CHAPTER 3

SATELLITES

Maintaining a microwave communication system in orbit in space is not a simple problem, so communications satellites are very complex, extremely expensive to purchase, and also expensive to launch. A typical large geostationary satellite, for example, is estimated to cost around $125 M, on station (see Chapter 2). The cost of the satellite and launch are increased by the need to dedicate an earth station to the monitoring and control of the satellite, at a cost of several million dollars per year. The revenue to pay these costs is obtained by selling the communication capacity of the satellite to users, either by way of leasing circuits or transponders, or by charging for circuit use, as in the international telephone and data transmission service.

Communications satellites are usually designed to have a typical operating lifetime of 10 to 15 years. The operator of the system hopes to recover the initial and operating costs well within the expected lifetime of the satellite, and the designer must provide a satellite that can survive the hostile environment of outer space for that long. In order to support the communications system, the satellite must provide a stable platform on which to mount the antennas, be capable of station keeping, provide the required electrical power for the communication system, and also provide a controlled temperature environment for the communications electronics. In this chapter we discuss the subsystems needed on a satellite to support its primary mission of communications. We also discuss the communications subsystem itself in some detail, and other problems such as reliability.

The emphasis throughout this chapter is on satellites in geostationary orbit. Communications satellites for low earth orbit are in most cases quite similar to small GEO satellites and have similar requirements. The discussion of satellites in this chapter is necessarily brief. For more details of the many subsystems used on satellites and their construction and operation the reader should refer to reference 1. Much information about individual satellites can be found on the web sites of satellite manufacturers and operators. See Table 1.1–1.4 in Chapter 1 for an extensive listing, which includes Web addresses.

3.1 SATELLITE SUBSYSTEMS

The major subsystems required on the satellite are given below. Figure 3.1 shows an exploded view of a typical geostationary (GEO) satellite with several of the subsystems indicated.

orientation & position of sat

Attitude and Orbit Control System (AOCS)

This subsystem consists of rocket motors that are used to move the satellite back to the correct orbit when external forces cause it to drift off station and gas jets or inertial devices that control the attitude of the satellite.

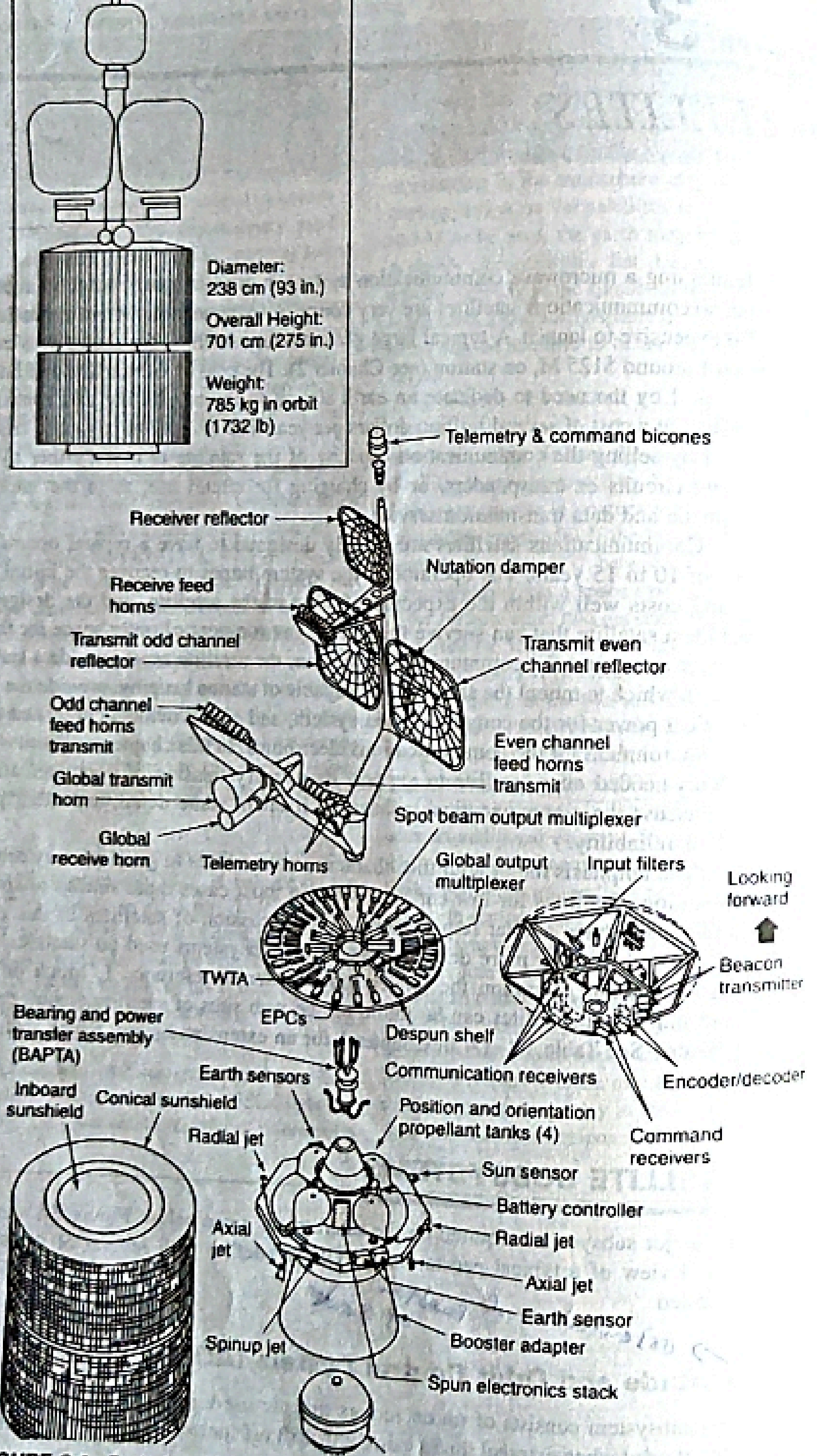

FIGURE 3.1 Exploded view of a spinner satellite based on the Boeing (Hughes) HS 376 design. INTELSAT IVA (courtesy of Intelsat)

Telemetry, Tracking, Command, and Monitoring (TTC&M)

These systems are partly on the satellite and partly at the controlling earth station[1]. The telemetry system sends data derived from many sensors on the satellite, which monitor the *satellite's health*, via a telemetry link to the controlling earth station. The tracking system is located at this earth station and provides information on the range and the elevation and azimuth angles of the satellite. Repeated measurement of these three parameters permits computation of orbital elements, from which changes in the orbit of the satellite can be detected. Based on telemetry data received from the satellite and orbital data obtained from the tracking system, the control system is used to correct the position and attitude of the satellite. It is also used to control the antenna pointing and communication system configuration to suit current traffic requirements, and to operate switches on the satellite.

Power System

All communications satellites derive their electrical power from *solar cells*. The power is used by the communications system, mainly in its transmitters, and also by all other electrical systems on the satellite. The latter use is termed *housekeeping*, since these subsystems serve to support the communications system.

Communications Subsystems

The communications subsystem is the major component of a communications satellite, and the remainder of the satellite is there solely to support it. Frequently, the communications equipment is only a small part of the weight and volume of the whole satellite. It is usually composed of one or more antennas, which receive and transmit over wide bandwidths at microwave frequencies, and a set of receivers and transmitters that amplify and retransmit the incoming signals. The receiver–transmitter units are known as *transponders*. There are two types of transponder in use on satellites: the linear or *bent pipe* transponder that amplifies the received signal and retransmits it at a different, usually lower, frequency, and the *baseband processing transponder*, which is used only with digital signals, that converts the received signal to baseband, processes it, and then retransmits a digital signal.

Satellite Antennas

Although these form part of the complete communication system, they can be considered separately from the transponders. On large GEO satellites the antenna systems are very complex and produce beams with shapes carefully tailored to match the areas on the earth's surface served by the satellite. Most satellite antennas are designed to operate in a single frequency band, for example, C band or Ku band. A satellite which uses multiple frequency bands usually has four or more antennas.

The subsystems listed above are discussed in more detail in this chapter. There are other subsystems that are not discussed here, but which are essential to the operation of the satellite—the thermal control system that regulates the temperature inside a satellite, for example. The reader who is interested in spacecraft design should refer to the literature of that field, particularly the *IEEE Transactions on Aerospace and Electronic Systems* and the *American Institute of Aeronautics and Astronautics Transactions* and annual *Conference Proceedings*[3,4]. Only a brief review of the subsystems that support the communication mission is included here.

3.2 ATTITUDE AND ORBIT CONTROL SYSTEM (AOCS)

The attitude and orbit of a satellite must be controlled so that the satellite's antennas point toward the earth and so that the user knows where in the sky to look for the satellite. This is particularly important for GEO satellites since the earth station antennas that are used with GEO satellites are normally fixed and movement of the satellite away from its appointed position in the sky will cause a loss of signal. There are several forces acting on an orbiting satellite that tend to change its attitude and orbit, as discussed in Chapter 2. The most important are the gravitational fields of the sun and the moon, irregularities in the earth's gravitational field, solar pressure from the sun, and variations in the earth's magnetic field.

Solar pressure acting on a satellite's solar sails and antennas, and the earth's magnetic field generating eddy currents in the satellite's metallic structure as it travels through the magnetic field, tend to cause rotation of the satellite body. Careful design of the structure can minimize these effects, but the orbital period of the satellite makes many of the effects cyclic, which can cause *nutation* (a wobble) of the satellite. The attitude control system must damp out nutation and counter any rotational torque or movement.

The presence of gravitational fields from the sun and the moon cause the orbit of a GEO satellite to change with time. At GEO orbit altitude, the moon's gravitational force is about twice as strong as the sun's. The moon's orbit is inclined to the equatorial plane by approximately 5°, which creates a force on the satellite with a component that is normal to the satellite's orbit. The plane of the earth's rotation around the sun is inclined by 23° to the earth's equatorial plane. As discussed in Chapter 2, there is a net gravitational pull on the satellite that tends to change the inclination of the satellite's orbit, pulling it away from the earth's equatorial plane at an initial rate of approximately 0.86° per year. The orbital control system of the satellite must be able to move the satellite back into the equatorial plane before the orbital inclination becomes excessive. LEO satellites are less affected by gravitational fields of the sun and moon. Since they are much closer to the earth than GEO satellites, the earth's gravity is much stronger, and the pull from the sun and moon are proportionately weaker.

The earth is not quite a perfect sphere. At the equator, there are bulges of about 65 m at longitudes 162° E and 348° E, with the result that a satellite is accelerated toward one of two stable points in the GEO orbit at longitude 75° E and 252° E, as shown in Figure 3.2. To maintain accurate station keeping, the satellite must be periodically accelerated in the opposite direction to the forces acting on it. This is done as a sequence of *station-keeping maneuvers*, using small rocket motors (sometimes called *gas jets* or *thrusters*) that can be controlled from the earth via the TTC&M system.

Attitude Control System

There are two ways to make a satellite stable in orbit, when it is weightless. The body of the satellite can be rotated, typically at a rate between 30 and 100 rpm, to create a gyroscopic force that provides stability of the spin axis and keeps it pointing in the same direction. Such satellites are known as *spinners*. The popular Hughes 376 (now Boeing 376) satellite is an example of a spinner design. Alternatively, the satellite can be stabilized by one or more *momentum wheels*. This is called a *three-axis stabilized satellite*, of which the Hughes (Boeing) 701 series is an example. The momentum wheel is usually a solid metal disk driven by an electric motor. Either there must be one momentum wheel for each of the three axes of the satellite, or a single momentum wheel can be mounted on

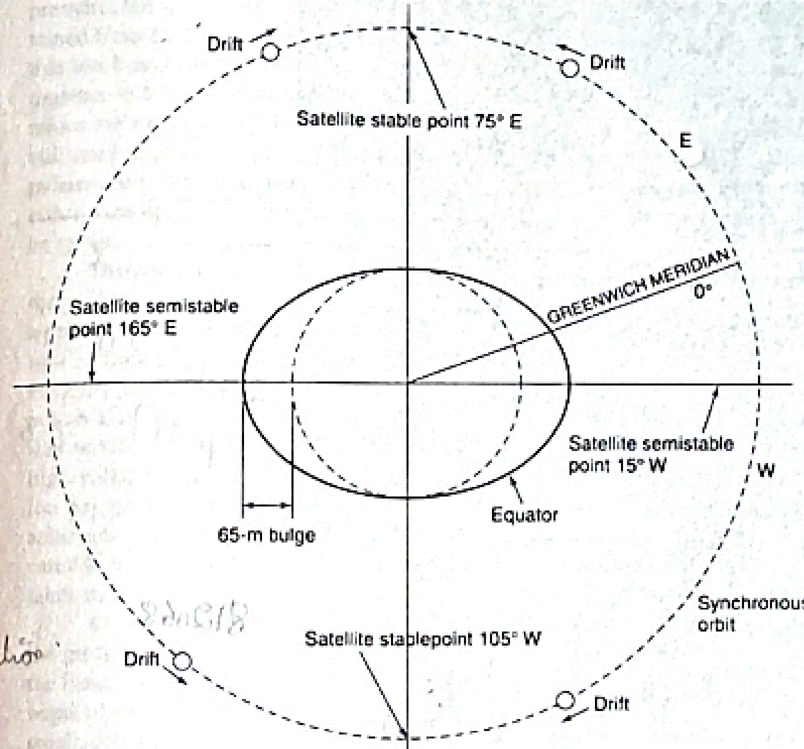

FIGURE 3.2 Forces on a synchronous satellite.

gimbals and rotated to provide a rotational force about any of the three axes[5,6]. Increasing the speed of the momentum wheel causes the satellite to precess in the opposite direction, according the principle of conservation of angular momentum. Figure 3.3 shows examples of both the spinner and three-axis design of satellite.

