Space Engineering 2 Summary

1. Space Structures

1.1 Structural Types and Loads

Space structures are present to support and protect the payload. First there is the **primary structure**, which is the backbone of the structure. There are also the **secondary structures**, supporting instruments and such. If the structure fails, than the mission fails, so structures need to be **reliable**. They should give **protection** against the harsh space environment. They should be **strong** enough to withstand the loads and **stiff** enough to prevent unwanted vibrations.

Now let's look at the loads. We can distinguish five groups.

- The **static loads**, which are more or less constant in time.
- The **low-frequency vibrations**. These are usually sinusoidal loads. Its frequency ranges from 5-100Hz.
- The **high-frequency vibrations**. These are random vibrations, with a frequency ranging from 20-1000Hz. Its power is proportional to a^2, where a is the acceleration of the spacecraft.
- The **acoustic loads** are induced by noise, usually coming from the engines. Its frequency ranges from 20-10000Hz. Especially secondary structures are sensitive to these loads.
- **Shock loads** are high loads only present for a very short time.

1.2 Stiffness and Frequencies

Every structure can vibrate at its own **eigenfrequency** (also called **natural frequency**). **Resonance** occurs when something else is also vibrating at this eigenfrequency. This is very dangerous and should be prevented.

To find the natural frequency, we first would like to know the stiffness k. This is defined as

$$k = \frac{P}{\delta}, \tag{1.2.1}$$

where P is an applied load and δ is the resulting displacement. So for a straight bar, the stiffness in axial and lateral directions are

$$k = \frac{EA}{L} \text{ axial}, \qquad k = \frac{3EI}{L^3} \text{ lateral}. \tag{1.2.2}$$

The natural frequency can now be found using

$$f = \frac{1}{T} = \frac{1}{2\pi}\sqrt{\frac{k}{m}}, \tag{1.2.3}$$

where T is the **vibration period** and m is the mass of the object. So we find that

$$f = \frac{1}{2\pi}\sqrt{\frac{EA}{mL}} \text{ axial}, \qquad k = \frac{1}{2\pi}\sqrt{\frac{3EI}{mL^3}} \text{ lateral}. \tag{1.2.4}$$

Designers can use these equations. It's their task to make sure that the frequency f is bigger than the minimum frequency f_{min}.

1.3 Margin of Safety

To see how well a structure is designed, the **margin of safety** can be used. It is defined as

$$MS = \frac{\sigma_{allowable}}{j\, \sigma_{design}} - 1, \tag{1.3.1}$$

where j is the safety factor. If $MS < 0$, then failure will occur. If $MS > 1.5$, then the structure is inefficient. So the margin needs to be between these values.

2. Thermal Control

2.1 Emission

Some parts of the spacecraft can only operate in a certain temperature range. So **thermal control** is needed. Its goal is to keep the temperatures with the allowable ranges. It should achieve this by exchanging heat with space, and dividing the heat within the structure.

For internal heat exchange, conduction is an important mechanism. However, to exchange heat with space, radiation should be considered. The **radiation power** E that every square meter of a spacecraft emits can be found by

$$E_{emission} = \varepsilon \sigma T^4, \tag{2.1.1}$$

where ε is the **emissivity** and $\sigma = 5.67 \cdot 10^{-8} W/m^2 K^4$ is the **Stephan Boltzmann's constant**.

2.2 Absorption

You of course don't only emit radiation. You also recieve it. Three things can happen with radiation. A part α (the **absorptivity** is being absorbed. Another part ρ (the **reflectivity**) is being reflected. The final part τ (the **transmissivity**) simply passes through the object. Naturally is must be true that

$$\alpha + \rho + \tau = 1. \tag{2.2.1}$$

The radiation power that is received from the sun, if you're near earth (1AU from the sun) is $\Phi = 1371 W/m^2$. Radiation from other plants/stars is usually negligible, although sometimes the **albedo** (sunlight reflected by the earth) should be taken into account.

If we don't consider the albedo, then the radiation that every square meter of a spacecraft will absorb will be

$$E_{absorbtion} = \alpha \Phi. \tag{2.2.2}$$

Because of this radiation, the temperature in the spacecraft will change. It will finally come to a stable point when $E_{emission} = E_{absorption}$, or equivalently,

$$\alpha A_s \Phi = \varepsilon A_{ext} \sigma T^4, \tag{2.2.3}$$

where A_s is the radiated surface area of the spacecraft and A_{ext} is the external surface area. From this **heat balance equation** we can find the stable temperature of the spacecraft, at which **thermal equilibrium** occurs.

