NANOSTAR Methodology Documentation

Release 1.0rc2

NANOSTAR Project

Sep 12, 2019

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CHAPTER

ONE

FIRST STEPS

1.1 Nanosatellites and CubeSats

A satellite is an artificial object which has been intentionally placed into orbit. Such objects are sometimes called artificial satellites to distinguish them from natural satellites such as Earth's Moon. Both Nanosatellites and CubeSats are artificial satellites. By one hand and in mass-classification and in strict terms, a nanosatellite (nanosat, nano-satellite) is any artificial satellite with a **wet mass from 1 kg to 10 kg (2.2-22 lb)**. The satellite mass classification is:

- **Large satellites: >1000 kg**
- **Medium satellites: 500 to 1000 kg**
- **Small satellites: <500 kg**
  - Minisatellites: 100 to 500 kg
  - Microsatellites: 10 to 100 kg
  - Nanosatellites: 1 to 10 kg
  - Picosatellites: 100 g – 1 kg
  - Femtosatellites: 10 g – 100 g
  - Attosatellites: 1 g – 10 g
  - Zeptosatellites: 0.1 g – 1 g

By the other hand, "CubeSats" (Cubesatellite, cube satellite) are a **type of nanosatellite defined by the Cubesat Design Specification (CDS)**, unofficially called the CubeSat standard. In 1999, California Polytechnic State University (Cal Poly) and Stanford University developed the CubeSat Specifications to promote and develop the skills necessary for the design, manufacture, and testing of small satellite intended for Low Earth Orbit (LEO) that perform a number of scientific research funtions and explore new space technologies. Uses typically involve experiments that can be miniaturized or serve purposes such as Earth observation or amateur radio.

CubeSats are employed to demonstrate spacecraft technologies intended for small satellites or that present questionable feasibility and are unlikely to justify the cost of a larger satellite. Scientific experiments with unproven underlying theory may also find themselves aboard CubeSats because their low cost can justify higher risks. Biological research payloads have been flown on several missions, with more planned. Several missions to the Moon and Mars are planning to use CubeSats. In May 2018, the two MarCO CubeSats became the first CubeSats to leave Earth orbit, on their way to Mars alongside the successful InSight mission.

With continued advances in the miniaturization and capability increase of electronic technology and the use of satellite constellations, nanosatellites are increasingly capable of performing commercial missions that previously required microsatellites. For example, a 6U CubeSat standard has been proposed to enable a constellation of thirty-five 8 kg (18 lb) Earth-imaging satellites to replace a constellation of five 156 kg (344 lb) RapidEye Earth-imaging satellites, at the same mission cost, with significantly increased revisit times: every area of the globe can be imaged every 3.5 hours rather than the once per 24 hours with the RapidEye constellation. More rapid revisit times are a significant improvement for nations performing disaster response, which was the purpose of the RapidEye constellation. Additionally, the nanosat option would allow more nations to own their own satellite for off-peak (non-disaster) imaging data collection. As costs lower and production times shorten, nanosatellites are becoming increasingly feasible ventures for companies.

According to the CDS, The standard dimensions of the CubeSats are:

- 1U CubeSat is 10 cm × 10 cm × 11.35 cm.
- 2U CubeSat is 10 cm × 10 cm × 22.70 cm.
- 6U CubeSat is 20 cm × 10 cm × 34.05 cm.
- 12U CubeSat is 20 cm × 20 cm × 34.05 cm.

Smallest existing CubeSat design is 0.25U and largest is 27U. Smallest launched CubeSat is 0.25U and largest is 12U. Very soon 16U from US or 20U from China.

There are 2 different classes of deployers/dispensers: **(1)** first type is the classic 4 rails in the corners. It's recommended to be aware of the specfications of modern deployers, because dispensers from ISIS (Innovative Solutions In Space) and NanoRacks allow larger deployables, wider solar panels and thinner rails compared to original P-POD. For example extruded height can be 9 mm instead of 6.5 mm and can be up to 2 kg per 1U which should not cause problems with launch providers. **(2)** Second type is tabs on Planetary Corp Systems (PSC) dispensers. The 13 CubeSats that will fly on SLS in 2018 will use them.

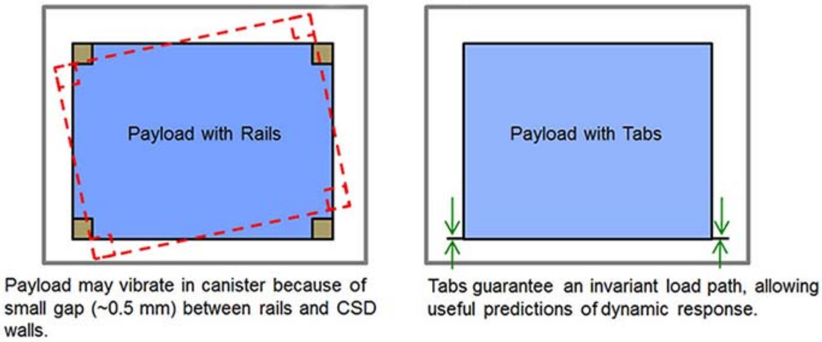

Fig. 1.1: CubeSats with Rails or Tabs comparison by Planetary Systems Corporation (PSC)

Sources: https://www.nanosats.eu/cubesat, https://en.wikipedia.org/wiki/Small_satellite

1.1.1 CubeSat Design Specification

- CubeSat 101: Basic Concepts and Processes for First-Time CubeSat Developers. https://www.nasa.gov/sites/default/files/atoms/files/nasa_csli_cubesat_101_508.pdf
- 1U-3U CubeSat Design Specification Rev. 13. https://static1.squarespace.com/static/5418c831e4b0fa4ecac1bacd/t/56e9b62337013b6c063a655a/1458157095454/cds_rev13_final2.pdf

- 6U CubeSat Design Specification Rev. 1. https://static1.squarespace.com/static/5418c831e4b0fa4ecac1bacd/t/5b75dfcd70a6adbee5908fd9/1534451664215/6U_CDS_2018-06-07_rev_1.0.pdf

Sources: http://www.cubesat.org/

1.1.2 Commercial Off-The-Shelf Components

Commercial off-the-shelf or COTS is an adjective that describes software or hardware products that are ready-made and available for sale to the general public. COTS products are designed to be implemented easily into existing systems without the need for customization. COTS are packaged solutions which are then adapted to satisfy the needs of the purchasing organization.