The spinner design of satellite is typified by many satellites built by the Hughes Aircraft Corporation for domestic satellite communication systems. As shown in Figures 3.1 and 3.3*a*, the satellite consists of a cylindrical drum covered in solar cells that contains the power systems and the rocket motors. The communications system is mounted at the top of the drum and is driven by an electric motor in the opposite direction to the rotation of the satellite body to keep the antennas pointing toward the earth. Such satellites are called *despun*.

SIDEBAR

In the early days of satellite communication *despun antennas* were not used, so antennas with a circular symmetric pattern were employed. These antennas have low gain and are now used only for basic TTC&M systems that must operate regardless of the satellite's orientation. By despinning the antennas and transponders, no RF rotating joints are needed in the communication system. Electrical power and some control signals must be brought through the despin bearing assembly with slip rings. The design of the bearing and slip rings to guarantee friction-free operation for 10 to 15 years in the total vacuum of outer space is a challenge.


(a)

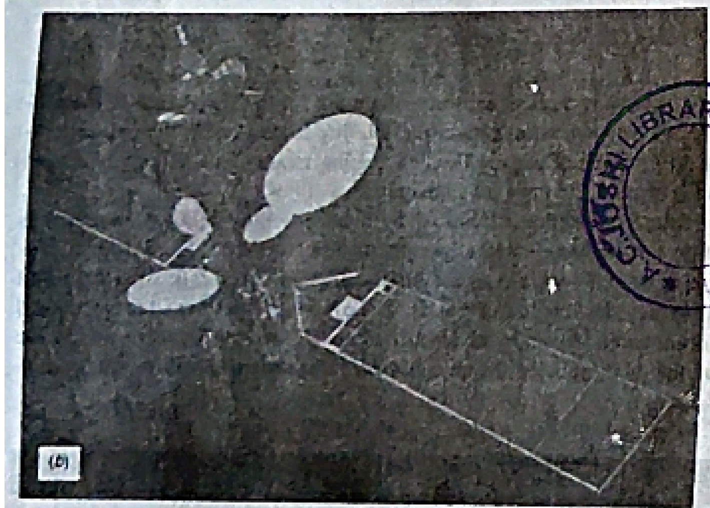
(b)

FIGURE 3.3 *(a)* A spinner satellite, INTELSAT IV A *(b)* A three-axis stabilized satellite, INTELSAT V (courtesy of Intelsat).

The satellite is *spun up* by operating small radial gas jets mounted on the periphery of the drum, at an appropriate point in the launch phase. The despin system is then brought into operation so that the main TTC&M antennas point toward the earth. The main TTC&M system operates at 6/4 GHz on the Intelsat satellite, with a 2-GHz backup system for use during the launch phase. A variety of liquid propulsion mixes have been used for the gas jets, the most common being a variant of *hydrazine* (N_2H_4), which is easily liquefied under

pressure, but readily decomposes when passed over a catalyst[7]. Increased power can be obtained from the hydrazine gas jets by electrically heating the catalyst and the gas. Satellites that use liquid fuel thrusters have standardized on bipropellant fuels, that is fuels that mix together to form the thruster fuel. The most common bipropellants used for thruster operations are mono-methyl hydrazine and nitrogen tetroxide, although standard hydrazine is still used in place of mono-methyl hydrazine by some satellite manufacturers. The bipropellants are hypogolic: that is they ignite spontaneously on contact, and so do not need either a catalyst or a heater. By adjusting the flow of the bipropellants, pulses of thrust can be generated at the correct time and in the correct direction.

There are two types of rocket motors used on satellites. The traditional bipropellant thruster described above, and *arc jets or ion thrusters*. The fuel that is stored on a GEO satellite is used for two purposes: to fire the apogee kick motor that injects the satellite into its final orbit, and to maintain the satellite in that orbit over its lifetime. If the launch is highly accurate, a minimum amount of fuel is used to attain the final orbit. If the launch is less accurate, more fuel must be used ~~up in maneuvering the satellite into position~~, and that reduces the amount left for station keeping. A new development in thrusters uses a high voltage source to accelerate ions to a very high velocity, thus producing thrust. The ion engine thrust is not large, but because the engine can be driven by power from the solar cells it saves on expendable fuel. Ion engines can also be used to slowly raise a GEO satellite from a transfer orbit to GEO orbit as described in Chapter 2, although the process takes months rather than hours as with a conventional rocket engine.

Arc jets or ion thrusters are mainly used for north–south station keeping, which is where the greatest use of fuel is required for station-keeping maneuvers, and became operational on the Hughes (Boeing) 600 series of satellite buses. Arc jets or ion thrusters lack the total thrust required to move satellites quickly (e.g., for major longitudinal changes in position) but a small, continuous thrust is adequate to maintain N–S and E–W position keeping.

In a three-axis stabilized satellite, one pair of gas jets is needed for each axis to provide for rotation in both directions of pitch, roll, and yaw. An additional set of controls, allowing only one jet on a given axis to be operated, provides for velocity increments in the *X*, *Y*, and *Z* directions. When motion is required along a given axis, the appropriate gas jet is operated for a specified period of time to achieve the desired velocity. The opposing gas jet must be operated for the same length of time to stop the motion when the satellite reaches its new position. Fuel is saved if the velocity of the satellite is kept small, but progress toward the destination is slow.

Let us define a set of reference Cartesian axes (X_R, Y_R, Z_R) with the satellite at the origin, as shown in Figure 3.4. The Z_R axis is directed toward the center of the earth and is in the plane of the satellite orbit. It is aligned along the local vertical at the satellite's subsatellite point. The X_R axis is tangent to the orbital plane and lies in the orbital plane. The Y_R axis is perpendicular to the orbital plane. For a satellite serving the Northern Hemisphere, the directions of the X_R and Y_R axes are nominally east and south.

Rotation about the X_R, Y_R, and Z_R axes is defined as *roll* about the X_R axis, *pitch* about the Y_R axis, and *yaw* about the Z_R axis, in exactly the same way as for an aircraft or ship traveling in the *X* direction. The satellite must be stabilized with respect to the reference axes to maintain accurate pointing of its antenna beams. The axes X_R, Y_R, and Z_R are defined with respect to the location of the satellite; a second set of Cartesian axes, *X*, *Y*, *Z*, as shown in Figure 3.4, define the orientation of the satellite. Changes in a satellite's attitude cause the angles θ, ϕ, and ψ in Figure 3.4 to vary as the *X*, *Y*, and *Z* axes move relative to the fixed reference axes X_R, Y_R, and Z_R. The *Z* axis is usually directed toward a reference point on the earth, called the *Z-axis intercept*. The location of the *Z*-axis intercept defines the pointing of the satellite antennas; the *Z*-axis intercept point may be

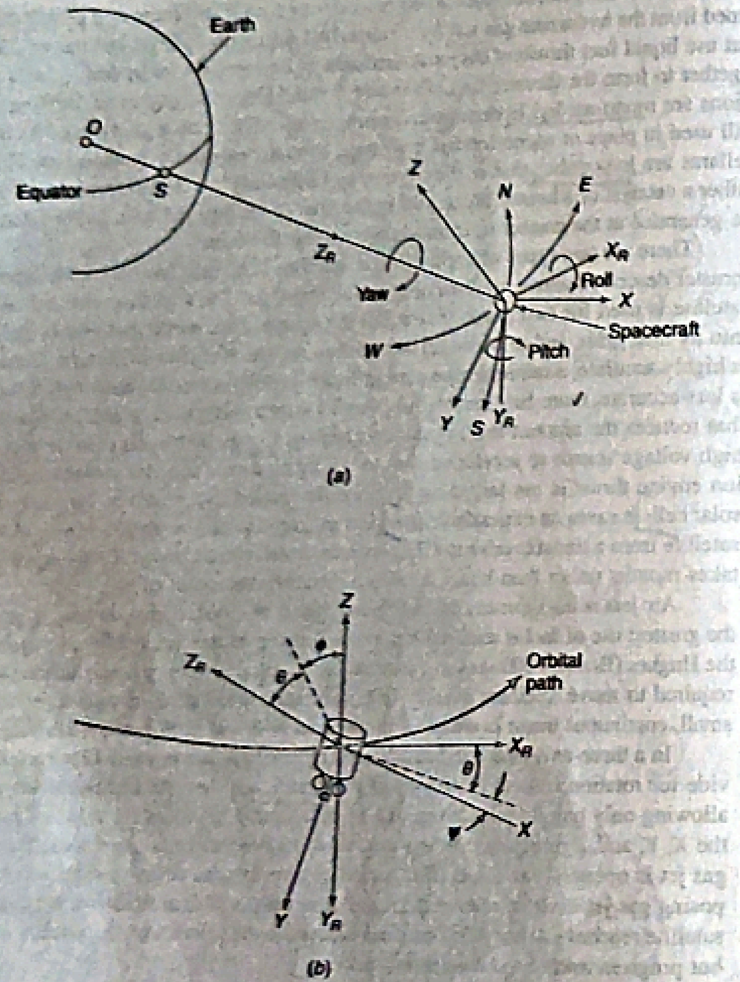

FIGURE 3.4 (a) Forces on a satellite. (b) Relationship between axes of a satellite.

moved to repoint all the antenna beams by changing the attitude of the satellite with the attitude control system.

In a spinner-type satellite, the axis of rotation is usually the Y axis, which is maintained close to the Y_R axis, perpendicular to the orbital plane. Pitch correction is required only on the despun antenna system and can be obtained by varying the speed of the despin motor. Yaw and roll are controlled by pulsing radially mounted jets at the appropriate instant as the body of the satellite rotates.

Attitude control of a three-axis stabilized satellite requires an increase or a decrease in the speed of the inertia wheel. If a constant torque exists about one axis of the satellite, a continual increase or decrease in momentum wheel speed is necessary to maintain the correct attitude. When the upper or lower speed limit of the wheel is reached, it must be *unloaded* by operating a pair of gas jets and simultaneously reducing or increasing the wheel speed. Closed-loop control of attitude is employed on the satellite to maintain

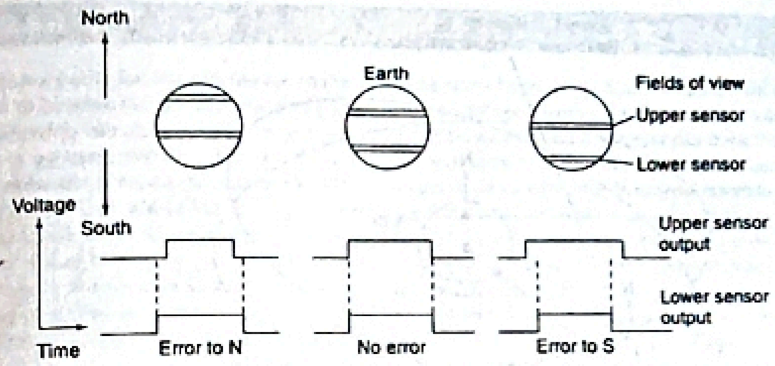

FIGURE 3.5 Principle of N–S control of a spinner satellite using infrared Earth sensors.

the correct attitude. When large, narrow beam antennas are used, the whole satellite may have to be stabilized within ±0.1° on each axis. The references for the attitude control system may be the outer edge of the earth's disk, as observed with infrared sensors, the sun, or one or more stars.

Figure 3.5 illustrates how an infrared sensor on the spinning body of a satellite can be used to control pointing toward the earth. Figure 3.6 shows a typical control system loop using the technique illustrated in Figure 3.5. The control system will be more complex for a three-axis stabilized satellite and may employ an onboard computer to process the sensor data and command the gas jets and momentum wheels.

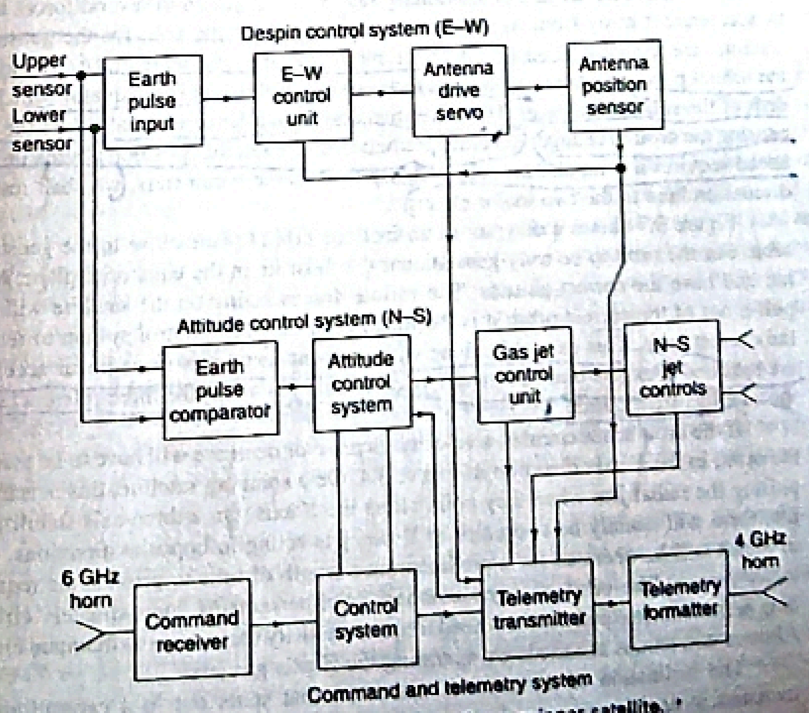

FIGURE 3.6 Typical onboard control system for a spinner satellite.

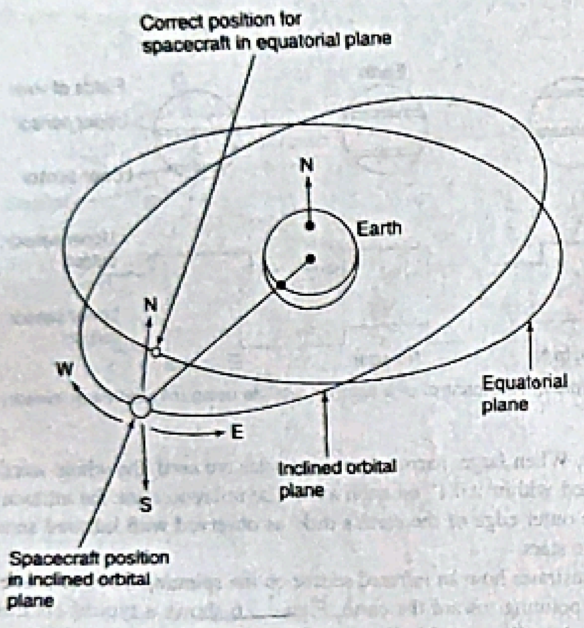

FIGURE 3.7 Satellite in inclined orbit.

