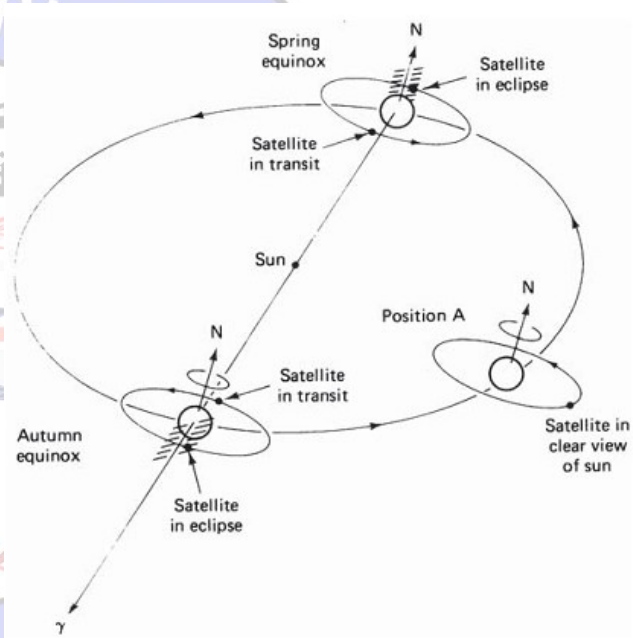
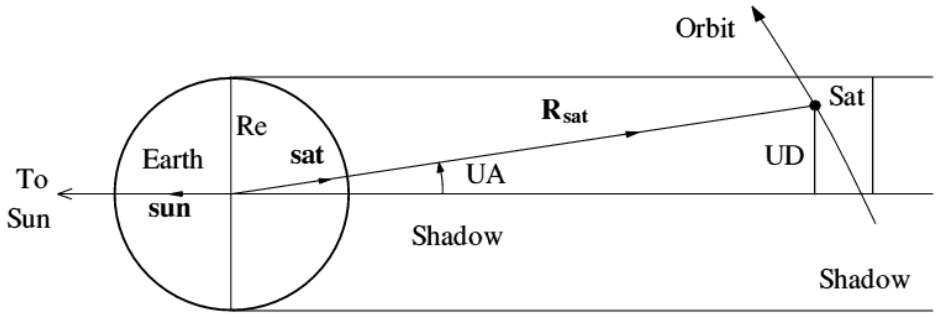
WEEK 5

Solar Eclipse

If the earth's equatorial plane coincided with the plane of the earth's orbit around the sun (the ecliptic plane), geostationary satellites would be eclipsed by the earth once each day. As it is, the equatorial plane is tilted at an angle of 23.4°to the ecliptic plane, and this keeps the satellite in full view of the sun for most days of the year, as illustrated by position A in Fig. Around the spring and autumnal equinoxes, when the sun is crossing the equator, the satellite does pass into the earth's shadow at certain periods, these being periods of eclipse as illustrated in Fig. The spring equinox is the first day of spring, and the autumnal equinox is the first day of autumn.

Eclipses begin 23 days before equinox and end 23 days after equinox. The eclipse lasts about 10 min at the beginning and end of the eclipse period and increases to a maximum duration of about 72 min at full eclipse. During an eclipse, the solar cells do not function, and operating power must be supplied from batteries. Figure (a) shows eclipse time as a function of height.





ECLIPSE GEOMETRY

Ex. A LEO satellite with 600 km height, calculate eclipse period & frequency. Take the Earth radius as 6400 Km.

According to the geometry,

$$\theta = \sin^{-1}\frac{6400}{7000} = 66$$

$2\theta/360$=0.36 which means 36% of its period the satellite will be in eclipse

We previously knew - from Kepler's rule - that, $a^3=\mu/n^2$

n=0.001078 rad/s

$p=2\pi/n$

p=5828.519 sec. = 97.142 min = 1.619 hrs.