NASA defined COTS as:

> An assembly or part designed for commercial applications for which the item manufacturer or vendor solely establishes and controls the specifications for performance, configuration, and reliability (including design, materials, processes, and testing) without additional requirements imposed by users and external organizations. For example, this would include any type of assembly or part from a catalog without any additional parts-level testing after delivery of the part from the manufacturer.

Commercial-off-the-shelf (COTS) technologies are playing an increasingly significant role in small satellites. Fig. 1.2 shows the evolution of the secondary payloads (e.g., CubeSats) launched each year including COTS components.

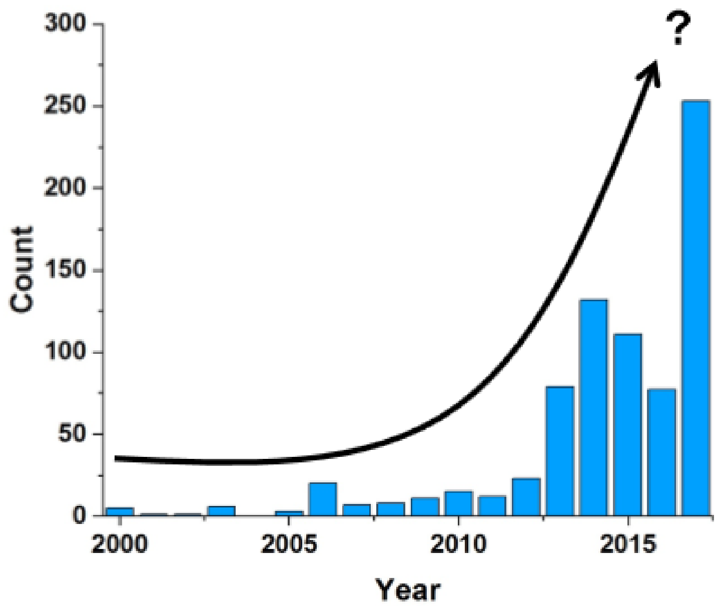

Fig. 1.2: Evolution of the secondary payloads (e.g., CubeSats) launched each year including COTS components. Chart adapted from: M. Swartwout, "Online CubeSat Database," https://sites.google.com/a/slu.edu/swartwout/home/cubesat-database (20-Dec-2017)

For some spacecraft manufacturers, the use of COTS parts is the only option to meet the performance and cost needs of a mission. For many satellite Original Equipment Manufacturers (OEMs), the price and long lead-times of fully-qualified components is simply unaffordable. Today, many COTS devices are operating succesfully in-orbit. The use of COTS components must be an integrated part of your complete design process: from initial parts selection, the assessment of their suitability for use in space, how devices are handled and stored once they arrive in Goods-In, worst-case and reliability analysis, and an approach to hardware design which reflects system reliability.

Sources: https://en.wikipedia.org/wiki/Commercial_off-the-shelf

Where to find COTS?

- CubeSatShop, https://www.cubesatshop.com/
- Pumpkin Inc. (CubeSat Kit), http://www.cubesatkit.com/index.html
- Clyde Space, https://www.clyde.space/products
- ISIS, https://www.isispace.nl/products/
- GOMSpace, https://gomspace.com/shop/subsystems/default.aspx
- Syrlinks, https://www.syrlinks.com/
- Anywaves, http://www.anywaves.eu/accueil/
- dhv Technology, https://dhvtechnology.com/
- Tethers Unlimited, http://www.tethers.com/index.html.
- SpaceQuest communications components, http://www.spacequest.com/components
- Solarmems, http://www.solar-mems.com/space-equipment/
- Space Micro, http://www.spacemicro.com/products.html
- Sputnix, https://sputnix.ru/en/equipment/cubesat-devices/
- Celestia STS, http://www.celestia-sts.com/
- BUSEK, http://www.busek.com/technologies__main.htm
- CUAerospace, http://www.cuaerospace.com/Products
- Sinclair Planetary, http://www.sinclairinterplanetary.com/
- Berlin Space Tech, https://www.berlin-space-tech.com/products/
- Adcole Maryland Aerospace, https://www.adcolemai.com/
- Astronautical Development, LLC. http://www.astrodev.com/public_html2/
- Astro-und Feinwerktechnik Adlershof GmbH, http://www.astrofein.com/astro-und-feinwerktechnik-adlershof/products/
- Blue Canyon Technologies, http://bluecanyontech.com/attitude-control-systems/
- MMADesign, https://mmadesignllc.com/products/
- Nanoavionics, https://n-avionics.com/
- Newspacessystems, http://www.newspacesystems.com/
- Skyfox Labs, http://www.skyfoxlabs.com/products
- Vorago Technologies, https://www.voragotech.com/vorago-products
- Xcam, http://www.xcam.co.uk/products
- Enpulsion, https://www.enpulsion.com/

- Flexitech Aerospace, https://flexitechaerospace.com/products/
- Gumush, https://gumush.com.tr/
- HCT, https://www.helicomtech.com/products

1.1.3 Resources

- Scholz, A. (ed.), CubeSat Standards Handbook: A Survey of International Space Standards with Application for CubeSat Missions, The LibreCube Initative, January 2017. Retrieved from: http://librecube.net/.