Orbit Control System

As discussed in Chapter 2, a geostationary satellite is subjected to several forces that tend to accelerate it away from its required orbit. The most important, for the geostationary satellite, are the gravitation forces of the moon and the sun, which cause inclination of the orbital plane, and the nonspherical shape of the earth around the equator, which causes drift of the subsatellite point. There are many other smaller forces that act on the satellite causing the orbit to change. Accurate prediction of the satellite position a week or 2 weeks ahead requires a computer program with up to 20 force parameters; we shall restrict our discussion here to the two major effects.

Figure 3.7 shows a diagram of an inclined orbital plane close to the geostationary orbit. For the orbit to be truly geostationary, it must lie in the equatorial plane, be circular, and have the correct altitude. The various forces acting on the satellite will steadily pull it out of the correct orbit; it is the function of the orbit control system to return it to the correct orbit. This cannot be done with momentum wheels since linear accelerations are required. Gas jets that can impart velocity changes along the three references axes of the satellite are required.

If the orbit is not circular, a velocity increase or decrease will have to be made along the orbit, in the X-axis direction in Figure 3.4. On a spinning satellite, this is achieved by pulsing the radial jets when they point along the X axis. On a three-axis stabilized satellite, there will usually be two pairs of X-axis jets acting in opposite directions, one pair of which will be operated for a predetermined length of time to provide the required velocity change. The orbit of a geostationary satellite remains approximately circular for long periods of time and does not need frequent velocity corrections to maintain circularity. Altitude corrections are made by operating the Z-axis gas jets.

The inclination of the orbit of a satellite that starts out in a geostationary orbit increases at an average rate of about 0.85° per year, with an initial rate of change of

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The precessional forces that cause inclination changes can be used to increase satellite station-keeping lifetime by deliberately launching the satellite so that the inclination is not zero, but the precessional forces will act to reduce the inclination to close to zero. During this period, the E–W station-keeping tolerance is closely maintained. Once the inclination is close to zero, normal station keeping is started to maintain a tight orbit control in both axes. In this way, approximately a year of additional maneuvering lifetime may be obtained for each degree the inclination is nonzero at the start of operations. Some communications systems do not need very tight station-keeping tolerance, either because the earth segment can track the satellite accurately or because no tracking is required (as would be the case for omnidirectional antennas). GEO satellites in such systems may relax their inclination (N–S) tolerance but may never relax the E–W station keeping tolerance as this would lead to unacceptable interference into other systems. GEO satellites in such relaxed orbits are sometimes called inclined-orbit satellites.

inclination for a satellite in an equatorial orbit between 0.75° to 0.94° per year (see Chapter 2). Most GEO satellites are specified to remain within a box of ±0.05° and so, in practice, corrections, called a *north–south station-keeping maneuver* are made every 2 to 4 weeks to keep the error small. It has become normal to split the E–W and N–S maneuvers so that at intervals of 2 weeks the E–W corrections are made first and then after 2 more weeks, the N–S corrections are made. If arc jets or ion thrusters are used for N–S station-keeping maneuvers, these tend to operate almost continuously since their thrust levels are low when compared with traditional liquid fueled engines.

Correcting the inclination of a satellite orbit requires more fuel to be expended than for any other orbital correction. This places a weight penalty on those satellites that must maintain very accurate station keeping, and reduces the communications payload they can carry. As much as half the total satellite weight at launch may be station keeping fuel when the satellite's expected lifetime on orbit is 15 years.

East–west station keeping is effected by use of the X-axis jets of the satellite. For a satellite located away from the stable points at 75° E and 252° E, a slow drift toward these points will occur. Typically, the X-axis jets are pulsed every 2 or 3 weeks to counter the drift and add a small velocity increment in the opposite direction. The satellite then drifts through its nominal position, stops at a point a fraction of a degree beyond it, and then drifts back again. East–west station keeping requires only a modest amount of fuel and is necessary on all geostationary communications satellites to maintain the spacing between adjacent satellites. With orbital locations separated by 2° or 3°, east–west drifts in excess of a fraction of a degree cannot be tolerated, and most GEO satellites are held within ±0.05° of their allotted longitude.

Some communications satellites such as the Russian Molniya series are not in geostationary orbit. (*Molniya* means lightning in Russian.) Early Molniya satellites were launched into a highly elliptical 12-h orbit with a large (65°) inclination angle to provide communication to northerly latitudes like Siberia. The Russian satellite gave its name to any satellite in a highly elliptical inclined orbit.

Low earth orbit (LEO) and medium earth orbit (MEO) satellites also need AOC systems to maintain the correct orbit and attitude for continuous communication. Because of the much stronger gravitational force of the earth in LEO orbit, attitude stabilization is often accomplished with a rigid *gravity gradient boom*. This is a long pole that points toward the center of the earth, providing damping of oscillations about the satellite's z axis by virtue of the difference in gravitational field at the top of the pole and at the bottom.

3.3 TELEMETRY, TRACKING, COMMAND, AND MONITORING

The TTC&M system is essential to the successful operation of a communications satellite. It is part of the satellite management task, which also involves an earth station, usually dedicated to that task, and a group of personnel. The main functions of satellite management are to control the orbit and attitude of the satellite, monitor the status of all sensors and subsystems on the satellite, and switch on or off sections of the communication system. The TTC&M earth station may be owned and operated by the satellite owner, or it may be owned by a third party and provide TTC&M services under contract.

On large geostationary satellites, some repointing of individual antennas may be possible, under the command of the TTC&M system. Tracking is performed primarily by the earth station. Figure 3.8 illustrates the functions of a controlling earth station.

Telemetry and Monitoring System

The monitoring system collects data from many sensors within the satellite and sends these data to the controlling earth station. There may be several hundred sensors located on the satellite to monitor pressure in the fuel tanks, voltage and current in the power conditioning unit, current drawn by each subsystem, and critical voltages and currents in the communications electronics. The temperature of many of the subsystems is important and must be kept within predetermined limits, so many temperature sensors are fitted. The sensor data, the status of each subsystem, and the positions of switches in the communication system are reported back to the earth by the telemetry system. The sighting devices used to maintain attitude are also monitored via the telemetry link; this is essential in case one should fail and cause the satellite to point in the wrong direction. The faulty unit must then be disconnected and a spare brought in, via the command system, or some other means of controlling attitude devised.

Telemetry data are usually digitized and transmitted as phase shift keying (PSK) of a low-power telemetry carrier using time division techniques. A low data rate is normally used to allow the receiver at the earth station to have a narrow bandwidth and thus maintain a high carrier to noise ratio. The entire TDM frame may contain thousands of bits of data and take several seconds to transmit. At the controlling earth station a computer can be used to monitor, store, and decode the telemetry data so that the status of any system or sensor on the satellite can be determined immediately by the controller on the earth. Alarms can also be sounded if any vital parameter goes outside allowable limits.

Tracking

A number of techniques can be used to determine the current orbit of a satellite. Velocity and acceleration sensors on the satellite can be used to establish the change in orbit from the last known position, by integration of the data. The earth station controlling the satellite can observe the Doppler shift of the telemetry carrier or beacon transmitter carrier to determine the rate at which range is changing. Together with accurate angular measurements from the earth station antenna, range is used to determine the orbital elements. Active determination of range can be achieved by transmitting a pulse, or sequence of pulses, to the satellite and observing the time delay before the pulse is received again. The propagation delay in the satellite transponder must be accurately known, and more than one earth station may make range measurements. If a sufficient number of earth stations with an adequate separation are observing the satellite, its position can be established

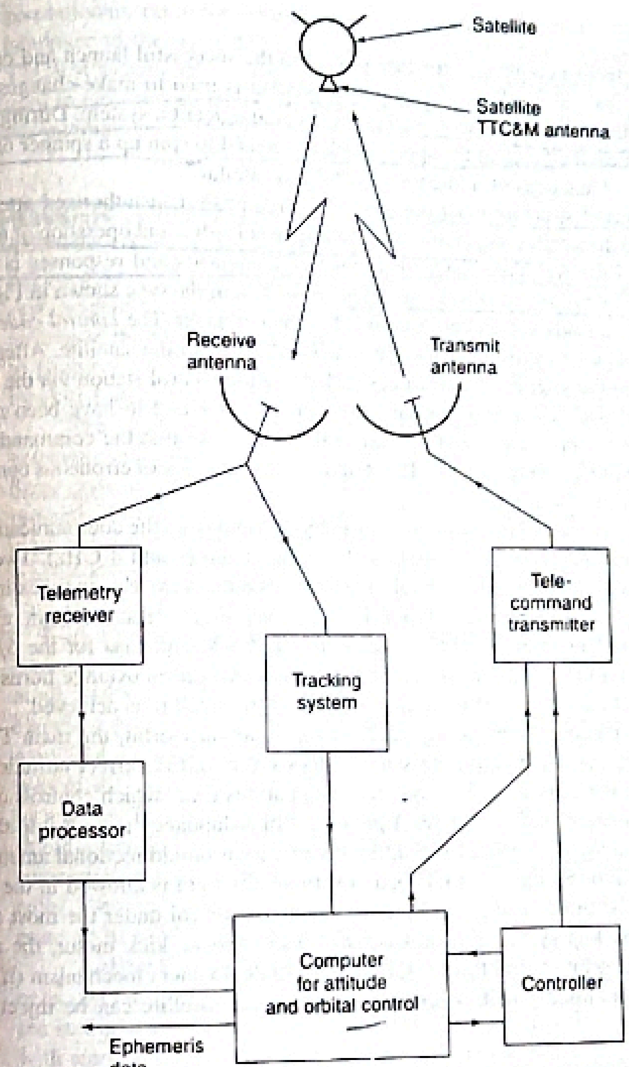


FIGURE 3.8 Typical tracking, telemetry, command and monitoring system.

by triangulation from the earth station by simultaneous range measurements. With precision equipment at the earth stations, the position of the satellite can be determined within 10 m.

Ranging tones are also used for range measurement. A carrier generated on board the satellite is modulated with a series of sine waves at increasing frequency, usually harmonically related. The phase of the sine wave modulation components is compared at an earth station, and the number of wavelengths of each frequency is calculated. Ambiguities in the numbers are resolved by reference to lower frequencies, and prior knowledge of the approximate range of the satellite. If sufficiently high frequencies are used, perhaps even the carrier frequency, range can be measured to millimeter accuracy. The technique is similar to that used in the terrestrial *tellurometer* and in aircraft radar altimeters.

Command

A secure and effective command structure is vital to the successful launch and operation of any communications satellite. The command system is used to make changes in attitude and corrections to the orbit and to control the communication system. During launch, it is used to control the firing of the apogee kick motor and to spin up a spinner or extend the solar sails and antennas of a three-axis stabilized satellite.

The command structure must possess safeguards against unauthorized attempts to make changes to the satellite's operation, and also against inadvertent operation of a control due to error in a received command. Encryption of commands and responses is used to provide security in the command system. A typical system of the type shown in Figure 3.8 will originate commands at the control terminal of the computer. The *control code* is converted into a *command word*, which is sent in a TDM frame to the satellite. After checking for validity in the satellite, the word is sent back to the control station via the telemetry link where it is checked again in the computer. If it is found to have been received correctly, an *execute* instruction will be sent to the satellite so that the command is executed. The entire process may take 5 or 10 s, but minimizes the risk of erroneous commands causing a satellite malfunction.

The command and telemetry links are usually separate from the communication system, although they may operate in the same frequency band (6 and 4 GHz). Two levels of command system are used in the Intelsat satellite: the main system operates in the 6-GHz band, in a gap between the communication channel frequencies; the main telemetry system uses a similar gap in the 4-GHz band. The TTC&M antennas for the 6/4 GHz system can be seen in Figure 3.1 on the satellite. These are earth-coverage horns, so the main system can be used only after correct attitude of the satellite is achieved.

During the launch phase and injection into geostationary orbit, the main TTC&M system may be inoperable because the satellite does not have the correct attitude or has not extended its solar sails. A backup system is used at this time, which controls only the most important sections of the satellite. A great deal of redundancy is built into this system, since its failure will jeopardize the entire mission. Near omnidirectional antennas are used at either UHF or S band (2–4 GHz), and sufficient margin is allowed in the signal-to-noise ratio (S/N) at the satellite receiver to guarantee control under the most adverse conditions. The backup system provides control of the apogee kick motor, the attitude control system and orbit control thrusters, the solar sail deployment mechanism (if fitted), and the power conditioning unit. With these controls, the satellite can be injected into geostationary orbit, turned to face the earth, and switched to full electrical power so that handover to the main TTC&M system is possible. In the event of failure of the main TTC&M system, the backup system can be used to keep the satellite on station. It is also used to eject the satellite from geostationary orbit and to switch off all transmitters when the satellite eventually reaches the end of its useful life.

SIDEBAR

Controlling a satellite in orbit is a complex process which requires considerable care. In one case, an incorrect sequence of command instructions caused loss of control of the Olympus satellite, a European large GEO satellite used for experiments in the 30/20 GHz band. The E–W thrusters fired for a lengthy period, causing the satellite to drift toward the east at 5° per day, and the satellite also began rotating with a period of 90 s. All communication with the satellite was lost, and the batteries discharged fully because the solar sails no longer pointed at the sun. The satellite drifted round the earth over a period of 2½ months, and was eventually recovered by a team of experts using large antennas in Australia and the United States to send telemetry commands to the satellite. The solar cells provided short bursts of power as they rotated past the sun's direction, allowing commands to be sent for a few seconds every 90 s. The rotation of the satellite was eventually stopped and Olympus returned to its correct location, but with shortened life expectancy due to the loss of station-keeping fuel.