There is a relation between the emissivity and the absorptivity. For a particular wavelength λ the absorptivity and emissivity are equal, so $\alpha_\lambda = \epsilon_\lambda$. This then also holds for the entire spectrum, so $\alpha = \epsilon$. However, keep in mind that this is only true for the same spectrum. So for the infrared spectrum, we can indeed say that $\alpha_{ir} = \epsilon_{ir}$. However, the sun emits a totally different spectrum, with thus a different α_s.

2.3 Heat Exchange

What happens when we're having heat exchange between two objects? Suppose we have two opposite plates i and j. If we use radiation to exchange heat, then the **heat flow** is found using

$$Q_{ij} = R_{ij} \left(\sigma T_i^4 - T_j^4 \right) = \frac{\varepsilon}{2 - \varepsilon} A \left(\sigma T_i^4 - T_j^4 \right), \tag{2.3.1}$$

where R_{ij} is the **radiative coupling**. Here we have assumed that $\alpha = \varepsilon$.

We can do the same for conduction. However, now the heat flow is

$$Q_{ij} = C_{ij} \left(T_i - T_j\right), \tag{2.3.2}$$

where C_{ij} is the **conductive coupling**. For simple beams with length L, cross-sectional area A and conductivity k, the conductive coupling is $C_{ij} = \frac{kA}{L}$. However, for two parts with interfacing contact area a and **contact conductance** h_c, we find that $C_{ij} = h_c A$.

When multiple conductors are stacked in a series, than $1/C_{total} = 1/C_1 + 1/C_2 + \ldots$. However, when they are stacked in parallel, then $C_{total} = C_1 + C_2 + \ldots$.

If thermal equilibrium is present, than $Q_{out} = Q_{in}$. If not, then the change in temperature over time can be found, using

$$Q_{in} - Q_{out} = \Delta Q = m c_p \frac{dT}{dt}, \tag{2.3.3}$$

where c_p is the **specific heat capacity** of the material. If necessary, Q_{in} can be increased by bringing along heat sources on the spacecraft.

2.4 Thermal Control Systems

Thermal control systems can be divided into two categories. Some spacecraft have a **passive** thermal control system. It consumes no power, it's simple and usually low-weight. An **active** system, however, offers more flexibility and has greater capacities.

To keep heat inside the spacecraft, **insulation** is important. Often **multi-layer insulation** (MLI) is used. This is a package of radiating screens. Since the amount of radiation that every layer lets through is small, the amount of radiation eventually getting out the spacecraft is very small. If the layers are also highly reflective, than few heat will enter the spacecraft as well.

Sometimes a spacecraft, however, wants to get rid of its heat. For this, **radiators** can be used. These are parts that need to send out as much radiation as possible. So their emissivity should be high. They shouldn't absorb much radiation though, so in particular the coefficient ϵ/α should be as high as possible.

3. Attitude Determination & Control

3.1 Attitude Data

The know the **attitude** and **motion** of a spacecraft, we need to know its **position**, **velocity**, **orientation** and **rotational velocity**. The first two concern translations, while the latter two concern rotations.

Let's consider an orbiting satellite. The origin lies at the COM of the spacecraft. The x-axis points in the direction of the **velocity vector**. The z-axis points to the center of the earth (the **nadir vector**). Finally the y-axis is perpendicular to the past two, according to the right-hand rule (the **orbit normal vector**). Rotation about the x-axis is called **roll**, rotation about the y-axis is called **pitch** and rotation about the z-axis is called **yaw**.

3.2 Attitude Dynamics and Kinematics

The attitude of a spacecraft can be calculated using dynamics and kinematic equations. To do this, it is usually assumed that the spacecraft is a **rigid body**. It can now be derived that

$$I_{xx}\dot{\omega}_x + (I_{zz} - I_{yy})\,\omega_y\omega_z = M_{c_x} + M_{d_x}, \tag{3.2.1}$$

$$I_{yy}\dot{\omega}_y + (I_{xx} - I_{zz})\,\omega_z\omega_x = M_{c_y} + M_{d_y}, \tag{3.2.2}$$

$$I_{zz}\dot{\omega}_z + (I_{yy} - I_{xx})\,\omega_x\omega_y = M_{c_z} + M_{d_z}, \tag{3.2.3}$$

where ω is the rotational velocity of the spacecraft, I is the moment of inertia, M_c is the control torque and M_d is the disturbance torque, about the corresponding axes.