This means that the satellite is orbiting the earth more than 14 times a day, so 14 eclipse per day take places.

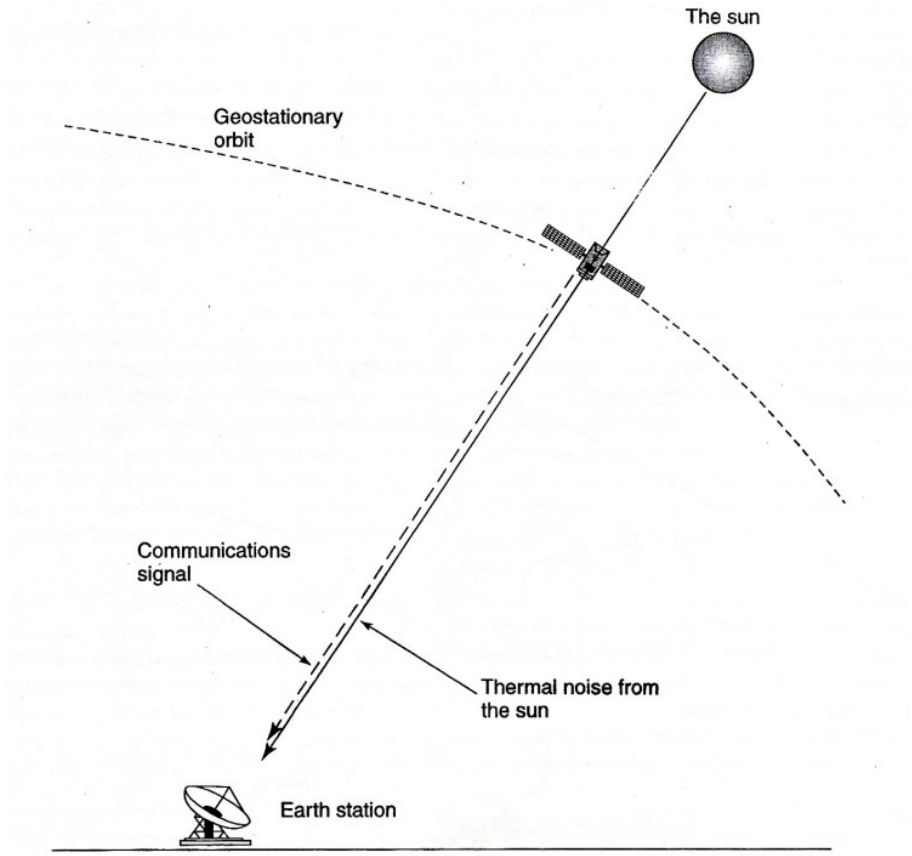
Now 36 % eclipse for a period of 97.142 min=~100min will be a 36min eclipse period.

Exercise1: Check the eclipse period & frequency for LEO satellite with 200 km height.

Exercise2: Check the eclipse period & frequency for GEO satellite in minutes.

Sun Transit Outage

During the equinox periods, not only does the satellite pass through the earth's shadow on the "dark" side of the earth, but the orbit of the satellite will also pass directly in front of the sun on the sunlit side of the earth (Figure 2.23). The sun is a "hot" microwave source with an equivalent temperature of about 6000 to 10,000 K, depending on the time within the 11-year sunspot cycle, at the frequencies used by communications satellites (4 to 50 GHz). The earth station antenna will therefore receive not only the signal from the satellite but also the noise temperature transmitted by the sun. The added noise temperature will cause the fade margin of the receiver to be exceeded and an outage will occur. These outages may be precisely predicted. For satellite system operators with more



