Satellite Links

Satellite links are used to provide communications over very large distances (global coverage). This is achieved by using the satellite as a repeater. A ground station relays a signal up to the satellite at a frequency known as the *uplink frequency*. The satellite receives this signal and re-broadcasts it on a *downlink frequency* to another ground station. If digital communications signalling is used, the signal may be *regenerated* before it is re-transmitted to Earth.

The analysis and design of satellite links is not unlike other links we have already studied in the course. There are particular details in evaluating certain components of the link budget that deserve particular attention: the characteristics of the satellite transponder, the fact that two frequencies (and hence two link budgets) are used, and the need to factor in the orbital geometry of the satellite to compute the free space loss.

1 Transponder Characteristics

A satellite receives a radio signal on its uplink receiver. It then downconverts this signal to the downlink frequency, and amplifies it before sending the downlink signal to the transmit antenna. These functions are carried out by the *satellite transponder*. Ideally, aside from the frequency shift, you can imagine this process as a 2-hop link (discussed earlier in the course) with some amplification between the links. At the satellite, apart from noise considerations, you would expect the downlink signal to be G dB higher than the received uplink signal, where G is the transponder gain.

In practise, it is not quite that simple. The transponder amplification stage is realized from a solid-state amplifier or travelling-wave tube (TWT) which can only supply a maximum power level due to device constraints. Hence, the transponder only behaves as a linear device over a range of input power levels. This is demonstrated graphically in Figure 1. The power of the uplink signal is shown on the x-axis. In this example, the transponder has a nominal gain of 30 dB, so in theory, the output signal power should be increased by 30 dB. We can see that over the majority of the characteristic, this is true. However, at a certain point, the output becomes *saturated* as the amplifier is no longer capable of delivering additional power (it is essentially clipping the input signal). In this example, the saturated output power is 35 dBW (about 3.2 kW). Well below this point, the curve departs from the ideal linear characteristic (shown as a dotted line) and the amplifier is said to be in *compression* or *saturation*.

Operating beyond the compression point of the amplifier causes additional problems. The amplifier is no longer linear. A nonlinear device is capable of generating new frequencies from the frequencies present in the input signal; in this case, the input communication signal will be distorted to produce *intermodulation products* which extend beyond the bandwidth of the input signal. This can create interference problems: if you imagine a single channel in the input signal as occupying a specific bandwidth, then the spillover will produce unwanted signals in the adjacent channels, or interference. This is essentially noise, so in effect we are increasing the noise in the re-broadcast signal when the amplifier is operated in compression.

Therefore, it is important to *limit* the input power to the transponder, which equates to managing