Let's define ϕ as the **roll angle** (the angle over which roll has taken place), θ as the **pitch angle** and ψ as the **yaw angle**. Now ϕ, θ and ψ are the **attitude angles**. In fact, the orientation of the spacecraft is given by the vector containing these angles. It can now also be shown that

$$\begin{bmatrix}\dot{\phi}\\ \dot{\theta}\\ \dot{\psi}\end{bmatrix} = \begin{bmatrix}\omega_x\\ \omega_y\\ \omega_z\end{bmatrix} + \omega_0\begin{bmatrix}\psi\\ 1\\ -\phi\end{bmatrix}, \tag{3.2.4}$$

where ω_0 is the angular velocity of the orbit. Note that the second part is necessary, because the satellite is rotating around Earth. Due to this, the z-axis continuously changes, as it is defined to point to the center of the Earth.

3.3 Disturbance Torques

Space isn't a perfect environment. There are many types of disturbances. First there are **aerodynamic disturbances**. This decreases as you are further away from earth (according to $e^{-\alpha r}$, where r is the distance from Earth). Earth's **magnetic field** can also cause disturbances. These also decrease as you are further away from earth (now according to r^{-3}). The same goes for **gravity gradients**, which are changes in Earth's gravitational field. The solar also influences the spacecraft attitude. **Solar radiation** can cause disturbances. On some places in the solar system also micrometeorites can play a role. And sometimes internal forces of the spacecraft can also causes disturbances.

How do we cope with these disturbances? One way to do this, is by using a **control wheel**. Let's suppose we have a spacecraft with rotational velocity Ω_v with respect to a reference axis. In that spacecraft is a control wheel, rotating with a velocity Ω_w with respect to the spacecraft. The wheel and the vehicle have moment of inertias of I_w and I_v and angular momentums of

$$H_w = I_w \left(\Omega_w + \Omega_V\right), \qquad \text{and} \qquad H_v = I_v \Omega_v. \tag{3.3.1}$$

If a disturbance torque M is acting on the spacecraft, an angular acceleration will take place. This happens according to

$$M = \frac{d}{dt}\left(H_w + H_v\right) = I_w \left(\dot{\Omega}_w + \dot{\Omega}_v\right) + I_v \dot{\Omega}_v = I_w \dot{\Omega}_w + \left(I_w + I_v\right) \dot{\Omega}_v. \tag{3.3.2}$$

If the angular acceleration of the spacecraft needs to be zero, then we find that $M = I_w \dot{\Omega}_w$. All the torque is then applied to the control wheel.

There are two types of disturbance torques that can be considered. There are **cyclic torques** which oscillate over an orbit. Since the torque is alternatingly positive and negative, the control wheel only needs to store the torque for a while. However, there are also **secular torques**. These torques build up angular momentum over time. So a control wheel eventually won't be enough. There should be some way to dump angular momentum (often by using thrusters).

3.4 Passive Control Systems

Let's take a look at the ways in which we can control the attitude of an spacecraft. In **spin stabilisation** the spacecraft is spinning about one axis. Due to gyroscopic stabilisation it will hardly rotate about other axes. Let's, for example, suppose it's spinning about its x-axis and a disturbance torque M_{d_y} appears. Because ω_x is big, ω_z will only get a small value (remember the equations of motion). Also it is the rotational velocity ω_z that will be nonzero (not the rotational acceleration $\dot{\omega}_y$). This means that the rotation caused by the disturbance torque will stop when the disturbance torque has disappeared. So we can conclude that the effect of the disturbance torque is limited.

In **dual spin stabilisation** only part of the spacecraft rotates. A spacecraft with a control wheel is an example of this. While the spacecraft has gyroscopic stabilisation, it is still possible to have a non-rotating platform on which equipment can be placed.

Some stabilisation methods use the environment. In **gravity gradient stabilisation** the spacecraft makes use of the fact that the gravity field of a planet decreases with increasing distance from that planet. The moment caused is given by

$$\mathbf{M_g} = \begin{bmatrix} M_{g_x} \\ M_{g_y} \\ M_{g_z} \end{bmatrix} = 3\omega_0^2 \begin{bmatrix} \left(I_{zz} - I_{yy}\right)\phi \\ \left(I_{zz} - I_{yy}\right)\theta \\ 0 \end{bmatrix}. \tag{3.4.1}$$

Let's now define the coefficients

$$k_x = \frac{I_{yy} - I_{zz}}{I_{xx}}, \qquad k_z = \frac{I_{yy} - I_{xx}}{I_{zz}}. \tag{3.4.2}$$

To have a gravitationally stable spacecraft, it must satisfy to

$$3k_x + k_x k_z + 1 > 0, \qquad 3k_x + k_x k_z + 1 > 4\sqrt{k_x k_z} \qquad \text{and} \qquad k_x k_z > 0. \tag{3.4.3}$$

If a spacecraft satisfies this condition, it doesn't automatically imply that it is gravitationally stable though.