CHAPTER

TWO

MANAGEMENT METHODOLOGY

2.1 Systems Engineering in Space Projects

A system is a **collection of components that interact and work synergically to satisfy specified needs or requirements**. Most systems do not exist independently, but are components of a supersystem. The objective of systems engineering is to design, fabricate, operate, and dispose of a system that satisfies a set of specifications in a manner that is **cost effective** and conforms to a predetermined **schedule** with a defined acceptable risk. Space systems engineering discusses the process and techniques to concurrently develop the different subsystems of a sophisticated space system. With the goal being the overall success of the system, the essence of systems engineering is compromise and trade-offs. Space systems development is generally characterized by:

- A broad range of requirements, Procurement in small numbers,
- Significant changes in the applicable technologies over the development cycle,
- Launch costs that are a significant fraction of the total costs, and
- The inability to access the space environment to carry out repairs or upgrades.

As a consequence, space systems are generally unique, robust and reliable, with minimal mass and power.

Systems engineering is the most prominent methodology in space engineering and it is very well supported and documented in the ECSS standards, the most relevant ones used the European space industry. The INCOSE defines systems engineering as:

> Systems Engineering is an interdisciplinary approach and means to enable the realization of successful systems. It focuses on defining customer needs and required functionality early in the development cycle, documenting requirements, then proceeding with design synthesis and system validation while considering the complete problem: Operations, Performance, Test, Manufacturing, Cost & Schedule, Training & Support, Disposal.
>
> Systems Engineering integrates all the disciplines and specialty groups into a team effort forming a structured development process that proceeds from concept to production to operation. Systems Engineering considers both the business and the technical needs of all customers with the goal of providing a quality product that meets the user needs.

As a quick summary of the systems engineering development cycle for a general project, in the next figure, the V-model is represented. In space systems engineering terms, first, the system and its mission is defined through feasibility studies, and then, requirements must be set to ensure the fitness of the solution. Once a preliminar design is achieved and reviewed, the team proceeds to refine the design, to completely define it and to deliver a detailed solution. The next step is its implementation through the assembly, integration, testing and verification phase. Finally, the spacecraft is launched and the mission starts to operate.

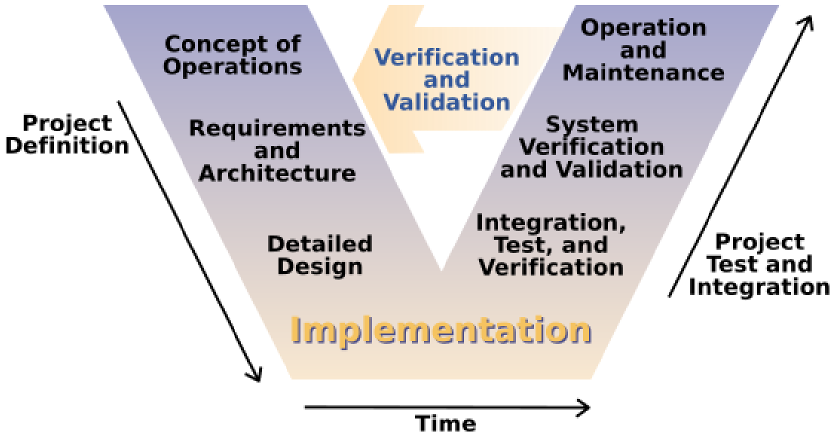

Fig. 2.1: The V-model of the systems engineering process

During the whole process (initial studies, design, production, testing and operation), the ECSS standards provide the number and type of documents that should be released as well as the reviews that should take place. Also, they both recommend and forbid the use of certain technologies and materials in space missions, depending on their specific application. Following these standards is usually a requirement imposed by ESA when working with them and, consequentially, with the most important companies and institutions in the European space industry. Space systems engineering defines a framework of requirements and objectives that must be fulfilled. In this approach, the design phase is usually an iterative top-down process that reaches a solution in compliance with said requirements and objectives. The systems engineer is the person that acts as the link between the different subsystems and coordinates the different aspects of the design process to ensure compliance with the solution.

According to *ECSS-M-ST-10C Rev.1 Project planning and implementation*, a space project is divided into the following phases:

- Phase 0: Mission analysis/needs identification
- Phase A: Feasibility
- Phase B: Preliminary Definition
- Phase C: Detailed Definition
- Phase D: Qualification and Production
- Phase E: Utilization
- Phase F: Disposal

This structure comprises and orders adequately all processes, tasks and work packages in the development of a traditional space mission. Important reviews take place at the end of each phase, i.e. to proceed with the next phase a formal review must be successfully passed. Some of these reviews are:

- MDR: Mission Design Review
- PDR: Preliminary Design Review
- PRR: Preliminary Requirements Review

- CDR: Critical Design Review
- AR: Acceptance Review
- LRR: Launch Readiness Review
- ELR: End-of-Life Review

Phase 0 and A usually happen together and are involved in the detection and development of an idea and its feasibility, producing at the end the PRR. Phase B is when a more precise definition of the mission and its design is made. This means that the first specific studies are carried out and the most important decisions have been made. For the end of Phase C, the design must be frozen as well as the rest of the decisions that will happen in the future, like the AIT (Assembly, Integration and Test) plan and the project schedule. This information is completely settled for the CDR. The assembly, integration, testing and verification are the most important tasks in Phase D, when everything is subjected to a qualification process that ends with the launch and beginning of the mission. Finally, the mission ends.

2.2 Concurrent Engineering

The concurrent design (CD) consists of an **engineering design style** based on a continuous flow of information. This information is shared as parameters and design values, previously defined in the database. All subsystems from a space project and the interfaces between them must be defined using these design parameters. The CD allows **instantly updating** any design modification that may affect other subsystems, so it is transmitted to all the computers in order to share this update to the rest of the subsystems. In this way, the CD method saves time while avoiding many errors due to design changes that may not be communicated to others responsible.