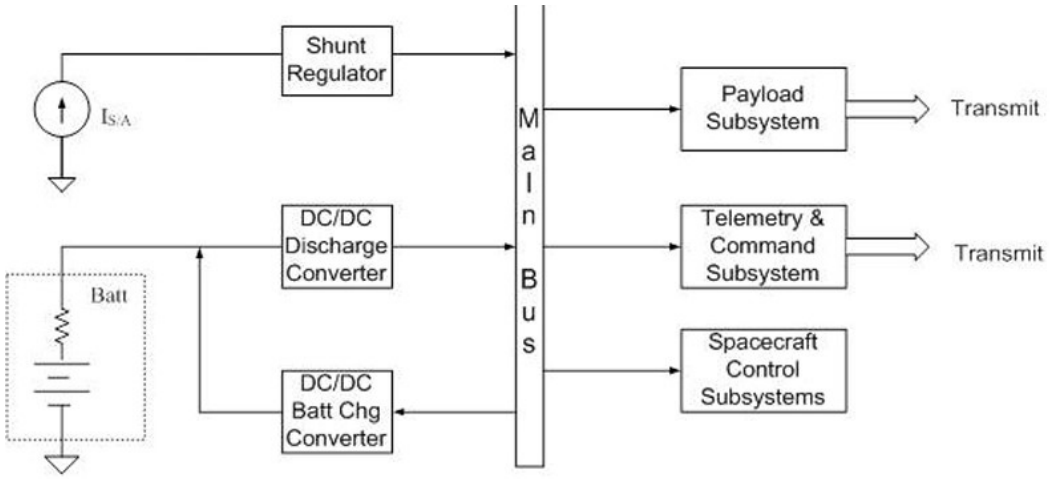
SATELLITE POWER SYSTEMS

The primary electrical power for operating the electronic equipment is obtained from solar cells. Individual cells can generate only small amounts of power, and therefore, arrays of cells in series-parallel connection are required.

Higher powers can be achieved with solar panels arranged in the form of rectangular solar sails. Solar sails must be folded during the launch phase and extended when in geostationary orbit. The solar cells are folded up on each side, and when fully extended, they stretch to its full area. The full complement of solar cells is exposed to the sunlight, and the sails are arranged to rotate to track the sun, so they are capable of greater power output than cylindrical arrays having a comparable number of cells. The HS 601 model can be designed to provide dc power from 2 to 6 kW.

The satellite power system is responsible for supplying electric energy to various satellite subsystems. Needless to mention that this subsystem has to meet stringent mass and volume limitation.

A typical power system is shown below:



Typical satellite power system

DC/DC Battery Charge Convertor -Battery Charge Regulator- (BCR)

BCR is a module responsible for charging the batteries during sun light. BCR monitors the charge state of the batteries and determines charging current accordingly; it should work at high efficiency.

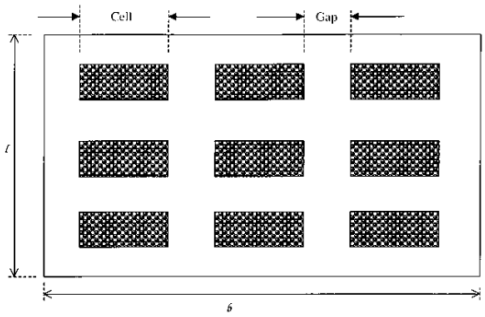
DC/DC Battery Discharge Convertor -Battery Discharge Regulator- (BDR)

During eclipse, when the solar panel stop generating electric power, and the spacecraft electric bus is powered by the rechargeable batteries, the BDR controls the output voltage from the batteries and provides the necessary bus voltage.

A key constraint on satellites is power. Solar cells can power the satellite transponder. Due to the inherent risk of nuclear fuel, solar energy power becomes attractive.

Above Earth's atmosphere, the average solar flux that falls on a spacecraft solar panel is about 1353 W/m^2 (the solar constant). If we are able to harness the sun's energy, we will be able to develop a power source for the satellite; the power source may be defined as:

$$P_s = k_s A_a \eta_{sc} \eta_e cos\theta$$



P_s : effective solar system power (w).
K_s : solar constant .
A_a: panel area.
η_e: Ratio of effective solar cell area to the panel area (80~98) %
η_{sc}: solar cells' conversion efficiency; depends on the material used, for example:

- η_{sc}: ~26% for gallium arsenide (GaAs) semiconductor.
- η_{sc}: (10~14) % for silicon semiconductor.