Magnetic stabilisation is slightly similar to gravitational stabilisation. However, now the spacecraft uses the magnetic field of the earth to orient itself. In **aerodynamic stabilisation** aerodynamic forces are used to control the spacecraft orientation.

3.5 Active Control Systems

The previously discussed methods of stabilisation are all **passive** methods. Once a spacecraft is in space, it has no further control on its attitude. An example of an **active** stabilisation method is **three axis active attitude control**. Here the attitude of the spacecraft is controlled about all three axis. This can be done with for example reaction wheels and thrusters.

Let's once more consider an spacecraft with a control wheel. Let's suppose the control wheel is exerting a torque of M_c on the spacecraft. The spacecraft has control over this M_c. We can set

$$M_c = -K_p\Theta - K_d\dot{\Theta}, \tag{3.5.1}$$

where K_p and K_d are positive constants. Also Θ is the angle of the spacecraft with respect to some reference point (so $\dot{\Theta} = \Omega_v$). We now have programmed a **PD controller**. Here the P stands for the Proportional-term $K_p\Theta$ and the D stands for the Differentiation-term $K_d\dot{\Theta}$. In a **PID controller** also an Integration-term is included, which removes steady state errors due to disturbances.

Also thrusters can be used to provide three axis stabilisation. When thrusters are applied, it is important to consider the **moment arm** of the thruster with respect to the COM of the spacecraft.

3.6 Sensors

It's nice to be able to rotate. But you first need to know your orientation, before you can decide to change it. Therefore sensors are used. There are many types of sensors.

Some types use external reference points. **Sun sensors** use the sun as a reference point. **Earth sensors** use the Earth. Both are relatively inaccurate. **Star sensors** are more accurate and also aren't troubled by eclipses. However, they are heavier and use more power.

Gyroscopes do not use external reference points. Although they are initially very accurate, their accuracy decreases as time passes. Combining gyroscopes with other sensor systems could solve this problem. Now the gyroscope can be calibrated regularly.

Magnetometers measure Earth's magnetic field and calculate the attitude from the acquired data. Finally, if the satellite is close to the ground (low Earth orbits), **GPS** can be used to determine position.

4. Electrical Power Subsystem

4.1 Power Introduction

The **electrical power subsystem** (EPS) often makes up 20-40% of the spacecraft mass and is therefore important. When designing the subsystem, attention should be given to a lot of parameters, like **power level**, **current type** (alternating current (AC) or direct current (AC)), backup and so on. Of course also general parameters like cost, weight, reliability and such are important.

When designing an EPS, first a **power budget** should be made. This is a list of all electrical apparatus on board, with for every part the power it needs in every operating mode.

An EPS usually consists of four parts. The **power source** provides power. The **power storage** stores it (normally used when the power provided/needed is not constant over time). The **power management** controls and converts power, if necessary. Finally the **power distribution** brings the power to the on-board equipment.

There are many types of energy sources. The **sun** can provide power due to radiation. **Nuclear** energy comes from the decay of atoms. Finally **chemical** energy comes from a chemical reaction between certain elements. Energy sources can also be **external** (like the sun) or **internal** (like chemical energy). The advantage of external power sources, is that they don't contribute to the mass of the spacecraft.

4.2 Solar Power

Let's go into a bit more detail on solar power. When the sun is used as power source, often **solar arrays** are used. A solar array is a **photo-voltaic** device. Solar arrays can be mounted on the spacecraft, or be deployable. The latter has as advantage that the cells can be directed more to the sun. If the solar array is not pointed to the sun, a correction factor needs to be taken account. The recieved solar power then isn't $\Phi = 1371 W/m^2$, but will become $\Phi_{actual} = \Phi \cos\theta$, where θ is the **incidence angle** of the solar array with respect to the sun.

When using solar cells, the **efficiency** η needs to be taken into account, defined as

$$\eta = \frac{P_{out}}{P_{in}}, \tag{4.2.1}$$

where P of course denotes the power. The **voltage** V and **current** I solar cells provide, depends on how they are connected. Placing cells in series increases the voltage, while placing cells in parallel increases the current. A series of cells is called a **string**. A series of strings is called a **section**.

When designing a solar array, **eclipses** need to be taken into account. In this time period, no energy is received form the sun. Let's suppose t_e, P_e and η_e are the eclipse time, power needed during eclipse and path efficiency during eclipse (going via the battery), respectively. t_d, P_d and η_d are the same for the day-time period. The power needed during day-time P_{sa} to operate the spacecraft and load the battery can now be found using

$$P_{sa} t_d = \frac{P_e t_e}{\eta_e} + \frac{P_d t_d}{\eta_d}. \tag{4.2.2}$$

When calculating the number of solar cells, the **degradation** should be taken into account. A solar panel won't provide the same amount of power over the years. Therefore there is a so-called **degradation factor** δ, giving the decrease in power production per year. The part of power left after a number of years x is then given by $(1 - \delta)^x$.