The concurrent design philosophy implies an **iterative design working**: the subsystems are defined and the information is shared and updated until the mission requirements are fulfilled. This is called an "iteration". Once a closed design is achieved, a new iteration modifying the necessary design parameters is carried out. This methodology allows achieving an optimization of the previous iteration, whether of size, cost, weight, etc. Once the iterative process has been completed, the results are studied to verify the viability of the project (phases 0 and A of the project).

Among the main advantages of concurrent design, specifically of its application in the field of space missions, it must be highlighted the time reduction to obtain relevant information to verify the feasibility of a complex mission, such as be the total power required, the volume or the preliminary cost of the mission.

By applying the concurrent design in a CDF, the communication between the subsystems responsible improves considerably, since they are all in the same place and at the same time. A CDF or Concurrent Design Facility is an advance design room equipped with a network of interconnected computers, multimedia devices and different software that allows implementing CD method.

The ESA has adopted the next definition for the Concurrent Engineering:

> Concurrent Engineering (CE) is a systematic approach to integrated product development that emphasises the response to customer expectations. It embodies team values of co-operation, trust and sharing in such a manner that decision making is by consensus, involving all perspectives in parallel, from the beginning of the product life-cycle.

Essentially, CE provides a collaborative, co-operative, collective and simultaneous engineering working environment.

When a space project starts at the CDF, the first step is to define completely the mission. One of the following options must be followed, depending on two different situations:

- If the client has provided the requirement list, you must check all these requirements in order to understand them and to assign each one of these requirements to one or several disciplines (all the subsystems defined in a satellite).
- If the client gives you a description list of the properties that must fulfill the satellite, you must extract from this list all the nominal requirements by analysing the description of the mission.

The next step is to create and coordinate the teams that must work on the satellite design depending on the defined requirements. The teams shall be defined depending on the different subsystems involved in the project (e.g. the Attitude Determination and Control subsystem, the structural subsystem, the communications subsystems. . .).

During the pre-design steps, you must have in mind that some of the solutions are trade-offs between two or more subsystems, e.g. increase the total external surface generates a larger total power (from the solar panels) but increase the satellite weight.

It also must be highlighted that some subsystems generate a close loop with some parameters, e.g. the system engineer needs the mass and power consumption from all the subsystems; but the power subsystem engineer also needs the power required. This required power from all subsystems affects to the power consumption of the power subsystem. The solution of this problem is to estimate a first value of one of these parameters and make iterations until a trade-off solution is achieved.

2.2.1 CDF Design Typicall phases

The main objective of the concurrent engineering process is to ensure that the study meets the customer requirements within the time and cost assigned. The process make the most efficient and effective use of experts and their tools in order to create a design.

An important challenge for each team is to develop a process that is consistent and repeatable, but flexible enough to allow the necessary changes during a concurrent engineering study/session. Given that the members of a concurrent engineering team generally vary according to the studies, it is important to have consistent processes in order to generate easily traceable results and reduce the variation in the output products of the study. It is not required that the process be the same in the concurrent teams in different centres, but it is necessary to define the interfaces between the different teams that carry out distributed collaborative design sessions, similar to the interface agreements between subsystems.

A consistent systematic process is essential to reach a conclusion and finalize a design (including results documentation) in an allotted time. The individual secondary steps differ in response to the needs of the customer and the requirements and composition of the individual concurrent teams. The scheme shown in Fig. 2.2 captures, at the highest level, a representative process for a design study sequence. This sequence begins with the customer initial mission concept and goes to the final products. The details of each of the steps can vary between the concurrent engineering teams, but the main steps are still quite similar. The amount of time it takes to complete a particular step or study can vary from days to weeks or months, depending on the level of detail of the study or the complexity of the mission concept.

1. Establishing the scope

In order to make the most of the design team, it is essential to start the study with a solid problem definition. The team lead/study facilitator meets with the customer to understand the problem to be solved and develop the requirements list for the study. The team leader and the customer agree with the objectives of the design study, the figures of merit, the required products and any engineering or other

Fig. 2.2: Typicall Concurrent Engineering Process

study constraints. The level of effort, the time of completion and the cost of the complete study vary depending on the scope and level of detail of the analysis and desired products.

2. Pre-Study background work

The amount of background work done before the concurrent engineering session varies according to the team and the type of the mission being evaluated. Before the study, team members should review similar previous missions and perform early work on long-term subsystems (eg, mission analysis). They can also discuss specific aspects of the mission with the client to gain a better understanding of the requirements and limitations of the mission before the design session.

3. Full-team concurrent design sessions

A design session is the physical or virtual meeting during which the members of the concurrent team perform the necessary analyses to design a collective system. The customer could be an active participant in the design team sessions. The activities and the results of the design effort depend on the type of study that is carried out and on the level of maturity of the mission concept. Different teams develop their designs in different time scales, depending on the amount of work done in real-time concurrent sessions versus independent work outside the sessions. The study may vary from the first feasibility studies of the mission to determine if a concept is viable, to the detailed designs of converging points, including the design of high-level subsystems, to the very detailed design of the system and the levels of subsystems, including cost estimates and schedules. During the design session, the concurrent design team works with the customer to achieve the desired level of fidelity. The designs (or a set of architectures for trade studies) are repeated until they converge, or are determined to be nonviable. Convergence is generally driven by a combination of certain key parameters and constraints, which generally include mass, power, cost and fitting within the launch vehicle constraints.

4. Post-session documentation and presentations

Although there is a great variation in the activities of the post-design sessions between teams, all of these teams develop a product that documents the final design using a coherent and consistent template, presenting also the design to the client. Products can include PowerPoint slides, text documents, trajectory files and configuration drawings, among others.