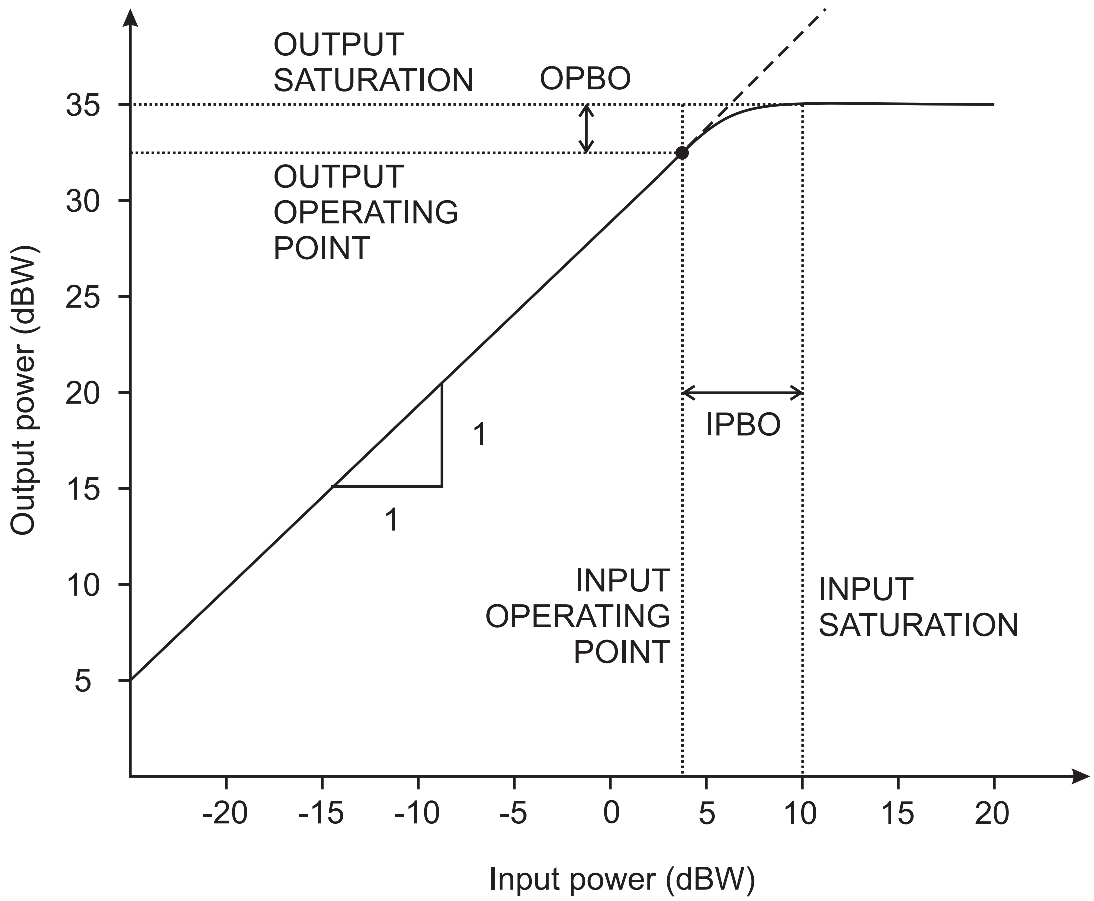

Figure 1: Satellite transponder input/output power characteristic

the EIRP of the ground station carefully. The goal is to obtain as much output power available from the satellite while minimizing the signal distortion. This usually means reducing or *backing off* the input power, so that the amplifier is operated just below compression. This point gives decent output power while producing manageable levels of distortion/interference. The saturated input power density may be specified for a satellite, as opposed to the saturated input power.

In this example, the output begins to compress at an input power of about 5 dBW. Increasing the input power beyond this point does not yield a linear increase in output power: for example, at an input power of 10 dBW, we would expect 40 dBW of output power and yet only are able to achieve a power of 35 dBW. The output is fully saturated at this point.

A suitable input power level could be right before the saturated region is entered, along with some margin. If the input power level was chosen to be 4 dBW, we would say the the input power back-off (IPBO) is 6 dB, since the power has been reduced 6 dB from the saturated power point. Correspondingly there is an output power back-off (OPBO) which in this case is 2.5 dB; that is, the output is 2.5 dB from being fully saturated.

A common empirical formula that is used to model the input/output characteristic described above is

$$\frac{W_{\text{out}}}{W_{\text{sat}}} = \frac{4P_{\text{in}}/P_{\text{sat}}}{[1+(P_{\text{in}}/P_{\text{sat}})]^2}, \tag{1}$$

where W_{sat} is the saturated output power from the transponder, W_{out} is the actual power power produced at the output of the transponder, P_{in} is the input power density to the transponder's receiving antenna, and P_{sat} is the power density at the transponders' receiving antenna that is known to saturate the output of the transponder.

2 Link Budget

As discussed, the link budgets on the uplink and downlink are treated separately, since different transmission distances may be involved, and there is a significant different in frequency/wavelength. On a per-Hz basis, the link budget equations are:

$$\left[\frac{C}{N_0}\right]_U = EIRP_G - FSL_U - L_U + \frac{G_S}{T_S} + 228.6 \qquad \text{(units: dB-Hz)} \tag{2}$$

$$\left[\frac{C}{N_0}\right]_D = EIRP_S - FSL_D - L_D + \frac{G_G}{T_G} + 228.6 \qquad \text{(units: dB-Hz)} \tag{3}$$

where the subscripts U and D refer to the uplink and downlink, respectively, and the subscripts S and G refer to the satellite and ground station respectively. Note that the EIRP of the ground station is selected so that the transponder is suitably backed-off from saturation. The terms L_U and L_D refer to other losses on the uplink and downlink, such as atmospheric absorption, rain attenuation, etc.

Free space loss forms the bulk of the loss in the link budget. This is especially true for satellite systems due to the huge distances involved which are typically tens of thousands of kilometres from station to station. Hence, it is important to understand the orbit geometry of the satellite so that the link distance and hence free space loss can be estimated accurately. The calculation of the link distance requires an understanding of orbital mechanics, which is discussed in a separate note.

Knowing the link distance the free-space loss can be calculated on the link being considered (uplink/downlink). The link losses on the uplink and downlink are not the same, even if the distance r is the same, because the uplink and downlink frequencies are not the same. In fact, usually the downlink is at lower frequencies. This is done for multiple reasons:

- Atmospheric attenuation is less at lower frequencies.
- The ground station is not power-limited like the satellite, and hence is better equipped to deal with attenuation on the uplink. The maximum EIRP of the ground station is ultimately limited by the satellite's transponder (see the next section)
- The G/T on the uplink is very poor, since the satellite "sees" a warm earth. Hence, there is little point in using a lower frequency with less attenuation since the SNR is going to be relatively poor, anyway.
- Improvements are thus best reserved for the downlink.