4.3 Batteries

A **battery** is a series of voltaic cells. They can be rechargeable (**secondary batteries**) or non-rechargeable (**primary batteries**). Non-rechargeable batteries only provide power for at most a day, so they are therefore almost only used in launchers.

A measure of how much power a battery can deliver, is the **capacity** C, given by

$$C = \frac{E}{V} = \frac{Pt}{V} = It, \tag{4.3.1}$$

where E is the energy provided. The required capacity of a battery that is used to cope with ellipses can be found using

$$C = \frac{P_e t_e}{V\,DOD\,\eta_t}, \tag{4.3.2}$$

where DOD is the **depth of discharge** (the part of the battery that can be discharged) and η_t is the **transmission efficiency**.

Just like in a solar cell, the battery cells can be put in series or parallel. The amount of cells that are put in series n depends on the cell power V_{cell} and the required battery power V_{bat}, according to

$$n = \frac{V_{bat}}{V_{cell}}. \tag{4.3.3}$$

The number of cell strings put in parallel m can then be found by using

$$m = \frac{C_{bat}}{C_{cell}}. \tag{4.3.4}$$

The total number of battery cells is then simply the product of n and m.

To find information about the battery mass and volume, we need the **specific energy** E_{sp} (energy per unit mass) and the **battery cell volume** E_δ (energy per unit volume). The mass and volume can then be found using

$$M = \frac{E}{E_{sp}}, \qquad V = \frac{E}{E_\delta}, \tag{4.3.5}$$

where V now denotes the volume.

Energy from the sun can also be used without solar arrays. By using **solar thermal-electric systems** the radiation is turned into heat, which is then turned into energy. There are two methods to do the latter part. In **static methods** there are no moving parts. Although the efficiency is low, the reliability is high. In **dynamic methods** there are moving parts, which can cause vibrations. Also leakage is a danger. Although the efficiency of dynamic methods is higher, the static methods are preferred.

4.4 Other Power Sources

Let's look at the other power sources. First we consider **fuel cells**. This is a chemical energy source. Fuel cells give a relatively high power, for a relatively short duration (about a month). The fuel cell mass consists of the dry mass M_{fc} and reactants M_r. If P_{sp} is the specific power, and C_r is the **reactant consumption rate**, then the total fuel cell mass is

$$M = M_{fc} + M_r = \frac{P}{P_{sp}} + EC_r. \tag{4.4.1}$$

In a **nuclear-electric source** first heat is generated. This heat is then converted to electricity. A lot of excess heat is also created, so this needs to be disposed of. A downside of this kind of power source is that the amount of power it provides decreases as time passes by. This happens according to

$$P_t = P_0 e^{-\frac{\ln 2}{\tau_{1/2}} t}, \tag{4.4.2}$$

where $\tau_{1/2}$ is the **half-life** of the radioactive isotope used in the reactor.

4.5 Power Management and Distribution

The task of the **power management and distribution** (PMD) is to make sure that all the equipment on board gets power, with the right type and voltage.

We can determine several types of PMDs. In a **regulated system** bus power and voltage are fully regulated. In an **unregulated system** only on-off switching of strings is possible. It is a simple system, applied for low mass missions. Other more complicated types are **quasi regulated systems** and **hybrid systems**.

In a PMD the electrical current goes through various parts. Every part has an efficiency η_i. To find the efficiency of the entire path, all the efficiencies simply need to be multiplied.

When using a battery, often a **battery charge regulator** (BCR) and a **battery discharge regulator** (BDR) are present. The BCR adjusts the incoming voltage and current in such a way that battery charging is optimal. This causes the battery life to increase. However, the path efficiency decreases, because the current needs to take an extra step. The BDR does a similar thing as the BCR, but now for the on-board equipment instead of the battery.

5. TT&C and C&DH

5.1 Telemetry, Tracking and Command System

The functions of a **telemetry, tracking and command system** (TT&C) involve **carrier tracking**, **command detection and reception** and **telemetry modulation and reception**. **Telemetry** data are data that are sent by the spacecraft to the ground.

In carrier tracking, the spacecraft receives an **uplink signal** from a ground station. It then sends a **downlink signal** back. It can do this with exactly the same frequency (this is called **phase coherence**). Sometimes the frequency of the uplink signal is multiplied by a **turnaround ratio**. We are then dealing with **coherent turnaround** (also called **two-way coherent mode**). The ground station then receives the downlink signal. This frequency isn't the same as what was send out by the satellite though. A **Doppler shift** has occurred. Using the change in frequency, the velocity of the spacecraft perpendicular to the ground station can be found.