3.4 POWER SYSTEMS

All communications satellites obtain their electrical power from solar cells, which convert incident sunlight into electrical energy. Some deep space planetary research satellites have used thermonuclear generators to supply electrical power, but because of the danger to people on the earth if the launch should fail and the nuclear fuel be spread over an inhabited area, communications satellites have not used nuclear generators.

The sun is a powerful source of energy. In the total vacuum of outer space, at geostationary altitude, the radiation falling on a satellite has an intensity of 1.39 kW/m². Solar cells do not convert all this incident energy into electrical power; their efficiency is typically 20 to 25% at *beginning of life* (BOL) but falls with time because of aging of the cells and etching of the surface by micrometeor impacts. Since sufficient power must be available at the *end of life* (EOL) of the satellite to supply all the systems on board, about 15% extra area of solar cells is usually provided as an allowance for aging.

A spin-stabilized satellite usually has a cylindrical body covered in solar cells. Because the solar cells are on a cylindrical surface, half of the cells are not illuminated at all, and at the edges of the illuminated half, the low angle of incidence results in little electrical power being generated. The output from the solar cells is slightly higher than would be obtained with normal incidence on a flat panel equal in area to the projected area of the cylinder, that is, its width times its height. The cells that are not illuminated by sunlight face cold space, which causes them to cool down. The solar cells on a spinner satellite have a lower temperate than those on solar sails, which increases their efficiency somewhat. Early satellites were of small dimensions and had relatively small areas of solar cells. More recently, large communications satellites for direct broadcast operation generate up to 6 kW from solar power.

A three-axis stabilized satellite can make better use of its solar cell area, since the cells can be arranged on flat panels that can be rotated to maintain normal incidence of the sunlight. Only one-third of the total area of solar cells is needed relative to a spinner, with some saving in weight. A primary advantage, however, is that by unfurling a folded solar array when the satellite reaches geostationary orbit, power in excess of 10 kW can be generated with large arrays. To obtain 10 kW from a spinner requires a very large body on which to place the solar cells, which may then exceed the maximum payload dimensions of the launch vehicle.

Solar sails must be rotated by an electric motor once per 24 h to keep the cells in full sunlight. This causes the cells to heat up, typically to 50° to 80°C, which causes a drop in output voltage. In the spinner design, the cells cool down when in shadow and run at 20° to 30°C, with somewhat higher efficiency. The bombardment of the sails by protons and electrons is also more severe, and a thicker layer of glass may be needed to slow down deterioration of the cells, with a consequent weight penalty. A rotary joint must be used with each solar sail to transfer current from the rotating sail to the body of the satellite.

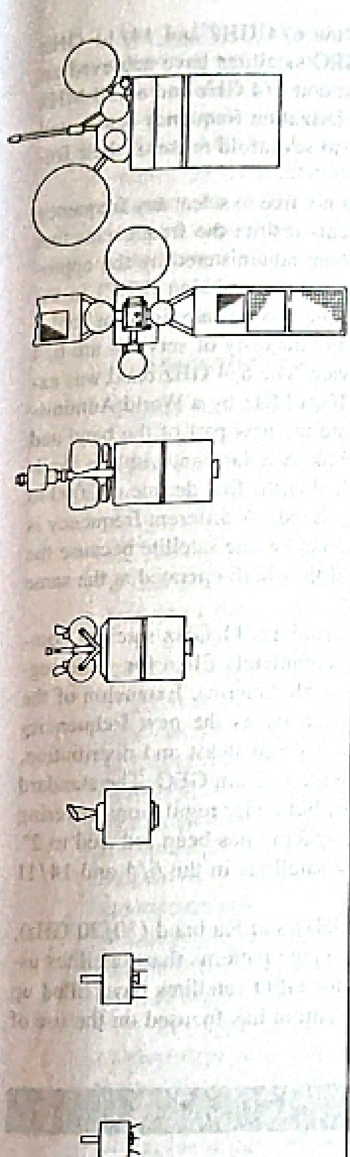
The satellite must carry batteries to power the subsystems during launch and during eclipses. Eclipses occur twice per year, around the spring and fall equinoxes, when the earth's shadow passes across the satellite, as illustrated in Figures 2.21 and 2.22. The longest duration of eclipse is 70 min, occurring around March 21 and September 21 each year. To avoid the need for large, heavy batteries, part or all of the communications system load may be shut down during eclipse, but this technique is rarely used when telephony or data traffic is carried. TV broadcast satellites may not carry sufficient battery capacity to supply their high-power transmitters during eclipse, and may shut down. By locating the satellite 20° W of the longitude of the service area, the eclipse will occur after 1 A.M. local time for the service area, when shutdown is more acceptable. Batteries are usually of the nickel–hydrogen type which do not gas when charging and have good reliability and long life, and can be safely discharged to 70% of their capacity. A power-conditioning unit controls the charging current and dumps excess current from the solar cells into heaters or load resistors on the cold side of the satellite. Sensors on the batteries, power regulator, and solar cells monitor temperature, voltage, and current and supply these data to both the onboard control system and the controlling earth station via the telemetry downlink. Typical battery voltages are 20 to 50 V with capacities of 20 to 100 ampere-hours.

3.5 COMMUNICATIONS SUBSYSTEMS

Description of the Communications System

A communications satellite exists to provide a platform in geostationary orbit for the relaying of voice, video, and data communications. All other subsystems on the satellite exist solely to support the communications system, although this may represent only a small part of the volume, weight, and cost of the satellite in orbit. Since it is the communications system that earns the revenue for the system operator, communications satellites are designed to provide the largest traffic capacity possible. The growth in capacity is well illustrated in Figure 3.9 for the Intelsat system. Successive satellites have become larger, heavier, and more costly, but the rate at which traffic capacity has increased has been much greater, resulting in a lower cost per telephone circuit or transmitted bit with each succeeding generation of satellite. The satellite transponders have limited output power and the earth stations are at least 36,000 km away from a GEO satellite, so the received power level, even with large aperture earth station antennas, is very small and rarely exceeds 10^{-10} W. For the system to perform satisfactorily, the signal power must exceed the power of the noise generated in the receiver by between 5 and 25 dB, depending on the bandwidth of the transmitted signal and the modulation scheme used. With low power transmitters, narrow receiver bandwidths have to be used to maintain the required signal-to-noise ratios.

Early communications satellites were fitted with transponders of 250 or 500 MHz bandwidth, but had low gain antennas and transmitters of 1 or 2 W output power. The earth station receiver could not achieve an adequate signal-to-noise ratio when the full bandwidth was used with the result that the system was *power limited*. Later generations of communications satellites have transponders with greatly increased output power—up to 200 W for DBS-TV satellites—and have steadily improved in bandwidth utilization efficiency, as seen in Figure 3.9. The total channel capacity of a satellite that uses a 500-MHz band at 6/4 GHz can be increased only if the bandwidth can be increased or reused. The trend in high-capacity satellites has been to reuse the available bands by employing several directional beams at the same frequency (*spatial frequency reuse*) and orthogonal polarizations at the same frequency (*polarization*



Spacecraft	INTELSAT I	INTELSAT II	INTELSAT III	INTELSAT IV	INTELSAT IV-A	INTELSAT V	INTELSAT VI
Year of first launch	1965	1967	1968	1971	1975	1980	1986 (planned)
Dimensions	0.71 m dia × 0.59 m high	1.42 m dia × 0.67 m high	1.42 m dia × 1.98 m high	2.38 m dia × 7.01 m high	2.38 m dia × 7.01 m high	15.27 m across solar sails × 6.71 m high	3.6 m dia × 11.7 m high
On orbit weight	34 kg	76 kg	152 kg	595 kg	786 kg	1020 kg	1800 kg
End of life primary power	46 W	85 W	125 W	569 W	708 W	1220 W	2100 W
Total bandwidth	50 MHz	130 MHz	300 MHz	450 MHz	720 MHz	2250 MHz	3360 MHz
Notional capacity two-way telephone circuits	240	240	1500	5000	11,000 plus 2 TV channels	24,000 plus 2 TV channels	33,000 plus 2 TV channels
Design lifetime	1.5 years	3 years	5 years	7 years	7 years	10 years	10 years
Spacecraft cost	$3.8 M	$3.5 M	$4.5 M	$14 M	$18 M	$25 M	$140 M (first five satellites)
Launch cost	$4.6 M	$4.6 M	$6 M	$20 M	$20 M	$23 M	?
Cost per telephone circuit year	$23,000	$11,000	$1,600	$810	$494	$200	?
Contractor	Hughes	Hughes	TRW	Hughes	Hughes	Ford Aerospace	Hughes

FIGURE 3.9 Illustration of the growth in size and weight of Intelsat satellites over 3 decades.

frequency reuse). Large GEO satellites also use both the 6/4 GHz and 14/11 GHz bands to obtain more bandwidth; for example, some GEO satellites have achieved an effective bandwidth of 2250 MHz within a 500-MHz band at 6/4 GHz and a 250-MHz band at 14/11 GHz by a combination of spatial and polarization frequency reuse, and later generations of Intelsat satellites have achieved up to sevenfold reuse of their frequency bands.

The designer of a satellite communication system is not free to select any frequency and bandwidth he or she chooses. International agreements restrict the frequencies that may be used for particular services, and the regulations are administered by the appropriate agency in each country—the Federal Communication Commission (FCC) in the United States, for example. Frequencies allocated to satellite services are listed in Tables 4.1 and 4.2 in Chapter 4. The bands currently used for the majority of services are 6/4 GHz and 14/11 GHz, with 30/20 GHz coming into service. The 6/4 GHz band was expanded from 500-MHz bandwidth in each direction to 1000-MHz by a World Administrative Radio Conference in 1979, but other services share the new part of the band and may cause interference to the satellite communication link. Similar bandwidth is available at 14/11 GHz, and Ka-band satellites will be launched in the first decade of 2000 to exploit the wider bandwidths available in the 30/20 GHz bands. A different frequency is required for the transmit path (normally the higher frequency) at the satellite because the high-power transmit signal would overload the receiver if they both operated at the same frequency.

The 500-MHz bands originally allocated for 6/4 and 14/11 GHz satellite communications have become very congested and are now completely filled for some segments of the geostationary orbit, such as that serving North America. Extension of the bands to 1000 MHz will eventually provide greater capacity as the new frequencies come into use. Many systems now use 14/11 GHz for TV broadcast and distribution, and 30/20 GHz systems are introducing Internet-like services from GEO. The standard spacing between GEO satellites was originally set at 3°, but under regulations covering North America and much of the rest of the world, the spacing has been reduced to 2°. The move to 2° spacing opened up extra slots for new satellites in the 6/4 and 14/11 GHz bands.

Satellite systems designed for Ku band (14/11 GHz) and Ka band (30/20 GHz), have narrower antenna beams, and better control of coverage patterns than satellites using C band (6/4 GHz). As the available orbital slots for GEO satellites have filled up with satellites using the 6/4 and 14/11 GHz bands, attention has focused on the use of the 30/20 GHz band. Originally, this band had 3 GHz bandwidth allocated to satellite services, but part of the band was reallocated to the Land Multipoint Distribution Service (LMDS). Approximately 2 GHz of bandwidth is still available for satellite systems at Ka band on an exclusive or shared basis, which is equal to the combined allocations of C and Ku bands. However, propagation in rain becomes a major factor at frequencies above 10 GHz. Attenuation (in dB) in rain increases at roughly the square of the frequency, so at 20 GHz rain attenuation is four times larger, in dB, than at 10 GHz. It is almost impossible to provide rain attenuation margins larger than 20 dB in any system, and many Ka-band systems will have margins of only 4 or 5 dB when operating to small earth station antennas.

SIDEBAR

Ka-band satellite links must accept larger outage times due to rain than 14/11 GHz links, and are therefore better suited to data transmission than voice. The transmission of data can be delayed when a rain fade affects the link, but a telephone conversation will be ended if the link fails for more than a few seconds. In the time frame 2001–2010 several Ka-band satellite systems will come into service. The major market appears to be data relay, particularly Internet access. Satellites support asymmetric data links such as Internet service to individual users very well. The user wants high-speed, high volume data delivery from the Internet to send video clips and high resolution photographs (known generically as multimedia), but does not send equivalent volumes of data into the Internet. A lower capacity link from the user is adequate, and this suits Ka-band links with large hub antennas and small user terminals. The link will suffer occasional outages, perhaps for a total of 25 h in a year, when heavy rain is falling through the path between the satellite and the user's small terminal.

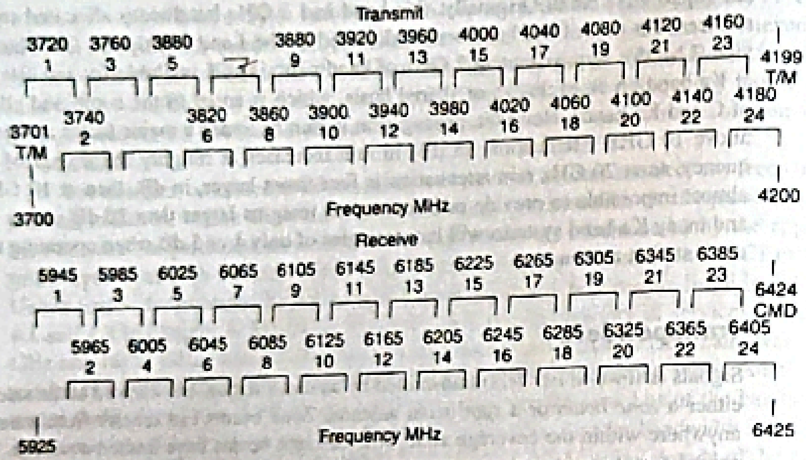
Transponders

Signals (known as *carriers*) transmitted by an earth station are received at the satellite by either a *zone beam* or a *spot beam* antenna. Zone beams can receive from transmitters anywhere within the coverage zone, whereas spot beams have limited coverage. The received signal is often taken to two low noise amplifiers and is recombined at their output to provide *redundancy*. If either amplifier fails, the other one can still carry all the traffic. Since all carriers from one antenna must pass through a low noise amplifier, a failure at that point is catastrophic. Redundancy is provided wherever failure of one component will cause the loss of a significant part of the satellite's communication capacity.