2.2.2 CDF Studies at the ESA/ESTEC

The following is a list of some of the studies/reviews carried out by the CDF at the ESTEC in 2018:

Table 2.1: Studies/reviews carried out by the CDF at the ESTEC. Source: http://www.esa.int/Our_Activities/Space_Engineering_Technology/CDF

Title	Customer
VEGA Standardisation MICRA Study	GSP
MICE Giants Assesment of Orbiter or Probe for Neptune or Uranus	SCI
EnVision M5 Assesment of Mission to Venus	SCI
RGE Rover Garage Element for Lunar ISRU Studies MICRA Study	HRE
THESEUS M5 Assesment of Transient High Energy Sky Surveyor	SCI
SPICA M5 Assesment of Cryobenic Infrared Telescope for ESA JAXA Mission	SCI
ATHENA SIM Phase A and B1 Continuation of ATHENA	SCI
MSR Mars Sample Review Reviews and Phase A/B1 work	HRE
ISRU Technology Demonstrator Design of Components for Lunar ISRU	HRE
QPPF Quantum Physic Study to Test Quantum Superposition Theory	SCI
LiteBird Next Generation measurement of Polarisation of Cosmic Microwave Background	SCI

An good example of a study developed by the ESA at the CDF is the ASIM Mission: Atmospheric monitoring of blue jets, sprites and elves. It was a completed study for the Directorate of Human Spaceflight, Microgravity and Exploration showing the feasibility of accommodating the atmospheric Space Interactions Monitor (ASIM) payload on the International Space Station (ISS) Columbus External Platform Facility (CEPF).

The ASIM payload has been proposed by the Danish Space Research Institute (DSRI) to observe Transient Luminous Events (TLEs) that occur in the Earth's upper atmosphere accompained by thunderstorms in the lower atmosphere. These events are known as blue jets, sprites and elves and they were first observed at 2003, a year a before the ASIM mission was studied at the CDF.

The CDF assessed the viability of conduncting science observations and conducted technical analyses to specify a design concept compatible with the ISS interfaces and constrainsts. The contamination and radiation environment and the instrument viewing requirements were analysed and taken into account. The CDF engineers designed equipment for power distribution, data handling and computing based on standard hardware avaliable to ESA external payload.

Launch: April 2nd, 2018

Vehicle: Dragon

Weight: 314 kg

Power consumption under nominal conditions: 200 W (including survival heaters: 430 W)

Data transmission: 200 kbit / s continuous

Main contractor: Terma, Denmark

Control center: B.Usoc, Belgium

Source: http://www.esa.int/Our_Activities/Space_Engineering_Technology/CDF

2.3 Agile Development

The agile development methodology was originated in the software industry and tries to facilitate software development in an iterative and incremental way, where requirements and solutions evolve through the collaborative effort of self-organising and cross-functional teams and their customers.

Agile system engineering practices are well established methodologies in software projects and are now being explored and studied to be applied in complex hardware projects. These practices permit a flexible and development working environment while allowing risk uncertainties to be managed in a disciplined manner.

The main principles are collected in the Manifesto for Agile Software Development, which are:

- Customer satisfaction by early and continuous delivery of valuable software
- Welcome changing requirements, even in late development
- Deliver working software frequently (weeks rather than months)
- Close, daily cooperation between business people and developers
- Projects are built around motivated individuals, who should be trusted
- Face-to-face conversation is the best form of communication (co-location)
- Working software is the primary measure of progress
- Sustainable development, able to maintain a constant pace
- Continuous attention to technical excellence and good design
- Simplicity –the art of maximizing the amount of work not done– is essential
- Best architectures, requirements, and designs emerge from self-organising teams
- Regularly, the team reflects on how to become more effective, and adjusts accordingly

These principles were written by software engineers with software engineering in mind, but they can be adapted to nanosatellite engineering easily, since it is a good methodology for small teams where everyone works with every subsystem, conforming a cross-functional team.

Another important difference is that nanosatellite engineering is focused on getting a physical system (the nanosatellite), that requires both software and hardware. This fact changes completely the paradigm, because hardware is not as iterable and revisable as software, so testing can happen as quickly and frequently as the agile methods suggest. Having that issue in mind, these methodologies can still be applied in the preliminary design phase, since no hardware is usually involved.

The principles that sustain this methodology decrease the constraints of development procedures and increase the responsiveness to the sponsor of the project. Some key points are self-organizing teams and a minimum number of documents written, maximizing its content and improving them continuously.

The main idea of these methods is to break the product development into small increments and tasks instead of approaching the problem with big work packages. In each short iteration, all members of the team work in design, analysis and testing/review/quality checking, with daily quick meetings to update their progress to the rest of the team. This process is similar to the first steps in a concurrent engineering process, but in this case every member of the team works in all parts of the software, so everyone gets a global view of the product.

Another difference with concurrent engineering is that these practices are applied throughout all the product development, something feasible with small teams. Applying these ideas are also possible for student teams, since their knowledge may not be as specialised as in traditional projects and they can benefit from working in the different subsystems that compose a nanosatellite.

As an example, we shall see how this methodology has been adopted for nanosatellite development at the Johns Hopkins University in USA and at the Observatoire de Paris' CERES campus. The latter made use of student teams and some lessons learnt are described in *[Segret]* (page 73).

A first design cycle was carried out applying a concurrent engineering model named by ESA the "Spiral Model". The iteration starts with a mission requirement analysis that leads to the actual mission analysis. Then, the subsystem design step takes place and later it is followed by a design verification. Finally, risks are assessed before re-examining the mission requirements, completing the iteration. This cycle took 600 h of study. Some conclusions mentioned by the author include how time consuming is to manage a team of nine students that does not have real engineering experience and that shows heterogeneity in their works. One issue that happened was a deep misunderstanding of the problem that was tackled in time by an intermediate study report.