For command detection and reception, and also for telemetry modulation and reception, the satellite needs to be able to communicate with a ground station. This can be done in multiple ways. In **simplex** there is only a one way link. Either the satellite can talk to Earth, or Earth can talk to the satellite. In **half duplex** there is a two way link. However, only one link can be active at a certain time. In **full duplex** there can be a two way link all the time.

5.2 Command and Data Handling System

The function of the **command and data handling system** (C&DH) (sometimes also called the **on-board data handling system** (OBDH)) is, not very surprisingly, to handle commands and data. **Commands** determine the behaviour of the spacecraft. They come from the on-board computer, or from a ground station. A **telecommand** is a command sent by the ground station to the spacecraft.

Let's take a closer look at those telecommands. These commands are usually put in a **packet data field** of 434 bytes. Together with a **packet ID field**, a **packet sequence control field** and a **packet length field** (each having 2 bytes), we have a **source packet** of 440 bytes. The packet ID defines the source of the packet and defines the content of the data field.

There are also **transfer frames**. A transfer frame has a **transfer frame data field**, which contains two of the just described source packets. So this field has a size of 880 bytes. Since 1 byte is 8 bits, it has a size of 7040 bits. In the transfer frame is also a **transfer frame primary header** and a **transfer frame trailer**, each having 48 bits. So the total size of a transfer frame is 7136 bits. This frame is then part of a **transfer frame structure**. This structure also has an **attached sync marker** of 32 bits and a **Reed Solomon check symbol** of 1024 bits. So the total size is 8192 bits. And before it is send, it is also **Viterbi encoded**, making it twice as big. So to send only two commands, we need a couple thousand of bits.

When a command is received by the C&DH system, it is first validated. Are the synchronization code and the message length OK? If anything is wrong, the command is rejected. Otherwise it is executed.

The C&DH system also handles data. It keeps track of housekeeping data (like on-board temperature, current, and such). It processes and stores payload data (such as pictures that were made). It also transmits the data to a ground station when a connection is possible.

Another function of the C&DH system is to keep track of time. This time is often used by the **watchdog timer**. The watchdog timer ensures that the computer is operating normally. The computer should reset the watchdog timer every now and then. If it doesn't, the timer will be reaching a critical value, meaning that something is probably wrong. This will cause the computer to restart.

5.3 Analog and Digital

Most signals, like temperature and voltage, are analog. Sending digital signals has various advantage, so most signals are send in a digital form. Therefore signals need to be converted from analog to digital. This is done by so-called **AD-convertors**. This conversion is done in three steps, being **sampling**, **quantization** and **encoding**.

In the sampling step, we need to determine the **sampling rate** f_s. Theoretically, this must be at least twice the highest frequency in the signal spectrum f_m. In practice, however, a factor 2.2 is necessary. So $f_s \geq 2.2 f_m$. If f_s is higher, then the signal will be more accurate, but more data needs to be send. Using the sample rate, we can split the signal up in pieces called **samples**.

In the second step, the signal will be quantized. First the range of the signal needs to be known. Then a number of bits n needs to be chosen. The range will then be split up in 2^n parts, each having its own magnitude. Every sample will be rounded off to one of these magnitudes. This gives a certain **quantization error**. The maximum quantization error m is given by

$$m = \frac{1}{2^{n+1}}. \tag{5.3.1}$$

So if the error needs to be less than 5%, then $m < 0.05$ and thus $n = 4$. Also often one **parity bit** is added. So for the example this would mean that $n = 5$.

The final step is encoding. Here every sample will get a binary code, specifying its magnitude. Now the analog signal is converted to a digital signal.

6. Telecommunications

6.1 Sending a Signal

The transmission of signals falls under the branch of **telecommunications**. Let's suppose we transmit a signal in all directions. This is done by a so-called **isotropic antenna**. The **transmitter power** P sadly isn't the power that is being emitted. There is so-called **line loss**, designated by the factor L_l. This line loss is similar to an efficiency. It is defined such that the emitted power is equal to PL_i. The power per square meter that a receiver at a distance S from the transmitter receives W_f (the so-called **power flux density**), is then given by

$$W_f = \frac{PL_l}{4\pi S^2}. \tag{6.1.1}$$

It is of course more efficient to send the signal only in the direction of the receiver. We therefore send the signal only to a certain **coverage area**. Now the **antenna gain** is defined as

$$G_t = \frac{\text{Power radiated to the center of the coverage area}}{\text{Power radiated by an isotropic antenna}}. \tag{6.1.2}$$

Once more an additional factor L_a (now called the **transmission path loss**) needs to be added. The power flux density becomes

$$W_f = \frac{PL_tG_tL_a}{4\pi S^2} = \frac{\text{EIRP}\, L_a}{4\pi S^2}, \tag{6.1.3}$$

where the EIRP $= PL_lG_t$ is the **effective isotropic radiated power**.