Figure 3.10 shows a simplified block diagram of a satellite communication subsystem for the 6/4 GHz band. The 500-MHz bandwidth is divided up into channels, often 36 MHz wide, which are each handled by a separate transponder. A transponder consists of a band-pass filter to select the particular channel's band of frequencies, a downconverter to change the frequency from 6 GHz at the input to 4 GHz at the output, and an output amplifier. The communication system has many transponders, some of which may be spares; typically 12 to 44 active transponders are carried by a high-capacity satellite. The transponders are supplied with signals from one or more receive antennas and send their outputs to a switch matrix that directs each transponder band of frequencies to the appropriate antenna or antenna beam. In a large satellite there may be four or five beams to which any transponder can be connected. The switch setting can be controlled from the earth to allow reallocation of the transponders between the downlink beams as traffic patterns change.

In the early satellites such as INTELSAT I and II, one or two 250-MHz bandwidth transponders were employed. This proved unsatisfactory because of the nonlinearity of the traveling wave tube transmitter used at the output of the transponder, and later GEO satellites have used up to 44 transponders each with 36, 54, or 72 MHz bandwidth. The reason for using narrower bandwidth transponders is to avoid excessive *intermodulation* problems when transmitting several carriers simultaneously with a nonlinear transmitter, as discussed in Chapter 6. Intermodulation distortion is likely to occur whenever a high power amplifier is driven close to saturation. Since we generally want to have more than one earth station transmitter sending signals via a satellite, one solution would be to provide one transponder for each earth station's signal. In the case of the Intelsat global system, this could result in a requirement for as many as 100 transponders per satellite. As a compromise, 36 MHz has been widely used for transponder bandwidth, with 54 and 72 MHz adopted for some satellites.

Many domestic satellites operating in the 6/4 GHz band carry 24 active transponders. The center frequencies of the transponders are spaced 40 MHz apart, to allow



Frequency plan. Note: Number below channel center frequency refers to transponder identity.

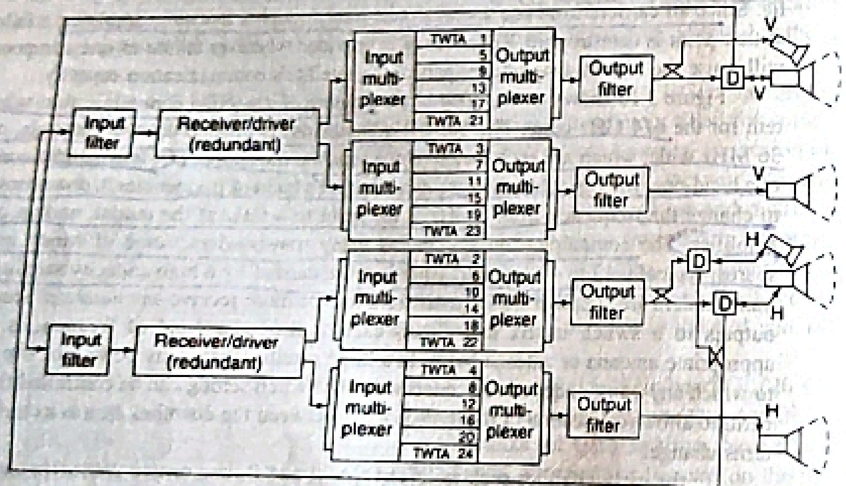


FIGURE 3.10 Transponder arrangement of RCA's SATCOM satellites and frequency plan. The translation frequency is 2225 MHz. [Reproduced with permission from W. H. Braun and J. E. Keigler, "RCA Satellite Networks: High Technology and Low User Cost," *Proceedings of the IEEE, 72*, 1483–1505 (November 1984). Copyright © 1984 IEEE.]

guard bands for the 36 MHz filter skirts. With a total of 500 MHz available, a single polarization satellite can accommodate 12 transponders across the band. When *frequency reuse by orthogonal polarizations* is adopted, 24 transponders can be accommodated in the same 500 MHz bandwidth. Traditional linear transponder-type satellites now have sixfold reuse (INTELSAT VI and IX) or even sevenfold reuse (INTELSAT VIII) at C band. The reuse is achieved through microwave switch interconnections between subbeams. Internet-like satellites need a plethora of beam interconnections—more than 50 in most cases. The only way to achieve this level of beam/path interconnections is via *on board processing* (OBP).

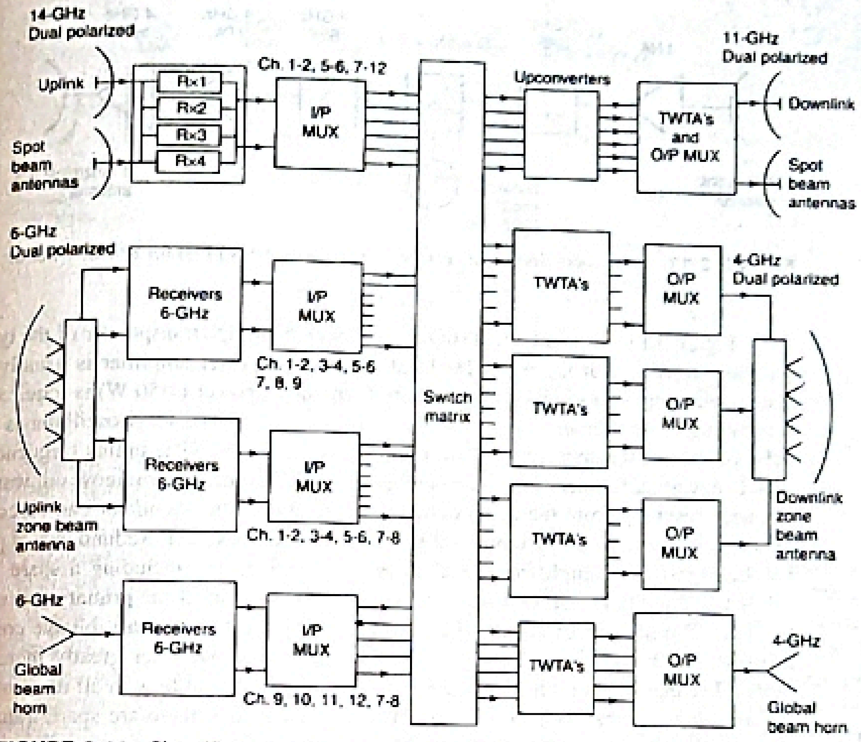


FIGURE 3.11 Simplified block diagram of an INTELSAT V communication system. Note that the switch matrix allows many possible interconnections between uplink beams and downlink transmitters. (Courtesy C. F. Hoeber, Ford Aerospace and Communications Corp.)

Figure 3.11 shows a simplified diagram of the communication system carried by INTELSAT V satellites. The later series of Intelsat satellites use a similar arrangement. The bulk of the traffic is carried by the 6/4 GHz section, with a total bandwidth of 2000 MHz available by frequency reuse. The switch matrix allows a very large number of variations in connecting the 6-GHz receivers to the 4-GHz transmitters, and also interconnects the 6/4 and 14/11-GHz sections. This provides Intelsat with a great deal of flexibility in setting up links through the satellite.

When more than one signal shares a transponder (using *frequency division multiple access*, FDMA) the power amplifier must be run below its maximum output power to maintain linearity and reduce *intermodulation* products. The degree to which the transmitter output power is reduced below its peak output is known as *output backoff*: in FDMA systems, 2 to 7 dB of output backoff is typically used, depending on the number of accesses to the transponder and the extent to which the characteristics of the HPA have been linearized. Backoff results in a lower downlink carrier-to-noise ratio at the earth station with FDMA when multiple accesses to each transponder are required. *Time division multiple access* (TDMA) can theoretically be used to increase the output power of transponders by limiting the transponder to a single access. However, most TDMA systems are hybrid FDMA–TDMA schemes known as multifrequency TDMA (MF-TDMA), in which several TDMA signals share the transponder bandwidth using FDMA. Linearity of the HPA remain an issue for MF-TDMA systems.

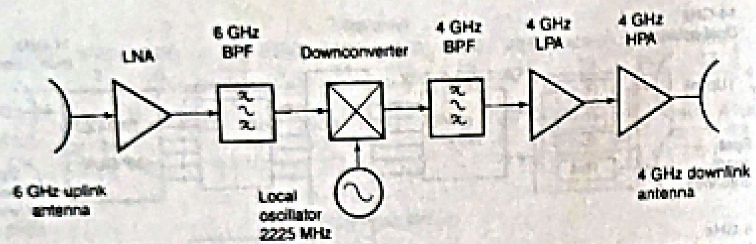


FIGURE 3.12 Simplified single conversion transponder (bent pipe) for 6/4 GHz band.

Figure 3.12 shows a typical single conversion bent pipe transponder of the type used on many satellites for the 6/4 GHz band. The output power amplifier is usually a solid state power amplifier (SSPA) unless a very high output power (>50 W) is required, when a *traveling wave tube amplifier* (TWTA) would be used[9]. The local oscillator is at 2225 MHz to provide the appropriate shift in frequency from the 6-GHz uplink frequency to the 4-GHz downlink frequency, and the band-pass filter after the mixer removes unwanted frequencies resulting from the down-conversion operation. The attenuator can be controlled via the uplink command system to set the gain of the transponder. Redundancy is provided for the high-power amplifiers (HPA) in each transponder by including a spare TWT or solid-state amplifier (SSPA) that can be switched into circuit if the primary power amplifier fails. The lifetime of HPAs is limited, and they represent the least reliable component in most transponders. Providing a spare HPA in each transponder greatly increases the probability that the satellite will reach the end of its working life with all its transponders still operational. Transponders can also be arranged so that there are spare transponders available in the event of a total failure. The arrangement is known as *M for N redundancy*. For example, it is common to have 16 for 10 redundancy or even 14 for 10. That is, 16 (or 14) output amplifiers are connected in a ring such that any of the 10 signals can pass through them. Thus, 6 (or 4) amplifiers are acting as back-up amplifiers while 10 are on line. Most HPAs have bandwidths much larger than the allocated frequency band and so it matters little which signals are passing through them. At Ku band, ring redundancy is still used, but it is much more like 2 for 1, that is, one spare for every active unit.

Transponders for use in the 14/11-GHz bands normally employ a double frequency conversion scheme as illustrated in Figure 3.13. It is easier to make filters, amplifiers, and equalizers at an intermediate frequency (IF) such as 1100-MHz than at 14 or 11 GHz, so the incoming 14-GHz carrier is translated to an IF of around 1 GHz. The amplification and filtering are performed at 1 GHz and a relatively high-level carrier is translated back to 11 GHz for amplification by the HPA.

Stringent requirements are placed on the filters used in transponders, since they must provide good rejection of unwanted frequencies, such as intermodulation products, and also have very low amplitude and phase ripple in their pass bands. Frequently a filter will be followed by an equalizer that smoothes out amplitude and phase variations in the pass band. Phase variation across the pass band produces *group delay distortion*, which is particularly troublesome with wideband FM signals and high-speed phase shift keyed data transmissions.

A considerable increase in the communications capacity of a satellite can be achieved by combining onboard processing with switched-beam technology. A switched-beam satellite generates a narrow transmit beam for each earth station with which it communicates, and then transmits sequentially to each one using time division multiplexing of the signals. The narrow beam has to cover only one earth station, allowing the satellite transmit antenna to have a very high gain compared to a zone-coverage antenna. A narrow scanning beam can also be used, or a combination of fixed and scanning beams. Unless the satellite has a

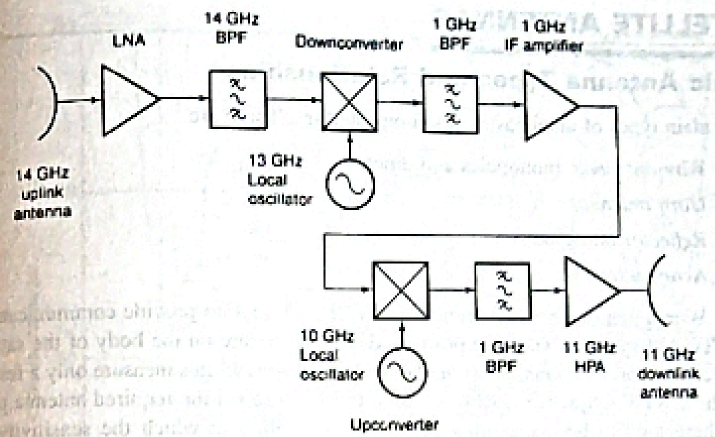


FIGURE 3.13 Simplified double conversion transponder (bent pipe) for 14/11 GHz band.

zone-coverage receiver antenna, data storage is required at the satellite since it communicates with only one earth station at a time. The high gain antennas used in switched-beam systems raise the EIRP (effective isotropically radiated power) of the satellite transmitter and thus increase the capacity of the downlink. Switched beam systems on GEO satellites work best at Ka band where the wavelength is short enough that the limited dimensions of antennas on the satellite still allow beams of less than 0.4° beamwidth to be generated. Multiple beam antennas with baseband processing transponders are used on GEO and LEO satellites providing service to mobile terminals and handheld telephones[10]. The low gain of the near omnidirectional antenna of a mobile earth station must be compensated by a high gain antenna on the satellite, necessitating the use of multiple beam antennas.

It is possible to conserve uplink bandwidth by using different modulation techniques on the uplink and downlink and by providing a baseband processor on the satellite. A high level modulation such as 16-QAM with four bits per symbol can be used on the link between the satellite and a large earth station to improve bandwidth efficiency. This approach has been adopted in the Astrolink and Spaceway 30/20 GHz satellites[16,17].