3 Satellite Orbits

As discussed in a separate note on orbital mechanics, the time T it takes the satellite to transit through one orbit is determined knowing the gravitational force produced by the Earth, and the distance the satellite is from the centre of that mass a. The orbital period is equal to

$$T = \frac{2\pi r^{3/2}}{\sqrt{\mu}} \tag{4}$$

where $\mu = m_E G$ is the product of the Earth's mass m_e and the universal gravitation constant G. This results a constant $\mu = 3.986 \times 10^{14}$ N·m^2/kg which is known as *Kepler's constant*. Knowing the orbital altitude of the satellite H, we can determine $a = R_e + H$ and find T. The orbital period for various different orbits and some well-known satellites are summarized in Table 1.

Orbit	Orbital Altitude	Orbital Period	Examples
Low earth orbit (LEO	160 – 2,000 km	87 – 127 min	International Space Station (ISS), Hubble Space Telescope, amateur radio satellites, Iridium
Medium earth orbit (MEO)	2000 - 35,786 km	127 min – 24 hr	New-ICO, GPS, GLONASS, Telstar
Geostationary earth orbit (GEO)	35,786	23 h 56 m 4.1 s	Intelsat, GOES, DirecTV satellites, Anik, EutelSAT

Table 1: Satellite orbits

Point of interest: GEO orbit Geostationary orbit is a special case of a satellite orbit meeting the following conditions:

1. The orbit is perfectly circular, with an eccentricity of zero unlike LEO, MEO, and other orbits.
2. The altitude of the satellite is such that the orbital period of the satellite is 23 hours, 56 minutes, and 4.1 seconds, which is one *sidereal day* which is the time it takes for the Earth to rotate about its axis exactly once.

3. The satellite's orbital plane must coincide with the the plane of the equator.

Under these conditions, the satellite appears perfectly stationary with respect to an stationary station on the Earth's surface. This makes this orbit highly coveted for telecommunications and broadcast applications, such as television. The geostationary orbital altitude is, from Table 1, $H = 35,786$ km. It follows knowing that $R_e = 6,378.2$ km that the ratio $\frac{R_e}{R_e+H} = 0.151 = 1/6.61$. Another way of remembering this is that the geosynchronous orbit altitude is approximately 6.6 Earth radii.

4 Orbital Geometry

Satellites generally orbit the Earth along an elliptical path and this ellipse is actually inclined relative to the equatorial plane of the Earth. Satellites also orbit the Earth at different speeds, since the orbital periods depend on the orbital altitude of the satellite. Generally, to locate a satellite at an instant in time requires knowledge of the geometry of the ellipse relative the the Earth as well as times that the satellite passes reference points along the ellipse. The parameters are known as the *Keplerian elements* of the satellite and are precisely known by satellite operators. The relation of these elements to the elliptical geometry we have described here is beyond the scope of this document, but it suffices to say that in general the distance from a ground station to the satellite depends on time, and that the satellite may not always be visible (above the horizon).

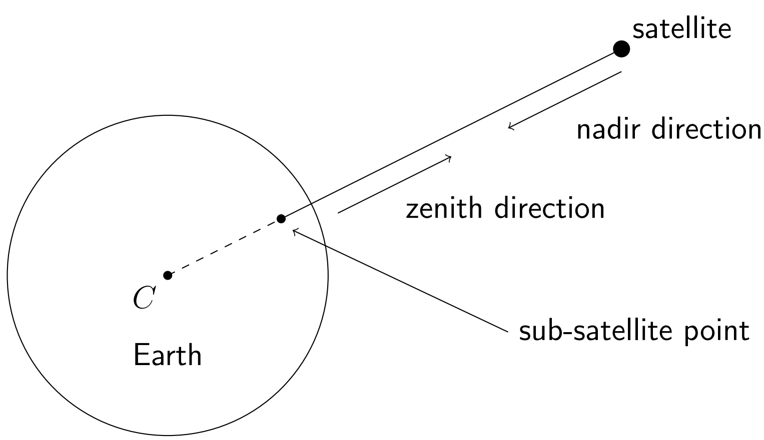

Figure 2: The sub-satellite point

From the perspective of a ground station, what matters is the location of the *sub-satellite point* on the Earth's surface which is a point on the ground in the nadir (downward) pointing direction of the satellite. This point is shown in Figure 2. This point can be described by its latitude and longitude. The Earth station also can be located by its own latitude and longitude. Combined, these parameters are:

- L_e, the Earth station latitude (the number of degrees the station is north of the equator);
- ℓ_e, the Earth station longitude (the number of degrees the station is west of the prime meridian);

- L_s, the latitude of the sub-satellite point; and
- ℓ_s, the longitude of the sub-satellite point.

