of the point of perigee (point closest to the Earth). This leads to rotation of the line of apsides. As the orbit itself is moving with respect to the Earth, the resultant changes are seen in the values of argument of perigee and right ascension of ascending node.

- Due to the non-spherical shape of Earth, one more effect called as the "Satellite Graveyard' is seen. The non-spherical shape leads to the small value of eccentricity (10^{-5}) at the equatorial plane. This causes a gravity gradient on GEO satellite and makes them drift to one of the two stable points which coincide with minor axis of the equatorial ellipse.

- Working satellites are made to drift back to their position but out-of-service satellites are eventually drifted to these points, and making that point a Satellite Graveyard.

  (Note: A graveyard orbit, also called a super synchronous orbit, junk orbit or disposal orbit, is an orbit significantly above GEO where satellites are intentionally placed at the end of their operational life. It is a measure performed in order to lower the probability of collisions with operational spacecraft and of the generation of additional space debris. The points where the graveyard is made are separated by 1800 on the equator and are set approximately on 750 E longitude and 1050 W longitude.)

Look Angles of Antenna

The look angles for the ground station antenna are Azimuth and Elevation angles. They are required at the antenna so that it points directly at the satellite. Look angles are calculated by considering the elliptical orbit. These angles change in order to track the satellite.

- For geostationary orbit, these angle values do not change as the satellites are stationary with respect to earth. Thus large earth stations are used for commercial communications, these antennas beam width is very narrow and the tracking mechanism is required to compensate for the movement of the satellite about the nominal geostationary position.
- For home antennas, antenna beam width is quite broad and hence no tracking is essential. This leads to a fixed position for these antennas.

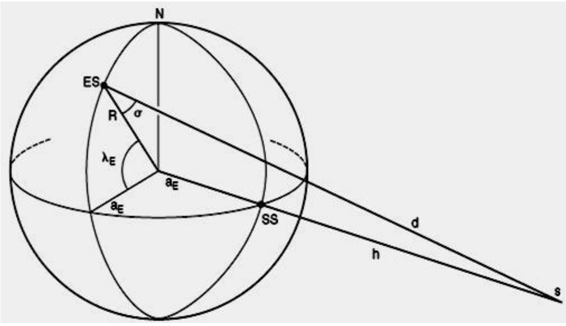

Figure1.5 : The geometry used in determining the look angles for Geostationary Satellites.

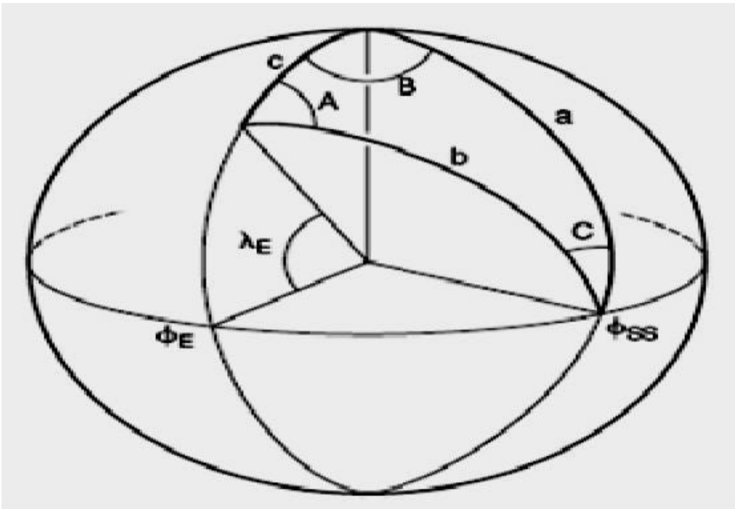

Figure 1.6: The spherical geometry related to previous figure

With respect to the figure 1.5 and 1.6, the following information is needed to determine the look angles of geostationary orbit.