Onboard processing may also be used to advantage to switch between the uplink access technique (e.g., MF-TDMA) and the downlink access technique (e.g., TDM) so that small earth stations may access each other directly via the satellite. The processor can provide the data storage needed for a switched-beam system and also can perform error correction independently on the uplink and downlink. A typical arrangement of the communication system for a satellite employing onboard processing is shown in Figure 3.14.

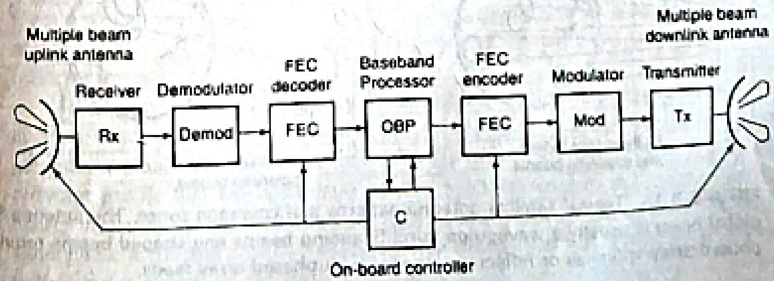


FIGURE 3.14 Onboard processing transponder.

3.6 SATELLITE ANTENNAS

Basic Antenna Types and Relationships

Four main types of antennas are used on satellites. These are

1. *Wire antennas*: monopoles and dipoles.
2. *Horn antennas*.
3. *Reflector antennas*.
4. *Array antennas*.

Wire antennas are used primarily at VHF and UHF to provide communications for the TTC&M systems. They are positioned with great care on the body of the satellite in an attempt to provide *omnidirectional* coverage. Most satellites measure only a few wavelengths at VHF frequencies, which makes it difficult to get the required antenna patterns, and there tend to be some orientations of the satellite in which the sensitivity of the TTC&M system is reduced by *nulls* in the antenna pattern.

An *antenna pattern* is a plot of the field strength in the far field of the antenna when the antenna is driven by a transmitter. It is usually measured in *decibels* (dB) below the maximum field strength. The *gain* of an antenna is a measure of the antenna's capability to direct energy in one direction, rather than all around. *Antenna gain* is defined in Chapter 4, Section 4.2. At this point, it will be used with the simple definition given above. A useful principle in antenna theory is *reciprocity*. Reciprocity means that an antenna has the same gain and pattern at any given frequency whether it transmits or receives. An antenna pattern measured when receiving is identical to the pattern when transmitting.

Figure 3.15 shows typical satellite antenna coverage zones. The pattern is frequently specified by its 3-dB beamwidth, the angle between the directions in which the radiated

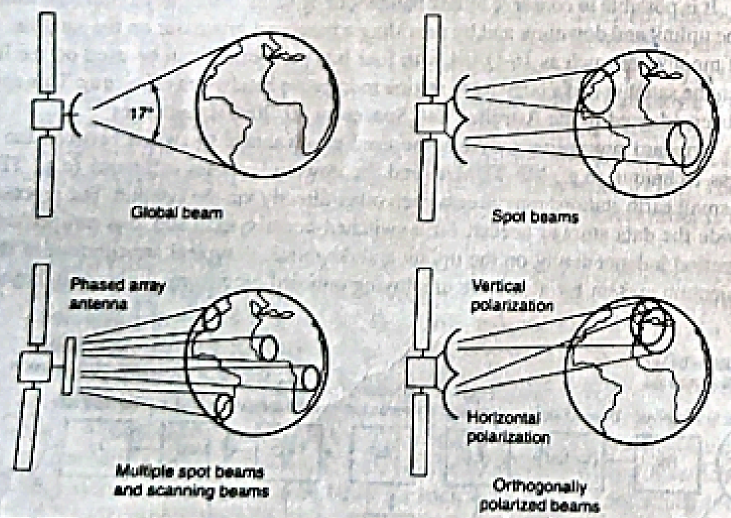


FIGURE 3.15 *Typical satellite antenna patterns and coverage zones. The antenna for the global beam is usually a waveguide horn. Scanning beams and shaped beams require phased array antennas or reflector antennas with phased array feeds.*

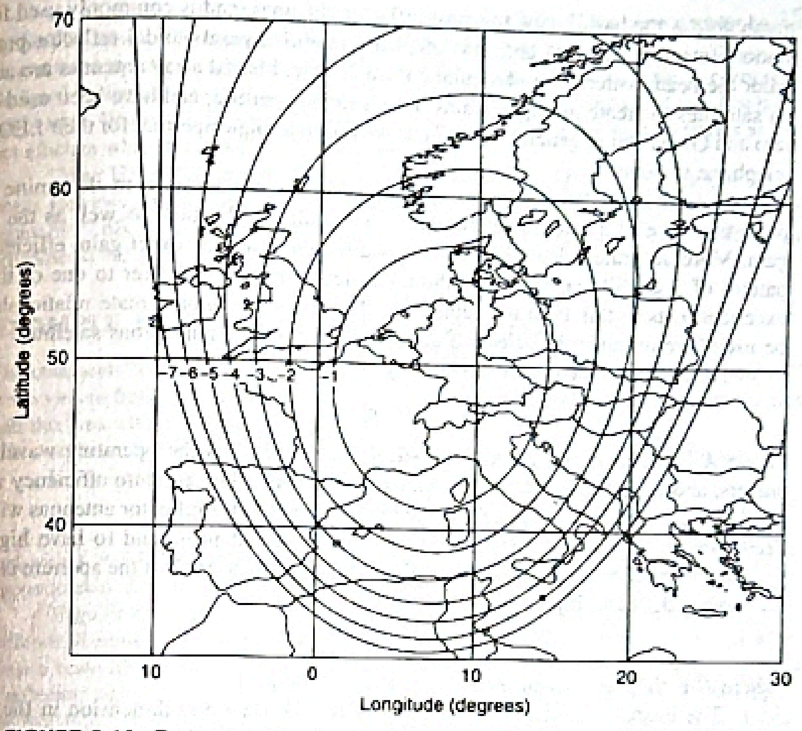


FIGURE 3.16 Typical coverage patterns for Intelsat satellites over the Atlantic Ocean.

(or received) field falls to half the power in the direction of maximum field strength. However, a satellite antenna is used to provide coverage of a certain area, or *zone* on the earth's surface, and it is more useful to have contours of antenna gain as shown in Figure 3.16.

When computing the signal power received by an earth station from the satellite, it is important to know where the station lies relative to the satellite transmit antenna contour pattern, so that the exact EIRP can be calculated. If the pattern is not known, it may be possible to estimate the antenna gain in a given direction if the antenna *boresight* or *beam axis* direction and its beamwidth are known.

Horn antennas are used at microwave frequencies when relatively wide beams are required, as for global coverage. A horn is a flared section of waveguide that provides an aperture several wavelengths wide and a good match between the waveguide impedance and free space. Horns are also used as feeds for reflectors, either singly or in clusters. Horns and reflectors are examples of *aperture antennas* that launch a wave into free space from a waveguide. It is difficult to obtain gains much greater than 23 dB or beamwidths narrower than about 10° with horn antennas. For higher gains or narrow beamwidths a reflector antenna or array must be used.

Reflector antennas are usually illuminated by one or more horns and provide a larger aperture than can be achieved with a horn alone. For maximum gain, it is necessary to generate a plane wave in the aperture of the reflector. This is achieved by choosing a reflector profile that has equal path lengths from the feed to the aperture, so that all the energy radiated by the feed and reflected by the reflector reaches the aperture with the same phase angle and creates a uniform phase front. One reflector shape that achieves this with a point source of radiation is the paraboloid, with a feed placed at its focus. The paraboloid is the basic shape for most reflector antennas, and is commonly used for earth station antennas. Satellite antennas often use modified paraboloidal reflector profiles to tailor the beam pattern to a particular coverage zone. Phased array antennas are also used on satellites to create multiple beams from a single aperture, and have been used by Iridium and Globalstar to generate up to 16 beams from a single aperture for their LEO mobile telephone systems[11].

Some basic relationships in aperture antennas can be used to determine the approximate size of a satellite antenna for a particular application, as well as the antenna gain. More accurate calculations are needed to determine the exact gain, efficiency, and pattern of a satellite antenna, and the interested reader should refer to one of the many excellent texts in this field for details.[12-14] The following approximate relationships will be used here to guide the selection of antennas for a communications satellite.

An aperture antenna has a gain G given by

$$G = \eta_A 4\pi A/\lambda^2 \tag{3.1}$$

where A is the area of the antenna aperture in meters, λ is the operating wavelength in meters, and η_A is the *aperture efficiency* of the antenna. The aperture efficiency η_A is not easily determined, but is typically in the range 55 to 68% for reflector antennas with single feeds, lower for antennas with shaped beams. Horn antennas tend to have higher efficiencies than reflector antennas, typically in the range 65 to 80%. If the aperture is circular, as is often the case, Eq. (3.1) can be written as

$$\boxed{G = \eta_A(\pi D/\lambda)^2} \tag{3.2}$$

where D is the diameter of the circular aperture in meters.

The beamwidth of an antenna is related to the aperture dimension in the plane in which the pattern is measured. A useful rule of thumb is that the 3 dB beamwidth in a given plane for an antenna with dimension D in that plane is

$$\boxed{\theta_{3\,dB} \approx 75\lambda/D \text{ degrees}} \tag{3.3}$$

where $\theta_{3\,dB}$ is the beamwidth between half power points of the antenna pattern and D is the aperture dimension in the same units as the wavelength λ. The beamwidth of a horn antenna may depart from Eq. (3.2) quite radically. For example a small rectangular horn will produce a narrower beam than suggested by Eq. (3.2) in its E plane and a wider beamwidth in the H plane.

Since both Eqs. (3.2) and (3.3) contain antenna dimension parameters, the gain and beamwidth of an aperture antenna are related. For antennas with $\eta_A \approx 60\%$, the gain is approximately

$$G \approx 33{,}000/(\theta_{3\,dB})^2 \tag{3.4}$$

where $\theta_{3\,dB}$ is in degrees and G is not in decibels. If the beam has different beamwidths in orthogonal planes, $\theta_{3\,dB}$ should be replaced by the product of the two 3 dB beamwidths. Values of the constant in Eq. (3.3) vary between different sources, with a range 28,000 to 35,000. The value 33,000 is typical for reflector antennas used in satellite communication systems.

EXAMPLE 3.6.1 *Global Beam Antenna*

The earth subtends an angle of 17° when viewed from geostationary orbit. What are the dimensions and gain of a horn antenna that will provide global coverage at 4 GHz?

If we design our horn to give a circularly symmetric beam with a 3-dB beamwidth of 17° using Eq. (3.2)

$$D/\lambda = 75/(\theta_{3\,dB}) = 4.4$$

At 4 GHz, $\lambda = 0.075$ m, so $D = 0.33$ m (just over 1 ft). If we use a circular horn excited in the TE_{11} mode, the beamwidths in the E and H planes will not be equal and we may be forced to make the aperture slightly smaller to guarantee coverage in the E plane. *A corrugated horn* designed to support the HE hybrid mode has a circularly symmetric beam and could be used in this application. Waveguide horns are generally used for global beam coverage. Reflector antennas are not efficient when the aperture diameter is less than 8λ.

Using Eq. (3.4), the gain of the horn is approximately 100, or 20 dB, at the center of the beam. However, in designing our communication system we will have to use the edge of beam gain figure of 17 dB, since those earth stations close to the earth's horizon, as viewed from the satellite, are close to the 3 dB contour of the transmitted beam. ■

EXAMPLE 3.6.2 *Regional Coverage Antenna*

The continental United States (48 contiguous states) subtends an angle of approximately 6° × 3° when viewed from geostationary orbit. What dimension must a reflector antenna have to illuminate half this area with a circular beam 3° in diameter at 11 GHz?
Can a reflector be used to produce a 6° × 3° beam? What gain would the antenna have?

Using Eq. (3.2), we have for a 3° circular beam

$$D/\lambda = 75/3 = 25$$

and with $\lambda = 0.0272$ m, $D = 0.68$ m (just over 2 ft). The gain of this antenna, from Eq. (3.3) is approximately 35 dB.

To generate a beam with different beamwidths in orthogonal planes we need an aperture with different dimensions in the two planes. In this case, a rectangular aperture $25\lambda \times 12.5\lambda$ would generate a beam 6° × 3°, and would have a gain of 32 dB, approximately. In order to illuminate such a reflector, a horn with unequal beamwidths is required, since the reflector must intercept most of the radiation from the feed if it is to have an acceptable efficiency. Rectangular, or more commonly elliptical, outline reflectors are used to generate unequal beamwidths. When orthogonal polarizations are to be transmitted or received, it is better to use a circular reflector with a distorted profile to broaden the beam in one plane, or a feed cluster to provide the appropriate amplitude and phase distribution across the reflector. ■

Satellite Antennas in Practice (SPACE CRAFT ANTENNAS)

The antennas of a communications satellite are often a limiting element in the complete system. In an ideal satellite, there would be one antenna beam for each earth station, completely isolated from all other beams, for transmit and receive. However, if two earth stations are 300 km apart on the earth's surface and the satellite is in geostationary orbit, their angular separation at the satellite is 0.5°. For $\theta_{3\,dB}$ to be 0.5°, D/λ must be 150, which requires an aperture diameter of 11.3 m at 4 GHz. Antennas this large have been flown on satellites (ATS-6 deployed a 2.5 GHz, 10-m diameter antenna, for example), and large unfurled antennas are used to create multiple spot beams from GEO satellites serving mobile users. However, at 20 GHz, an antenna with $D/\lambda = 150$ is only 1.5 m wide, and such an antenna can readily be flown on a 30/20 GHz satellite. A phased array feed is used to create many 0.5° beams which can be clustered to serve the coverage zone of the satellite.