θ : inclination angle of the panel with respect to the sun light.

- ☒ Solar cell efficiency is reduced by 30% when operated in outer space for long period (~7 years) due to the exposure to the nuclear radiation.

During eclipses, solar cells are inactive. To keep the communication system operating, backup batteries make up the power system.

Ex. A system requires 150W of continuous power for a communication satellite. Design a power source that meets this requirement, you have only gallium arsenide with an unused area of 15%.

There are several types of solar panel configuration exist, the most commonly used can be order into the following categories:

i- A spherical (or nearly spherical) satellite can be designed where solar panel covers the satellite skin. In this case, no orientation (attitude) on the satellite is required. The area exposed to the sun is given by:

$$A_{Sun} = \frac{1}{4} A_{total}$$

where A_{Sun} is the effective area illuminated by Sun.
A_{total} is the total spherical area.

ii- Cylindrical satellite with spin stabilization. Solar panels cover the side skin of the satellite. In this case the effective area exposed to the Sun is given by:

$$A_{sun} = \frac{1}{\pi} A_{total} \cos\delta$$

Where δ is the angle made by the sun with respect to the plane perpendicular to the spin axis.

iii- Body stabilized satellite: In this class deployable solar panel can be used, solar panels normally are design to track the Sun in its motion. However, variation of declination of the sun is normally not compensated for the sake of simplicity. In this case:

$$A_{sun} = A_{total} \cos\theta$$

where θ is the angle between the sun elevation and the plane perpendicular to the rotation axis of the sun tracking motor.

Secondary Energy Source:

The secondary energy source store energy from the primary source & provide the stored energy when the primary stop functioning (e.g. during eclipse). Types of batteries used are Nickle-Cademium and Nickle-Hydrogen.

Their capacity is determined by:

i- Spacecraft load during the eclipse.
ii- Eclipse duration.
iii- Frequency of eclipse.

To compute the battery capacity, the following equation is used:

$$Battery\ Capacity(in\ A\,H) = \frac{W_{eclipse}}{V_B} \cdot \frac{T_{eclipse}}{DoD}$$

where:

$W_{eclipse}$: Maximum load during eclipse.

V_B: Battery Voltage

$T_{eclipse}$: Maximum duration of eclipse time.

DoD: Allowable depth of discharge.

For geo satellites, they encounter 84 eclipses per year & DoD is around 0.4 to 0.5.

For LEO the case is different, the satellite encounters around 15 eclipse per day with about 35 minute duration. This means the battery has more than 5000 charge/discharge per year, for this reason; batteries for LEO are allowed for (0.15~0.2) DoD.

Transponder

A transponder is the series of interconnected units which forms a single communications channel between the receive and transmit antennas in a communications satellite. Some of the units utilized by a transponder in a given channel may be common to a number of transponders. Thus, although reference may be made to a specific transponder, this must be thought of as an equipment channel rather than a single item of equipment.

The bandwidth allocated for C-band service is 500 MHz, and this is divided into sub-bands, one for each transponder. A typical transponder bandwidth is 36 MHz, and allowing for a 4-MHz guard band between transponders, 12 such transponders can be accommodated in the 500-MHz bandwidth. By making use of polarization isolation, this number can be doubled. Polarization isolation refers to the fact that carriers, which may be on the same frequency but with opposite senses of polarization, can be isolated from one another by receiving antennas matched to the incoming polarization. With linear polarization, vertically and horizontally polarized carriers can be separated in this way, and with circular polarization, left-hand circular and right-hand circular polarizations can be separated.

Because the carriers with opposite senses of polarization may overlap in frequency, this technique is referred to as *frequency Re-use*. Figure shows part of the frequency and polarization plan for a C-band communications satellite.