1. Earth Station Latitude: λE
2. Earth Station Longitude: ΦE
3. Sub-Satellite Point's Longitude: ΦSS
4. ES: Position of Earth Station

5. SS: Sub-Satellite Point
6. S: Satellite
7. d: Range from ES to S
8. ζ: angle to be determined

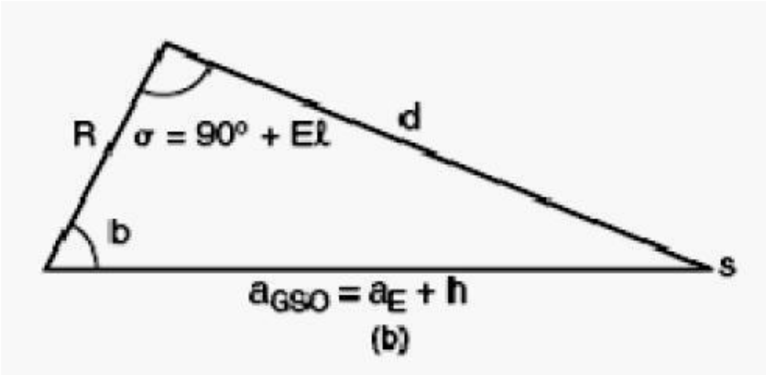

Figure1.7: A plane triangle obtained from previous figure

Considering figure 1.7, it's a spherical triangle. All sides are the arcs of a great circle. Three sides of this triangle are defined by the angles subtended by the centre of the earth.

- Side a: angle between North Pole and radius of the sub-satellite point.
- Side b: angle between radius of Earth and radius of the sub-satellite point.
- Side c: angle between radius of Earth and the North Pole.

a =900 and such a spherical triangle is called quadrantal triangle. c = 900 – λ
Angle 'B' is the angle between the plane containing c and the plane containing a.
Thus, B = ΦE-ΦSS
Angle 'A' is the angle between the plane containing b and the plane containing c.
Angle C is the angle between the plane containing a and the plane containing b.
Thus, a = 900 c = 900 - λE
Thus,   b = arccos (cos B cos λE)
And   A = arcsin (sin |B| / sin b)
Applying the cosine rule for plane triangle to the triangle of figure

$$d = \sqrt{R^2 + a_{GSO}^2 - 2Ra_{GSO}\cos b}$$

Applying the sine rule for plane triangles to the triangle of figure 3.3 allows the angle of elevation to be found:

$$El = \arccos\left(\frac{a_{GSO}}{d}\sin b\right)$$

Determination of Azimuth Angle

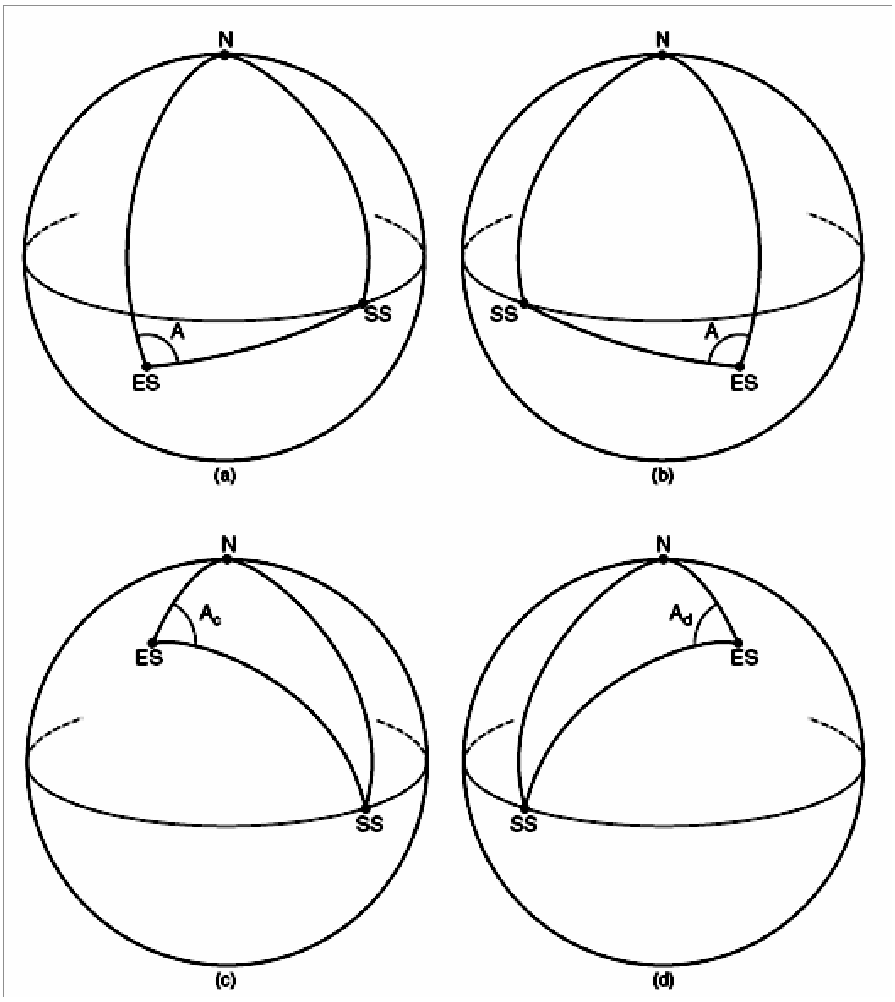

Fig 1.8: Relative Positions of Earth Station and Sub-satellite Points

Figure	Location of Earth Station(ES) Hemisphere	ES is West / East w.r.t. Sub Satellite Point	Azimuth Angle
(a)	**Southern**	**West**	**A**
(b)	**Southern**	**East**	**360-A**
(c)	**Northern**	**West**	**180-A**
(d)	**Northern**	**East**	**180+A**

Where $A = \arcsin \left[\frac{Sin|B|}{Sin|b|}\right]$

Example: A geostationary satellite is located 90^0 W. Calculate the azimuth angle for an Earth station antenna located at latitude 35^0 N and longitude 100^0 W. Also find Range and antenna elevation angle.

Solution: The given quantities are:

Φ_E = - 100 degrees; Φ_{SS}= - 90 degrees; $\boldsymbol{\lambda_E}$ = 35 degrees

B = ΦE- ΦSS = -10 degrees
B = acos{cos(B). cos(λE)} **= 36.23** degrees
A= arcsin $\left[\frac{Sin|B|}{Sin|b|}\right]$ **= 17.1** degrees

$$d = \sqrt{R^2 + a_{GSO}{}^2 - 2Ra_{GSO}\cos b}$$

$$= \sqrt{6371^2 + 42164^2 - 2 \; X \; 6371 \; X \; 42164 \; X \; cos 36.23}$$

= 37215 km

$$El = \arccos\left(\frac{a_{GSO}}{d}\sin b\right)$$

El = arcos $\left\{\frac{42164}{37215}\sin 36.23^o\right\} = 48^o$

Azimuth Angle: The ES is in Northern Hemisphere and to the west of SS.
Azimuth angle from table= 180-A = 180- 17.1 = 162^o

Limits of visibility:
The east and west limits of geostationary are visible from any given Earth station. These limits are set by the geographic coordinates of the Earth station and antenna elevation. The lowest elevation is zero (in theory) but in practice, to avoid reception of excess noise from Earth. Some finite minimum value of elevation is issued. The earth station can see a satellite over a geostationary arc bounded by **+- (81.30)** about the earth station's longitude.

Earth Eclipse of a Satellite
It occurs when Earth's equatorial plane coincides with the plane of the Earth's orbit around the sun. Near the time of spring and autumnal equinoxes, when the sun is crossing the equator, the satellite passes into sun's shadow. This happens for some duration of time every day.

These eclipses begin 23 days before the equinox and end 23 days after the equinox. They last for almost 10 minutes at the beginning and end of equinox and increase for a maximum period of 72 minutes at a full eclipse. The solar cells of the satellite become non-functional during the eclipse period and the satellite is made to operate with the help of power supplied from the batteries.