To provide a separate beam for each earth station would also require one antenna feed per earth station if a multiple-feed antenna with a single reflector were used. A compromise between one beam per station and one beam for all stations has been used in many satellites by using zone-coverage beams and orthogonal polarizations within the same beam to provide more channels per satellite. Figure 3.3b shows a GEO satellite that

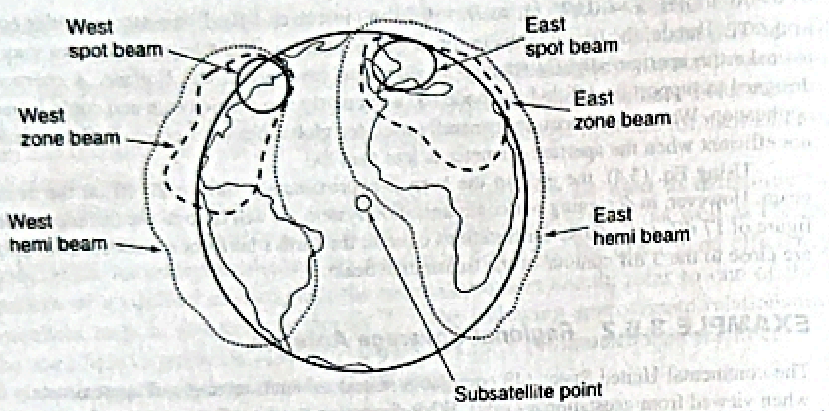


FIGURE 3.17 Contour plot of the spot beam of ESA's OTS satellite projected onto the earth. The contours are in 1 dB steps, normalized to 0 dB at the center of the beam. (Courtesy of ESA.)

has four reflector antennas. Each reflector is illuminated by a complex feed that provides the required beam shape to permit communication between earth stations within a given coverage zone. Figure 3.17 shows the coverage zones provided by a typical Intelsat satellite. The largest reflector on the satellite transmits at 4 GHz and produces the "peanut" shaped patterns for the zone beams, which are designed to concentrate the transmitted energy onto densely populated areas such as North America and western Europe where much telecommunications traffic is generated. The smaller antennas are used to provide hemisphere transmit and receive beams, and the 14/11 GHz spot beams. In addition, there are horn antennas providing global beam coverage.

Countries such as the United States create an enormous demand for communication services, and a number of domestic satellite communication systems have been established to meet that demand. In 2000 the geostationary orbit had domestic satellites spaced every 2°, operating at 6/4 GHz and 14/11 GHz from longitude 60° W to 140° W. This encompasses all orbital locations that can be simultaneously viewed by earth stations in the United States and Canada, and each operator has been given a limited number of *orbital slots* in which to place a satellite. As a result, there is a great deal of pressure on the operating companies to obtain the maximum number of channels per satellite in order to give the operator the greatest possible revenue-earning capacity. This has encouraged the development of frequency reuse antennas by means of orthogonal polarizations and multiple beams, the combination of 6/4 and 14/11 GHz communication systems on one satellite, and the use of multilevel digital modulation and TDMA to increase capacity.

SIDEBAR

Some of the largest commercial GEO satellites pro-
…osed to date are the Inmarsat I-4 series[19]*. These satel-*
…ites are based on the Matra Marconi Space Eurostar
…000 platform, a 5 metric ton three-axis stabilized
…tellite with 9 kW onboard power. The satellites have 200 spot beams for mobile services and Internet access. Three Inmarsat I-4 satellites are to be built by the European consortium Astrium at a total cost of $700 M, with two scheduled for launch in 2003–2004. The third satellite is a spare.

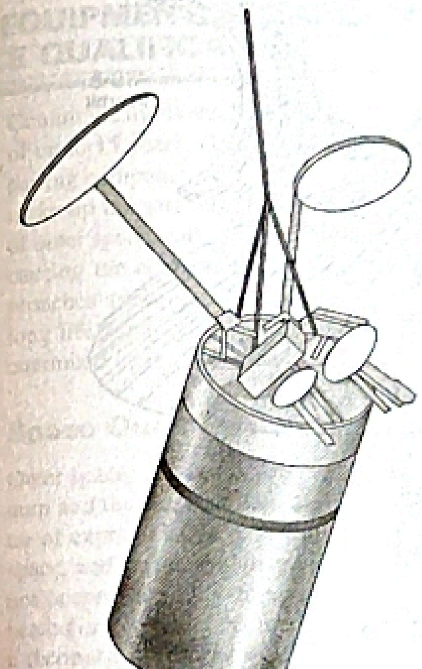


FIGURE 3.18 Intelsat VI satellite on station.

The requirements of narrow antenna beams with high gain over a small coverage zone leads to large antenna structures on the satellite. Frequently, the antennas in their operating configuration are too large to fit within the shroud dimensions of the launch vehicle, and must be folded down during the launch phase. Once in orbit, the antennas then can be deployed. In many larger satellites, the antennas use offset paraboloidal reflectors with clusters of feeds to provide carefully controlled beam shapes. The feeds mount on the body of the satellite, close to the communications subsystem, and the reflector is mounted on a hinged arm. Figure 3.18 shows an example of this design of antenna for the INTELSAT VI satellite. For launch, the solid reflectors fold down to provide a compact structure; in orbit, the hinged arms are swung out and locked in place to hold the reflectors in the correct position. When the satellite is in geostationary orbit it is weightless, so very little energy is required to move the large reflector.

SIDEBAR

One interesting idea is the inflatable antenna, several examples of which have been flown experimentally[18]. The antenna can be squeezed into a small space for launch and inflated from a pressurized gas bottle when the satellite is in orbit. Once inflated, a foam material emitted along with the inflation gas hardens to make a rigid structure. Plastic materials can be sprayed with a metallic coating made up of very small particles of aluminum to create a reflecting surface for electromagnetic (EM) waves. Inflatable antennas can be made very large without a significant weight penalty, and are therefore attractive for any satellite requiring multiple narrow beams.

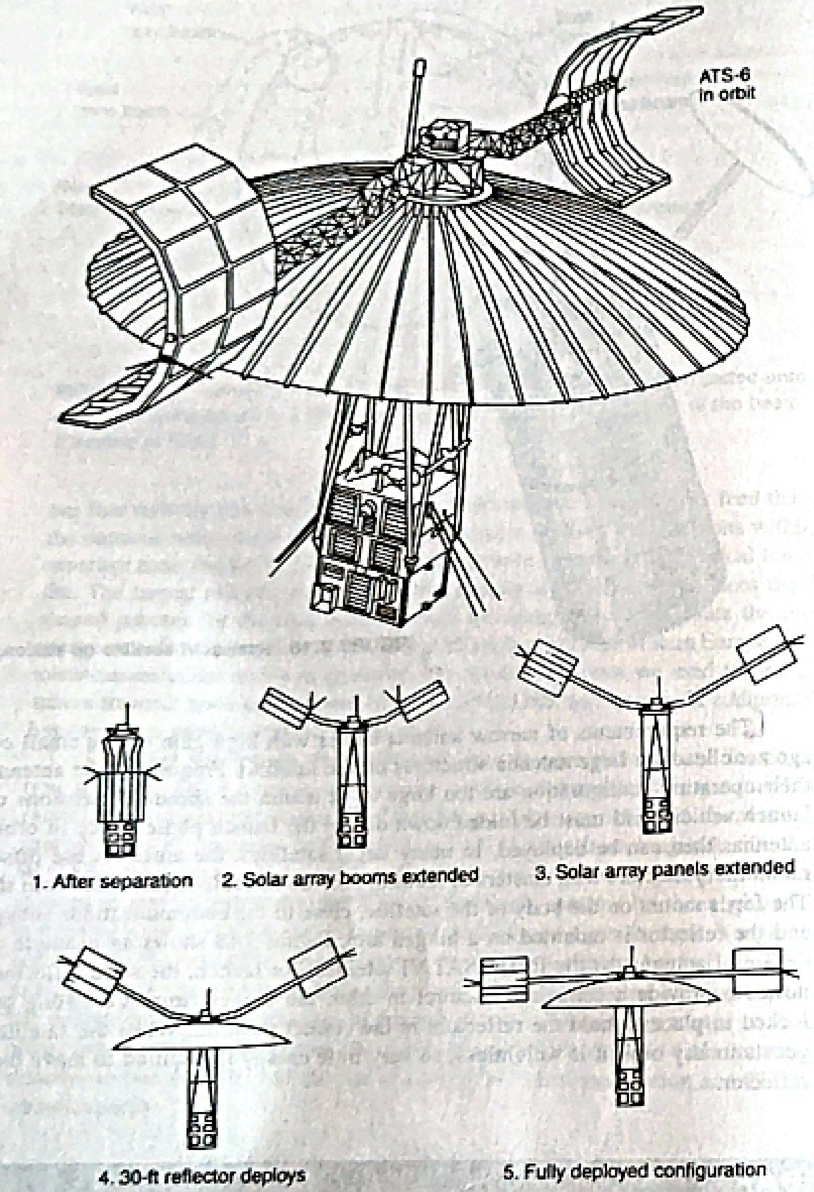


FIGURE 3.19 Deployment sequence of ATS-6 10-m antenna. (Courtesy of NASA.)

Figure 3.19 shows the deployment sequence used for the 30-ft antenna carried by ATS-6: the antenna was built as a series of petals that folded over each other to make a compact unit during launch, which then unfurled in orbit. The solar sails folded down over the antenna, and were deployed first. Springs or pyrotechnic devices can be used to provide the energy for deployment of antennas or solar sails, with a locking device to ensure correct positioning after deployment. Similar unfurlable antennas are used on GEO satellites that provide satellite telephone service at L band using multiple narrow beams

3.7 EQUIPMENT RELIABILITY AND SPACE QUALIFICATION

Communications satellites built in the 1980s and 1990s have provided operational lifetimes of up to 15 years. Once a satellite is in geostationary orbit, there is little possibility of repairing components that fail or adding more fuel for station keeping. The components that make up the satellite must therefore have very high reliability in the hostile environment of outer space, and a strategy must be devised that allows some components to fail without causing the entire communication capacity of the satellite to be lost. Two separate approaches are used: *space qualification* of every part of the satellite to ensure that it has a long life expectancy in orbit and *redundancy* of the most critical components to provide continued operation when one component fails.

Space Qualification

Outer space, at geostationary orbit distances, is a harsh environment. There is a total vacuum and the sun irradiates the satellite with 1.4 kW of heat and light on each square meter of exposed surface. Where surfaces are in shadow, heat is lost to the infinite sink of space, and surface temperature will fall toward absolute zero. Electronic equipment cannot operate at such extremes of temperature and must be housed within the satellite and heated or cooled so that its temperature stays within the range 0° to 75°C. This requires a thermal control system that manages heat flow throughout a GEO satellite as the sun moves around once every 24 h. Thermal problems are equally severe for a LEO satellite that moves from sunlight to shadow every 100 min.

The first stage in ensuring high reliability in a satellite is by selection and screening of every component used. Past operational and test experience of components indicates which components can be expected to have good reliability. Only components that have been shown to have high reliability under outer space conditions will be selected. Each component is then tested individually (or as a subsystem) to ensure that it meets its specification. This process is known as *quality control* or *quality assurance* and is vital in building any equipment that is to be reliable. Once individual components and subsystems have been space qualified, the complete satellite must be tested as a system to ensure that its many systems are reliable.

When a satellite is designed, three prototype models are often built and tested. The *mechanical model* contains all the structural and mechanical parts that will be included in the satellite and is tested to ensure that all moving parts operate correctly in a vacuum, over a wide temperature range. It is also subjected to vibration and shock testing to simulate vibration levels and G forces likely to be encountered on launch. The *thermal model* contains all the electronics packages and other components that must be maintained at the correct temperature. Often, the thermal, vacuum, and vibration tests of the entire satellite will be combined in a thermal vacuum chamber for what is known in the industry as a *shake and bake* test. The antennas are usually included on the thermal model to check for distortion of reflectors and displacement or bending of support structures. In orbit, an antenna may cycle in temperature from above 100°C to below −100°C as the sun moves around the satellite. The *electrical model* contains all the electronic parts of the satellite and is tested for correct electrical performance under total vacuum and a wide range of temperatures. The antennas of the electrical model must provide the correct beamwidth, gain, and polarization properties.

Testing carried out on the prototype models is designed to overstress the system and induce failure in any weak components: temperature cycling will be carried out to 10% beyond expected extremes; structural loads and G forces 50% above those expected in

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flight may be applied. Electrical equipment will be subjected to excess voltage and current drain to test for good electronic and thermal reliability. The prototype models used in these tests will not usually be flown. A separate flight model (or several models) will be built and subjected to the same tests as the prototype, but without the extremes of temperature, stress, or voltage. Preflight testing of flight models, while exhaustive, is designed more to cause failure of parts, rather than to check that they will operate under worst-case conditions.

Space qualification is an expensive process, and one of the factors that makes large GEO satellites expensive. Some low earth orbit satellites have been built successfully using less expensive techniques and relying on lower performance in orbit. LEO satellite systems require large numbers of satellites that are generally less expensive than large GEO satellites. The Iridium system, for example, was designed with 66 operational satellites in its constellation to provide continuous worldwide coverage, with at least eight spare satellites in orbit at any time. If one operational satellite fails, a spare is moved in to take its place. This allowed Iridium satellites to be built with a higher probability of failure than a GEO satellite. Experimental satellites have also been built using low cost techniques. The University of Surrey, U.K., for example, built a series of digital store and forward satellites that were used by radio amateurs and others which each cost less than $1 M[15]. Most of the components on the Surrey satellites were not space qualified, but were selected carefully to ensure best possible lifetimes at reasonable cost and then the entire satellite was subjected to shake and bake tests.

Many of the electronic and mechanical components that are used in satellites are known to have limited lifetimes, or a finite probability of failure. If failure of one of these components will jeopardize the mission or reduce the communication capacity of the satellite, a backup, or *redundant*, unit will be provided. The design of the system must be such that when one unit fails, the backup can automatically take over or be switched into operation by command from the ground. For example, redundancy is always provided for traveling wave tube amplifiers used in the transponders of a communications satellite, as these are known to have a limited lifetime.

The success of the testing and space qualification procedures used by NASA has been well illustrated by the lifetime achieved by many of its scientific satellites. Satellites designed for a specific mission lasting 1 or 2 years have frequently operated successfully for up to 25 years. Sufficient reliability was designed into the satellite to guarantee the mission lifetime such that the actual lifetime has been much greater. In the next section we will look at how reliability can be quantified.