Consider the link between a ground station and orbiting satellite, shown in Figure 3.

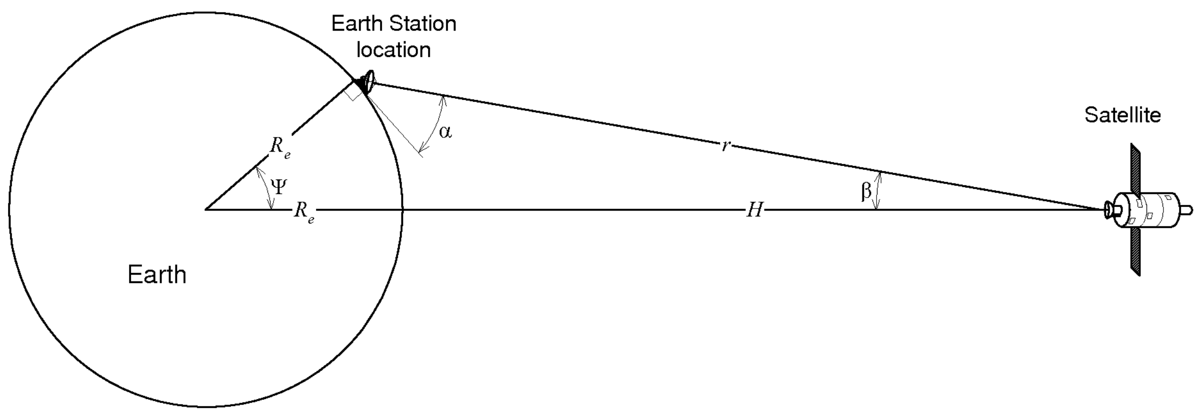

Figure 3: Orbit geometry

The angle ψ is related to the coordinates of the Earth station and the sub-satellite point according to

$$\cos\psi = \cos L_e \cos L_s \cos(\ell_s - \ell_e) + \sin L_e \sin L_s. \tag{5}$$

The remaining angles shown in the diagram are easily related by applying the law of sines to the triangle formed by the orbit geometry. Hence,

$$\frac{R_e}{\sin\beta} = \frac{R_e + H}{\sin(\alpha + 90^\circ)} = \frac{r}{\sin\psi}. \tag{6}$$

Since $\sin(\alpha + 90^\circ) = \cos\alpha$,

$$\alpha = \cos^{-1}\left(\frac{R_e + H}{r}\sin\psi\right). \tag{7}$$

α is the elevation angle (angle above the horizon) that the Earth station antenna needs to be pointed to to make contact with the satellite. Therefore, knowing the coordinates of the Earth station and the sub-satellite point, the appropriate elevation angle can be found. An additional angle, the azimuth angle, is also needed, but is not discussed here.

Knowing the angle ψ we can find the total distance r the satellite is from the ground station. Using the law of cosines,

$$r^2 = (R_e + H)^2 + R_e^2 - 2R_e(R_e + H)\cos\psi, \tag{8}$$

or

$$r = (R_e + H)\left[1 + \left(\frac{R_e}{R_e + H}\right)^2 - 2\frac{R_e}{R_e + H}\cos\psi\right]^{1/2}. \tag{9}$$

The elevation angle can then be found knowing $\sin(\alpha + 90^\circ) = \cos\alpha$, so

$$\cos\alpha = \frac{(R_e + H)\sin\psi}{r} = \frac{\sin\psi}{\left[1 + \left(\frac{R_e}{R_e+H}\right)^2 - 2\frac{R_e}{R_e+H}\cos\psi\right]^{1/2}}. \tag{10}$$

If instead, we wish to determine ψ for a given elevation angle α in advance, we can use the fact that $\beta = 90^\circ - (\alpha + \psi)$, $\sin\beta = \cos(\alpha + \psi)$ and

$$\psi = \cos^{-1}\left(\frac{R_e}{R_e + H}\cos\alpha\right) - \alpha. \tag{11}$$

The satellite is only visible from an earth station is the elevation angle α to be above some minimum value, which is at least 0°. From Figure 3, this requires

$$R_e + H \geq \frac{R_e}{\cos\psi} \tag{12}$$

or

$$\psi \leq \cos^{-1}\frac{R_e}{R_e + H}. \tag{13}$$

For a nominal GEO orbit, $\psi \leq 81.3^\circ$ for the satellite to be visible.