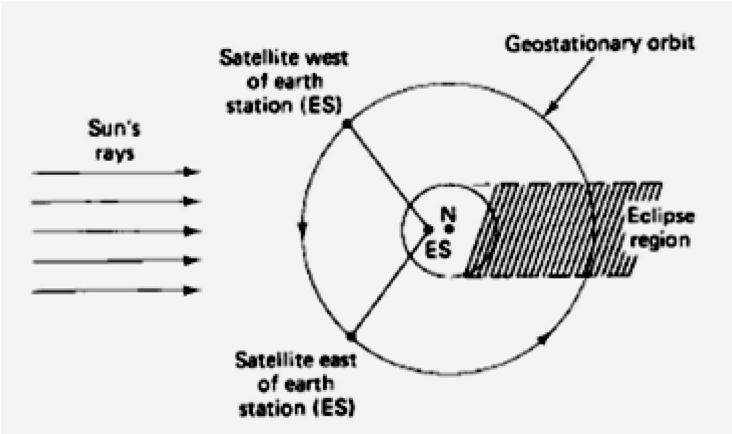

Fig 1.9: Earth eclipse of a satellite

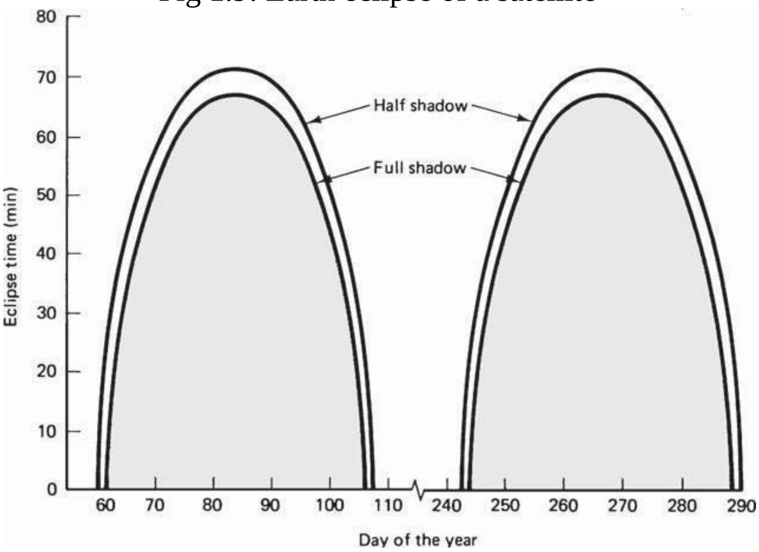

Fig 1.10: Eclipse duration around the year

A satellite will have the eclipse duration symmetric around the time
t = Satellite Longitude/15 + 12 hours.
A satellite at Greenwich longitude 0 will have the eclipse duration symmetric around 0/15 UTC +12hours = 00:00 UTC. The eclipse will happen at night but for satellites in the east it will happen late evening local time. For satellites in the west eclipse will happen in the early morning hour's local time. An earth caused eclipse will normally not happen during peak viewing hours if the satellite is located near the longitude of the coverage area. Modern satellites are well equipped with batteries for operation during eclipse.

Sun Transit Outage:
Sun transit outage is an interruption in or distortion of geostationary satellite signals caused by interference from solar radiation.
Sun appears to be an extremely noisy source which completely blanks out the signal from satellite. This effect lasts for 6 days around the equinoxes. They occur for a maximum period of 10 minutes.
Generally, sun outages occur in February, March, September and October, that is, around the time of the equinoxes. At these times, the apparent path of the sun across the sky takes it directly behind the line of sight between an earth station and a satellite. As the sun radiates strongly at the microwave frequencies used to communicate with satellites (C-band, Ka band and Ku band) the sun swamps the signal from the satellite. The effects of a sun outage can

include partial degradation, that is, an increase in the error rate, or total destruction of the signal.

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