6.2 Noise

When sending a signal, there is often **noise**. This noise can come from the electronic circuit, from distortions in the radio signal, or from other sources. **Electrical noise** comes from random thermal motions of atoms in the circuit. This thermally generated electrical power is the **noise power spectral density** N_0. It can be found by using

$$N_0 = kT_s, \tag{6.2.1}$$

where $k = 1.38 \cdot 10^{-23} J/K$ is a constant and T_s is the system temperature. The total **received noise power** N can then be found by multiplying N_0 by the **frequency band** B (in Hz). So we find that

$$N = N_0B. \tag{6.2.2}$$

Amplifiers are often used to increase the power of a signal. The signal has a certain power P_{in} as it enters the amplifier, and a different power P_{out} as it exits it. The **amplifier gain** G is the found using

$$G = \frac{P_{out}}{P_{in}}. \tag{6.2.3}$$

The noise of the signal as it enters the amplifier is $N_{in} = kT$. Under ideal circumstances, the noise that comes out of the amplifier is $N_{out} = GkT$. This is however never the case. The noise that comes out of the amplifier is always bigger. Therefore a so-called **amplifier noise temperature** T_n is introduced, such that

$$N_{out} = Gk\left(T + T_n\right). \tag{6.2.4}$$

6.3 Receiving a Signal

How much power does the receiver receive? First we need to know the the diameter of the receiving device D_r. Then the total receiving area is $\pi D_r^2/4$. This isn't the area that is useful, so an efficiency η needs to be added. The **effective receiver aperture area** then becomes

$$A_r = \frac{1}{4}\pi D_r^2 \eta. \tag{6.3.1}$$

The **received power** C is now given by

$$C = W_f A_r = \frac{P L_l G_t L_a D_r^2 \eta}{16 S^2}. \tag{6.3.2}$$

We can write the previous equation slightly different, by using a second type of antenna gain. This gain, denoted by G_r, is defined as

$$G_r = \left(\frac{\pi D_r^2 \eta}{4}\right)\left(\frac{4\pi}{\lambda^2}\right) = \frac{\pi^2 D_r^2 \eta}{\lambda^2}. \tag{6.3.3}$$

If we now also define the **space loss** L_s as

$$L_s = \left(\frac{\lambda}{4\pi S}\right)^2, \tag{6.3.4}$$

then the received power will be

$$C = P L_t G_t L_s L_a G_r. \tag{6.3.5}$$

A digital signal is send by using bits. The number of bits per second is the **data rate** R. The received energy per bit can then be found by

$$E_b = \frac{C}{R}. \tag{6.3.6}$$

Now the so-called **link budget** E_b/N_0 can be found using

$$\frac{E_b}{N_0} = \frac{C}{N_0 R} = \frac{P L_t G_t L_s L_a L_{pr} G_r L_r}{k T_s R}. \tag{6.3.7}$$

Note that the variables L_{pr} and L_r have appeared out of nowhere. They are two more loss factors. L_{pr} is the **antenna pointing error** and L_r is the **antenna loss**.

6.4 Decibels

Multiplying numbers is sometimes difficult. Adding up numbers is a lot easier. Therefore the **decibel** (unit dB) is often used in communications. We can set the unit of a parameter X to dB using

$$X_{dB} = 10 \log_{10} X. \tag{6.4.1}$$

If we would apply this to the right hand side of the link budget equation, we would get

$$P_{dB} + L_{l_{dB}} + G_{t_{dB}} + L_{pr_{dB}} + L_{s_{dB}} + L_{a_{dB}} + G_{r_{dB}} + L_{r_{dB}} - k_{dB} - T_{s_{dB}} - R_{dB}. \tag{6.4.2}$$

However, it does make a difference whether you take a decibel from a variable in W or in mW. Therefore there are more specific decibel-units. The most used are the dBW (the decibel-Watt) and the $dBmW$ (the decibel-mW, also written as dBm). Keep this in mind when working with decibels.

6.5 Modulation

Modulation means joining the info of a low-frequency signal with that of a high-frequency signal. The low-frequency signal is called the **modulating signal**. The high-frequency signal is the **carrier signal**. The signal is modulated at the transmitter side, and then demodulated at the receiving side. **Demodulation** means that the low-frequency signal is recovered.