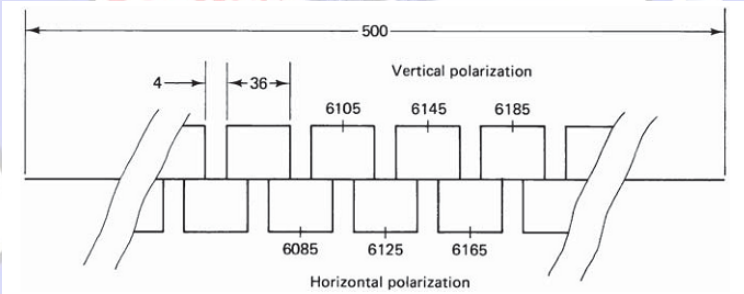
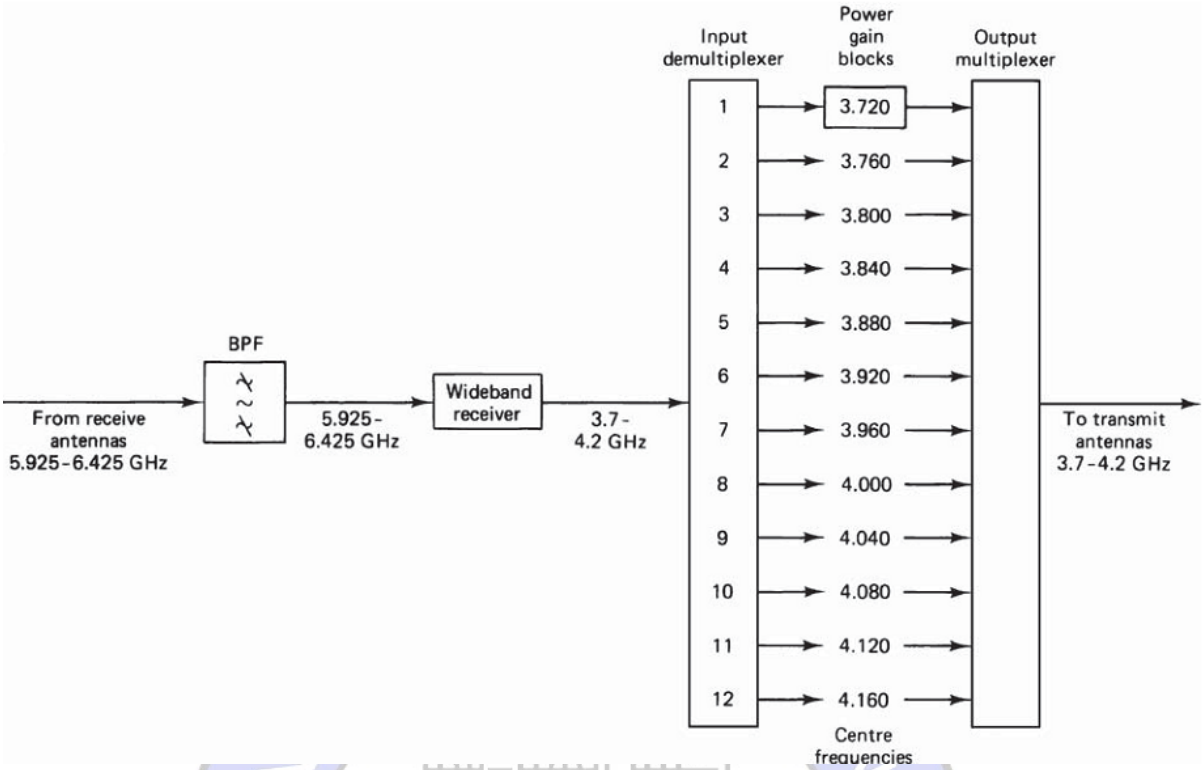
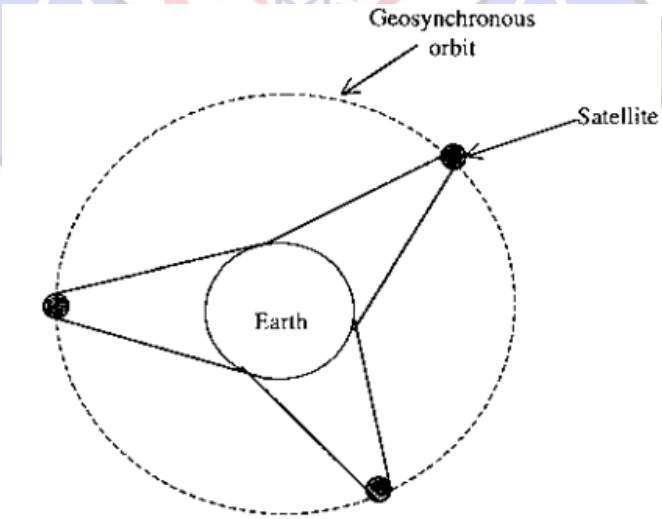