Reliability

We need to be able to calculate the reliability of a satellite subsystem for two reasons: we want to know what the probability is that the subsystem will still be working after a given time period, and we need to provide redundant components or subsystems where the probability of a failure is too great to be accepted. The owner of a satellite used for communications expects to be able to use a predetermined percentage of its communications capacity for a given length of time. Amortization of purchase and launch costs will be calculated on the basis of an expected *lifetime.* The manufacturers of satellites must provide their customers with predictions (or guarantees) of the reliability of the satellite and subsystems: to do this requires the use of *reliability theory.* Reliability theory is a mathematical attempt to predict the future and is therefore less certain than other mathematical techniques that operate in absolute terms. The application of reliability theory has enabled satellite engineers to build satellites that perform as expected

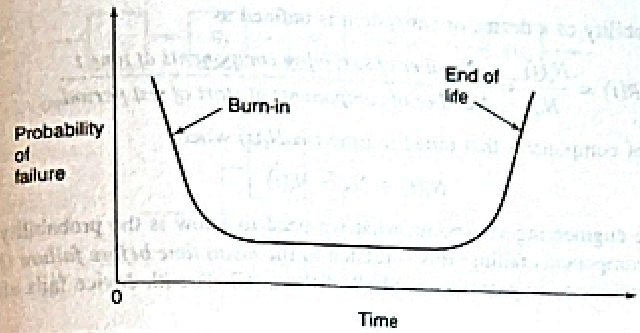

FIGURE 3.20 Bathtub curve for probability of failure.

at acceptable construction costs. It should be noted, however, that the cost of a satellite is very high compared to other equipment with a comparable number of components: a large GEO satellite costs around $100 M to build, close to the cost of a Boeing 747 jet airliner. The cost is acceptable because of the high revenue-earning capability of the satellite.

The reliability of a component can be expressed in terms of the probability of failure after time t, $P_F(t)$. For most electronic equipment, probability of failure is higher at the beginning of life—the burn-in period—than at some later time. As the component ages, failure becomes more likely, leading to the *bathtub curve* shown in Figure 3.20.

Components for satellites are selected only after extensive testing. The aim of the testing is to determine reliability, causes of failure, and expected lifetime. The result is a plot similar to Figure 3.20. Testing is carried out under rigorous conditions, representing the worst operating conditions likely to be encountered in space, and may be designed to accelerate failure in order to shorten the testing duration needed to determine reliability. Units that are exposed to the vacuum of space are tested in a vacuum chamber, and components subjected to sunlight are tested under equivalent radiant heat conditions. The initial period of reduced reliability can be eliminated by a burn-in period before a component is installed in the satellite. Semiconductors and integrated circuits that are required to have high reliability are subjected to burn-in periods from 100 to 1000 h, often at a high temperature and excess voltage to induce failures in any suspect devices and to get beyond the initial low reliability part of the bathtub curve.

SIDEBAR

The bathtub curve is familiar to owners of automobiles. A new car may have defects when it is delivered, and errors in manufacturing may lead to components failing soon after purchase. This is one of the reasons that manufacturers offer warranties for the initial life of their products. Once these defects have been overcome, by repair or replacement, reliability improves for a number of years until mechanical parts start to wear and failures occur. In an automobile, preventive maintenance can be carried out to replace parts that are known to wear most quickly. For example, spark plugs and drive belts may be replaced every 40,000 miles. The skill in owning an automobile is to judge the time at which the vehicle is starting up the end-of-life portion of the bathtub curve. That is the optimum time to dispose of the vehicle, and the worst possible time to buy one. Preventive maintenance is not possible with a geostationary satellite, at present, so other strategies must be adopted.

The reliability of a device or subsystem is defined as

$$R(t) = \frac{N_s(t)}{N_0} = \frac{\textit{Number of surviving components at time } t}{\textit{Number of components at start of test period}} \tag{3.5}$$

The numbers of components that failed in time t is $N_f(t)$ where

$$N_f(t) = N_0 - N_s(t) \tag{3.6}$$

From the engineering viewpoint, what we need to know is the probability of any one of the N_0 components failing: this is related to the *mean time before failure* (MTBF). Suppose we continue testing devices until all of them fail. The ith device fails after time t_i where

$$\text{MTBF} = m = \frac{1}{N_0}\sum_{i=1}^{N_0} t_i \tag{3.7}$$

The *average* failure rate λ, is the reciprocal of the MTBF, m. If we assume that λ is a constant, then

$$\lambda = \frac{\text{Number of failures in a given time}}{\text{Number of surviving components}}$$

$$\lambda = \frac{1}{N_s}\frac{\Delta N_f}{\Delta t} = \frac{1}{N_s}\frac{dN_f}{dt} = 1/\text{MTBF} \tag{3.8}$$

Failure rate λ is often given as the average failure rate per 10^9 h. The rate of failure, dN_f/dt, is the negative of the rate of survival dN_s/dt, so we can redefine λ as

$$\lambda = -\frac{1}{N_s}\frac{dN_s}{dt} \tag{3.9}$$

By definition from Eq. (3.5), the reliability R is N_s/N_0, so

$$\lambda = \frac{-1}{N_0 R}\frac{d}{dt}(N_0 R) = \frac{-1}{R}\frac{dR}{dt} \tag{3.10}$$

A solution of Eq. (3.10) is

$$R = e^{-\lambda t} \tag{3.11}$$

Thus the reliability of a device decreases exponentially with time, with zero reliability after infinite time, that is, certain failure. However, end of useful life is usually taken to be the time t_1, at which R falls to 0.37(1/e), which is when

$$t_1 = 1/\lambda = m \tag{3.12}$$

The probability of a device failing, therefore, has an exponential relationship to the MTBF and is represented by the right-hand end of the bathtub curve.

Redundancy

The equations in the preceding section allow us to calculate the reliability of a given device when we know its MTBF. In a satellite, many devices are used, each with a different MTBF, and failure of one device may cause catastrophic failure of a complete subsystem. If we incorporate redundant devices, the subsystem can continue to function correctly. We can define three different situations for which we want to compute subsystem reliability:

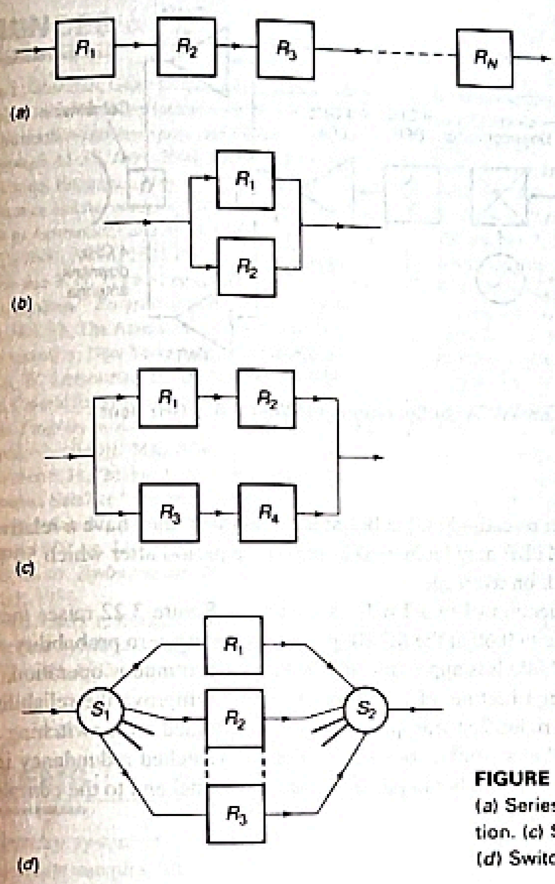

FIGURE 3.21 Redundancy connections. (*a*) Series connection. (*b*) Parallel connection. (*c*) Series/parallel connection. (*d*) Switched connection.

series connection, used in solar cells arrays, parallel connection, used to provide redundancy of the high power amplifiers in satellite transponders, and a switched connection, often used to provide parallel paths with multiple transponders. These are illustrated in Figure 3.21; also shown is a hybrid arrangement, a series/parallel connection, widely used in electronic equipment.

The switched connection arrangement shown in Figure 3.21*d* is also referred to as *ring redundancy* since any component can be switched in for any other. Switches S_1 and S_2 are a little more complicated than as shown, affording the choice of multiple paths in an M for N ring redundancy configuration. The important point to note is that the active devices ($R_1, R_2, \ldots, R_n$) have sufficient bandwidth, power output range, etc., to be able to handle any of the channels that might be switched through to them. Most TWTAs and SSPAs are such wideband, large power range devices.

An example of parallel redundancy for the HPA of a 6/4 GHz bent pipe transponder is shown in Figure 3.22. The transponder translates incoming signals in the 6-GHz uplink band by 2225 MHz and retransmits them in the 4-GHz band. The high-power output stage of the transponder has two parallel TWT amplifiers. One TWTA will be switched off, but must present a matched load when both off and on. If one TWTA fails, the other is switched on either automatically, or by command from earth. The TWT is a thermionic device with a heated cathode and a high voltage power supply. In common with other

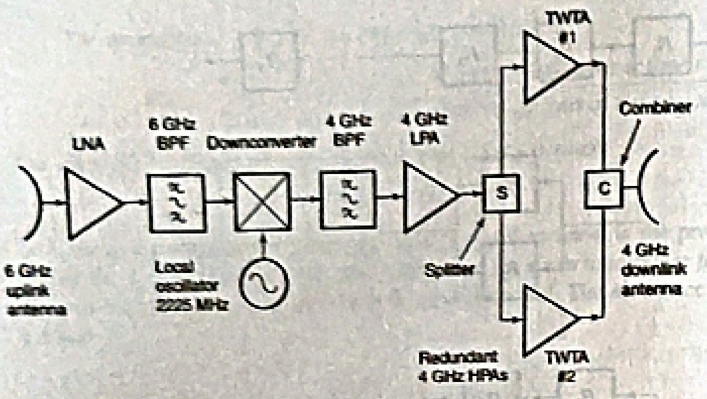

FIGURE 3.22 Redundant TWTA configuration in HPA of a 6/4 GHz bent pipe transponder.

thermionic devices such as cathode ray tubes and magnetrons, they have a relatively short MTBF. Although the MTBF may be 50,000 h, this is the period after which 50% of such devices will have failed, on average.

The parallel connection of two TWTs, as shown in Figure 3.22 raises the reliability of the amplifier stage to 0.60 at the MTBF period, assuming zero probability of a short circuit. A-lifetime of 50,000 h is approximately 6 years of continuous operation, which is close to the typical design lifetime of a satellite. To further improve the reliability of the transponders, a second redundant transponder may be provided with switching between the two systems. Note that a combination of parallel and switched redundancy is used to combat failures that are catastrophic to one transponder channel and to the complete communication system.

3.8 SUMMARY

Satellites that carry communications relays must provide a stable platform in orbit. Large GEO satellites have payload design lives that exceed 10 years and sufficient fuel to provide a maneuvering lifetime that typically exceeds 15 years. The satellite must carry a number of subsystems to support its communications mission. The attitude and orbital control system keeps the satellite in the correct orbit and on station, and pointing in the correct direction. The telemetry, tracking, command, and monitoring system allows an earth station to control the subsystems in the satellite and to monitor their health. The power system provides the electrical energy needed to run the satellite (housekeeping) and the communications system. Solar cells generate the electrical power, a power conditioning unit controls its distribution, and batteries provide power during launch and eclipses.

Satellites often employ frequency reuse, either by using the same frequencies again in spatially separated beams or by using the same frequencies in orthogonal polarizations within the same beam. Sometimes, both reuse techniques are used simultaneously. Frequency reuse allows the same RF spectrum to be used more than once to increase the satellite's capacity.

Antennas are a limiting factor in all radio communication systems. Very complex antennas have been developed for satellites to provide multiple beams and orthogonal polarizations from a single antenna. Reflector antennas with clustered feeds and phased array antennas are used to generate shaped and multiple beams. Reliability is an important issue in satellites. Redundancy can be used to provide additional receivers and high-power amplifiers that can take over when a unit fails.

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PROBLEMS

1. The telemetry system of a geostationary communications satellite samples 100 sensors on the spacecraft in sequence. Each sample is transmitted to earth as an 8-bit word in a TDM frame. An additional 200 bits are added to the frame for synchronization and status information. The data are then transmitted at a rate of 1 kilobit per second using BPSK modulation of a low-power carrier.

a. How long does it take to send a complete set of samples to earth from the satellite?

b. Including the propagation delay, what is the longest time the earth station operator must wait between a change in a parameter occurring at the spacecraft and the new value of that parameter being received via the telemetry link? (Assume a path length of 40,000 km.)

2. A spinner satellite has solar cells wrapped round a cylindrical drum 3.00 m in diameter, with a height of 5.0 m on station. The drum is rotated at 60 rpm to spin-stabilize the satellite. At the end of life, the solar cells are required to deliver 4.0 kW of electrical power.

a. Calculate the efficiency of the solar cells at end of life. Assume an incident solar power of 1.39 kW/m^2, and that the effective solar radiation absorbing area of the solar cells is equal to the cross-sectional area of the drum.

b. If the solar cells degrade by 15% over the lifetime of the satellite, so that the end-of-life output power is 85% of the beginning-of-life output power, what is the output of the solar cells immediately after launch?

c. If the drum covered in solar cells of the spinner design had been replaced by solar sails that rotated to face the sun at all times, what area of solar sails would have been needed? Assume that cells on solar sails generate only 90% of the power of cells on a spinner due to their higher operating temperature.

3. A Direct Broadcast Television (DBS-TV) satellite is in geostationary orbit. The electrical power required to operate the satellite and its transmitters is 4 kW.

Two designs of satellite can be used: three axis stabilized with solar cells and a spinner.

a. A three-axis stabilized satellite has two solar sails of equal area that rotate to face the sun at all times. The efficiency of the solar cells at end of life is predicted to be 15%.