The second design cycle followed the agile principles. A common workspace (a FTP server) was set up. Each team of designers had a folder where to submit their deliverables. Important and valuable deliverables (documents, software, specifications, test routines, etc) were in a read-only folder that was frequently updated by the project manager, with backups of previous iterations.

During this cycle, there were periods of time, called *runs*, during which all efforts were concentrated towards a specific goal, with a duration of two or three weeks. The *chairman* oversees the run, usually a system engineer, the project manager or the deputy of the systems engineer. He/she requests every designer to deliver their work in their folders before a deadline and asks for the roadmap that the designer is going to follow in this run. Right after the deadline, the chairman analyses the deliveries and prepares a meeting that everyone attends, and where they have time to present a topic of interest, discuss the feedback and check the deliveries. Finally, the chairman decides what is saved in the common data archive, with a cretain confidence level in the results.

Other important roles that are set for each iteration are the *tracker* and the *testman*. The first one is in charge of preventing misunderstandings and bad communication between the different designers. The testman must test the project and advise the designer about possible improvements to make the tests more accurate, useful and usable by others.

This methodology can be combined with the concurrent engineering approach very easily and it does not require a linear flow as rigid as the traditional one: some subsystems could be more developed than others, for the flexibility and adaptability of these principles can accommodate easily major changes.

- An interesting video about how they apply Agile Development at Spotify: https://labs.spotify.com/2014/03/27/spotify-engineering-culture-part-1/

CHAPTER

THREE

PRELIMINARY DESIGN

3.1 Command & Data Handling

The **Command and Data Handling** (C&DH) system performs two major functions: **(1)** It receives, validates, decodes, and distributes commands to other spacecraft systems and gathers, processes; **(2)** and formats spacecrafts housekeeping and mission data for downlink or use by an onboard computer. This equipment often includes additional functions, such as spacecraft timekeeping, computer health monitoring (watchdog) and security interfaces.

The C&DH subsystem is the “brain” of the whole spacecarft. It consists of an Onboard Computer, OBC, which controls the operation of the satellite. The OBC has software installed that manages the programs written to handle various tasks; for example, a program whose function is to create a telemetry stream about the status of the payload and then encode the stream. The primary component of the OBC is the microcontroller.

The objective of the C&DH subsystem is **to provide the spacecraft with operation sequences to various subsystems**. Because of the size restrictions of the typicall satellites, the C&DH subsystem needs to be efficient, small, lightweight, and easy to integrate with all of the other subsystems. The ideal C&DH system is one which has been proven on another spacecraft and which requires no modification for the mission under development. However, new missions are usually supported by systems which evolve from older designs.

Table 3.1 summarizes a list of different space missions in terms of S/C mass. It includes de misions name and objetives and the spacecraft size, mass, power consumption, and the C&DH equipment configuration.

Table 3.1: List of space mission and S/C properties (name, type, size, mass, power, C&DH) to preliminary size the C&DH Subsystem

Satellite	Mission	Size	Mass	Power	C&DH Subsystem
MCubed-2 / COVE	Imaging / Optical	10 cm x 10 cm x 10 cm (1U)	~ 1 kg	1.2 W of average power, 4.7 W of peak power	Stamp9G20 microcontroller (OBC), 400 MHz ARM9 core, 54 MB SDRAM, 128 MB NAND flash
OUFTI-1	Tech / Navigation / Communication	10 cm x 10 cm x 10 cm (1U)	~ 1 kg	2.2 W	Pumpkin FM430 microcontroller (OBC), two 2GB SD cards for data storage
StudSat-1	Imaging / Optical	10 cm x 10 cm x 13.5 cm (1U)	1.3 kg	3.2 W	Centralized 32-bit Atmel ARM based AVR32 UC3A0512 microcontroller (OBC), 512 kB FRAM
Galassia	Tech / Navigation / Communication	10 cm x 10 cm x 20 cm	2 kg	2 W	16 bit DSPIC33 Microcontroller (OBC), ARM7 Processor
Raiko	Imaging / Optical	10 cm x 10 cm x 23 cm (2U)	2.66 kg	3 W (Paddles closed), 5.9 W (Paddles open)	A Main Processor Unit (MPU), Power Control Unit (PCU), Flash memory
GOMX-1	Imaging / Optical	10 cm x 10 cm x 20 cm (2U)	2.66 kg	3.5 W	NanoMind A712 Microcontroller (OBC), 2 GB of data storage

Continued on next page

Table 3.1 – continued from previous page

Satellite	Mission	Size	Mass	Power	C&DH Subsystem
QBITO	Space Science	10 cm x 10 cm x 23 cm (2U)	2.66 kg	•	Electronic Systems CubeComputer V3 (OBC)
PicSat	Space Science	10 cm x 10 cm x 30 cm (3U)	3.5 kg	•	ISIS ARM9 Processor (OBC), two 3 GB SD cards
Aalto-1	Tech / Navigation / Communication	10 cm x 10 cm x 34 cm (3U)	~ 4 kg	4.8 W	AT91RM9200 ARM 9 Processor (OBC), External RAM, EEPROM, DataFLash, NAND Flash and microSD for data storage, two 32 MB SDRAM modules
INSPIRE	Tech / Navigation / Communication	10 cm x 10 cm x 30 cm (3U)	~ 4 kg	20 W	Pumpkin MSP430 processor (OBC), Two 2GB SD cards for data storage
VELOX-1 NSat	Imaging / Optical	10 cm x 10 cm x 34 cm (3U)	4.28 kg	28.8 W	Main board with 100 MHz 8051 MCU (OBC), 2 GB SD cards for data storage, 32 MB RAM
OPS-SAT	Tech / Navigation / Communication	10 cm x 10 cm x 30 cm (3U)	5.4 kg	•	GomSpace Nanomind A712D
SPROUT	Imaging / Optical	21.3 cm x 21.3 cm x 21 cm	~ 7.1 kg	•	2 C&DH (1 for the Engineering misión and another for the outreach mision)