We can make a distinction between **analog-analog modulation**, **digital-analog modulation**, **analog-digital modulation** and **digital-digital modulation**. The first term indicates the modulating signal type, while the second term indicates the carrier signal type.

We primarily look at analog-analog modulation. The carrier signal here is given by

$$V_c = A_c \cos\left(2\pi f_c t + \phi_c\right), \tag{6.5.1}$$

where A_c is the **amplitude**, f_c is the **frequency** and ϕ_c is the **phase**. Identically, the modulating signal is given by

$$V_m = A_m \cos\left(2\pi f_m t + \phi_m\right). \tag{6.5.2}$$

This gives rise to three types of analog modulation, being **amplitude modulation** (AM), **frequency modulation** (FM) and **phase modulation** (PM). In amplitude modulation, we change the amplitude of the carrier signal to

$$A_c + V_m. \tag{6.5.3}$$

So the modulating signal is hidden in the amplitude of the carrier signal. In frequency modulation, the modulating signal is hidden in the (constantly changing) frequency of the carrier in a similar way. Finally, in phase modulation the modulating signal is hidden in the phase of the carrier signal.

In digital-analog modulation there are also several types. There is **amplitude shift keying** (ASK), **frequency shift keying** (FSK) and **phase shift keying** (PSK). In the latter one are several more variations, of which **binary phase shift keying** (BPSK) and **quaternary phase shift keying** (QPSK) are the most familiar (or the leased unfamiliar) types. Finally, there are more complex modulations possible. Although complex modulations can increase the information-carrying capacity of a channel, it often also leads to an increase in errors.

6.6 Coding

Coding a signal is changing it to a more useful form. There are three branches of coding. In **encryption** you give unwanted listeners a hard time understanding your message. In **compression** you use tricks to reduce the size of your message. Finally, there is **adding redundancy**, which is what we will look at in this chapter.

When sending a digital signal, errors occur. If an error occurs in a digital signal, this usually means that a bit has changed value. The bit will be erroneous. To indicate the quality of the signal, the **bit error rate** (BER) is defined as

$$\text{BER} = \frac{\text{\# of bits recieved in error}}{\text{\# of bits transferred}}. \tag{6.6.1}$$

The chance that a bit gets corrupted is then simply equal to the BER. The chance that an entire message of N bits has no errors then is

$$P_{success} = \left(1 - \text{BER}\right)^N. \tag{6.6.2}$$

For high N, the chance of a message without errors decreases rapidly.

To decrease the chance of an erroneous message, you add redundancy to the system. In **channel coding** (also called **error coding**) you add bits such that errors can be detected or even corrected. In a **bi-directional** system errors can only be detected, while in a **uni-directional** system they can also be fixed. **Forward error coding** (FEC) is the most used form of uni-directional coding.

One way to implement FEC is by using **block coding**. Here you change every message piece (called the word) by a different longer piece (called the codeword). This is done in such a way that no two codewords are similar. Even if one bit changes in a codeword, it is still clear what the original codeword was.

Another method is the **Reed-Solomon code** (RS). When applying this method, often the amount of **message bytes** and **parity bytes** are noted. A (64,40) RS code has a total size of 64 bytes. The message consists of 40 bytes. The remaining 24 bytes are present for error detection and correction.

Another method is **convolutional coding**. The decoder of a convolutional coded stream of bits is called a **Viterbi decoder**. Convolutional coding is more suitable for serial data, encoding a few bits at a time. Read Solomon coding is more suited for big blocks of data.

6.7 Multiple Access

Sometimes multiple persons want to use the same line. How do you then share communication links? There are three ways for this.

In **time division multiple access** (TDMA) all users use the same frequency channel. Each user then gets a certain time slot in which he can use the connection. In this way every user can use the channel with a very high efficiency. However, much communication time gets wasted in this way.

In **frequency division multiple access** (FDMA) each user gets a discrete part of the spectrum, thus operating at a different frequency than the others. Although each user can continuously transmit, there is often interference and low power efficiency. It is also hard to separate the channels.

In **code division multiple access** (CDMA) everyone transmits all the time at the same frequency. Every user has its own pseudo random noise code (PN code). Because of this, the entire signal just looks like noise. However, by multiplying this signal by the right PN code, the correct signal can be extracted. This is called **de-spreading**. This method may have no interference, but the number of users and the data rates are quite limited.

Instead of always operating on the microwave frequencies, it is also possible to use **optical frequencies**. Due to clouds, this is usually an unreliable option for Earth-satellite communication. However, for inter-satellite communication it has a lot of advantages. One of the most important ones is the very high data rates that can be acquired. Accurate pointing of the signal is required though, and this can sometimes be difficult.