Fig.(4) Section of an uplink frequency and polarization plan. Numbers refer to frequency in megahertz.



COVERAGE AREA AND SATELLITE NETWORKS

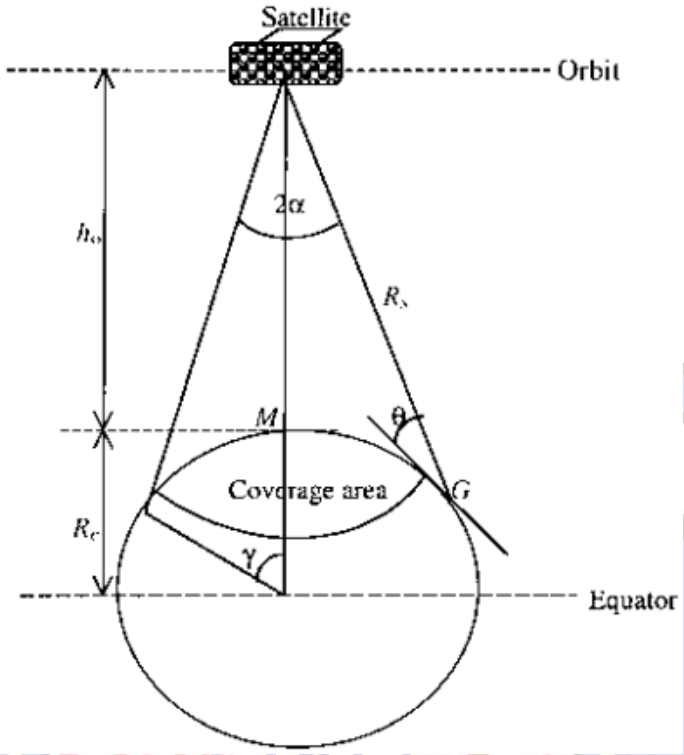
Clarke foresaw in his article that it would be possible to provide complete radio coverage of the world from just three satellites, provided that they could be precisely placed in geosynchronous orbit. Figure below demonstrates this.



Complete coverage of the earth's surface from three satellites.

The amount of coverage is an important feature in the design of earth observation satellites. Coverage depends on altitude and look angles of the equipment, among several factors.

To establish the geometric relationship of the coverage, we take a section of the satellites in the above figure as an illustration.



The maximum geometric coverage can be defined as the portion of the earth within a cone of the satellite at its apex, which is tangential to the earth's surface. Consider the angle of view from the satellite to the earth terminal as α; then the apex angle is 2α. The view angle has a mathematical physical function given by:

$$\alpha = \sin^{-1}\left(\frac{R_e}{h_0 + R_e}\right) = \sin^{-1}\left(\frac{R_e}{r}\right)$$

Exercise: Compute the minimum angle that's required by a satellite with 1000 km height to cover the maximum geometric coverage of the earth.