Continued on next page

Table 3.1 – continued from previous page

Satellite	Mission	Size	Mass	Power	C&DH Subsystem
ASTERIA	Space Science	11.6 cm x 23.9 cm x 36.5 cm (6U)	10.16 kg	48 W	Spaceflight Industries CORTEX 160 Flight Computer, Xilinx Virtex 4FX / PowerPC405, 14.5 GB for data storage
MasCO	Space Science	11.8 cm x 24.3 cm x 36.6 cm (6U)	~ 12 kg	35 W	AstroDev MSP430F2618 Microcontroller (OBC), 128 kB flash memory, 8 kB RAM
IceCube	Space Science	11.8 cm x 24.3 cm x 36.6 cm (6U)	~ 14 kg	120 W	Proton P400K-SGMII-2-PCI104S-SD Space Computer (OBC), Freescale advanced 45 nm dual-core microprocessor
BUAA-Sat	Imaging / Optical	30 cm x 30 cm x 50 cm	~ 30 kg	11 - 15.5 W	32 bits ARM RISC Processors, PC2294 of 60 MHz, SRAM 2 MB

3.2 Attitude Determination and Control

3.2.1 Introduction

The Attitude Determination and Control System (ADCS) is required when the orientation (*attitude*) of the spacecraft is important and must be known and controlled. It stabilises the vehicle and orients it in the desired directions during the mission despite external disturbance torques acting on it. The vehicle uses sensors to determinate the attitude and controls it with actuators.

A very clear example of when this subsystem is needed is a observation mission: if the payload requires pointing to a specific direction, the ADCS shall act to provide this pointing in a given time with the required stabilisation and accuracy. Therefore, this subsystem plays an important role for the payload or communications. Other examples are accurate pointing of thrusters for orbital manoeuvres or to use intelligently the heating and cooling effects of sunlight and shadow for thermal control.

We must keep in mind that the space environment is constantly interacting with the vehicle. Solar radiation, the gravity field generated by near celestial bodies, their magnetic field or in low altitude orbits the atmosphere act on the spacecraft causing disturbances in the form of forces (important for the orbital motion) and torques (important for the attitude motion). These torques can be cyclic if they vary in a sinusoidal manner during an orbit or secular if they accumulate with time and do not average out over an orbit. Without any counter measures, these torques will quickly reorient the vehicle. The ADCS is the subsystem in charge of providing this resistance, passively (exploiting certain characteristics of the vehicle) or actively (measuring the torques and applying torques in the opposite direction). Also, the ADCS is in charge of performing manoeuvres to reorient the vehicle to point to another direction, like taking new photographs or pointing the antenna to another ground station or vehicle.

Spacecraft attitude changes according to the fundamental equations of motion for rotational dynamics, the Euler equations, that in a body-fixed reference frame and expressed in vector form are:

$$\dot{\mathbf{H}} = \mathbf{T} - \omega \times \mathbf{H}$$

Attitude determination is the process of combining the information provided by different sensors with knowledge of the spacecraft dynamics to provide an estimation of the orientation state of the vehicle as function of time. These sensors use information from the environment, like sunlight, the position of the stars or the horizon of the Earth. Another important concept is **attitude control**, which is the process carried out by the spacecraft to achieve a certain attitude state in a passive or active way.

3.2.2 Steps in Attitude System Design

1. **Define control modes and system-level requirements**: given the mission requirements, the mission profile and the type of insertion from the launch vehicle, one can specify the control modes during the complete mission, establish the requirements for the ADCS and detect its constraints.
2. **Quantify the disturbance environment**: depending on the mission, the ADCS may have to correct and maintain certain conditions that will be disturbed by the environment in the form of torques. For that, it is necessary to have the spacecraft geometry, the mission profile, the orbit and some environment models.
3. **Select type of spacecraft control by attitude control mode**: select between 3-axis stabilisation, spinning stabilisation, gravity gradient, etc given the payload, power and thermal needs and the accuracy requirements.
4. **Select and size ADCS hardware**: select or size the sensors, the actuators and the data handling avionics for the spacecraft considering the requirements, the accuracy and the characteristics of the mission.
5. **Define determination and control algorithms**:
6. **Iterate**

3.2.3 Control Modes and Requirements

3.2.4 Spacecraft Control Type

- **Passive control**
    - Gravity-gradient control: uses the inertial properties of a vehicle to keep it pointed toward the Earth. An elongated object tends to align its minimum principal axis to

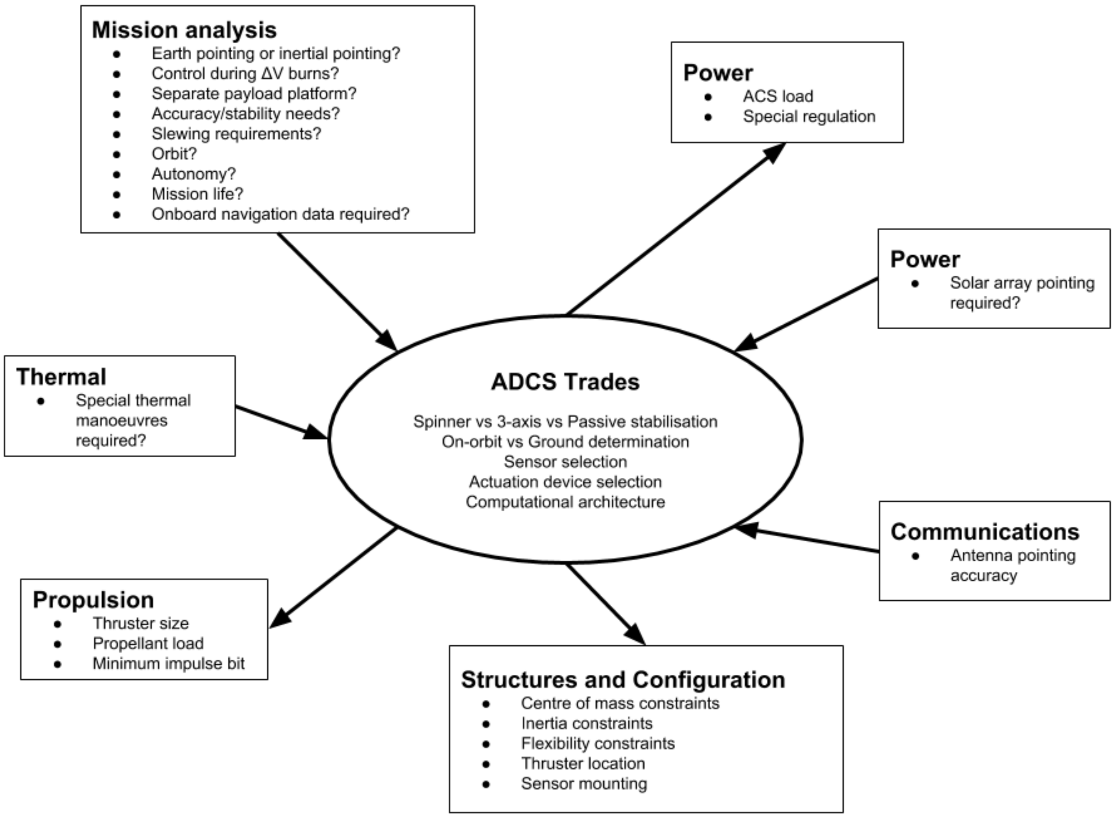

Fig. 3.1: The impact of mission requirements and other subsystems on the ADCS. Adapted from [WL99] (page 74)

  nadir, thus leaving only one rotational degree of freedom free and that can be controlled with a momentum wheel.

  – Spin stabilisation: the entire spacecraft rotates so that its angular momentum remains approximately fixed in inertial space. It is stable if it rotates about the principal axis with the largest moment of inertia (so disk-shaped vehicles).

  – Dual-spin stabilisation: there is one part of the spacecraft that rotates and that provides the spin stabilisation in a similar fashion to the previous technique. It allows for more complex shapes since the rotor can be disk-shaped whereas the platform may not be like that. The platform can also not be rotating.

- **Active control**: they are more common nowadays although they are more complex and expensive. They can manoeuvre relatively easily and can be more stable and accurate. They require the use of actuators, sensors and control laws to work properly since they do not rely on the environment or underlying physics to provide the control.

In Table 3.2 are summarised the most important aspects of each control type.

Table 3.2: Spacecraft Control Type, from [WL99]

Type	Pointing Options	Attitude Manoeuvrability	Typical Accuracy	Lifetime Limits
Gravity-gradient	Earth local vertical only	Very limited	±5º (two axes)	None
Gravity-gradient and momentum wheel	Earth local, vertical only	Very limited	±5º (three axes)	Life of wheel bearings
Passive Magnetic	North/South only	Very limited	±5º (two axes)	None
Pure spin stabilization	Inertially fixed any direction	Repoint with precession manoeuvres; very slow with torquers, faster with thrusters	±0.1º - 1º (two axes)	Thruster propellant (if applies)
Dual-spin stabilization	Limited only by articulation on despun platform	Momentum vector same as above. Despun platform constrained by its own geometry	±0.1º - 1º (two axes)	Thruster propellant (if applies). Despin bearings
Momentum wheel (1 axis)	Best suited for local vertical pointing	Momentum vector of the bias wheel prefers to stay normal to orbit plane, constraining yaw manoeuvre	±0.1º - 1º	Propellant (if applies). Life of sensor and wheel bearings
Thruster (3 axes)	No constraints	No constraints, high rates possible	±0.1º - 5º	Propellant
Reaction wheels (3 axes)	No constraints	No constraints	±0.0001º - 1º	Propellant (if applies). Life of sensor and wheel bearings
Control momentum gyroscopes (3 axes)	No constraints	No constraints, high rates possible	±0.001º - 1º	Propellant (if applies). Life of sensor and wheel bearings

3.2.5 Disturbance Sizing

The ADCS should be able to accommodate the environment, therefore it is important to size the disturbances that it introduces in the attitude motion of the satellite.

Since most of these disturbances decrease with the altitude, we shall use the values at the periapsis as the dimensioning altitude of our system. This will ensure that our estimations are valid for the most extreme cases. Finally, we shall add all the disturbance torques to have the total disturbance torque, which will be the dimensioning value for the actuators.

Gravity-Gradient Torque

Due to this disturbance torque, satellites tend to align its longitudinal axis with the nadir-zenit direction. This is a constant torque for Earth-oriented vehicles, and cyclic for inertially oriented vehicles, and it is mostly influenced by spacecraft inertias and the orbit altitude. Assuming that the axis with the minimum moment of inertia is the longitudinal axis, the equation that models this phenomenon is:

$$T_g = \frac{3\mu}{r_p^3}|I_{\max} - I_{\min}| \sin(2\theta)$$

- $\mu = GM$: standard gravitational parameter. For Earth, μ is 3.986·10^{14} m^3/s^2.
- r_p: periapsis radius, in meters.
- $I_{\min} = \min\{I_x\ ,\ I_y\ ,\ I_z\}$: minimum moment of inertia in kg·m^2.
- $I_{\max} = \max\{I_x\ ,\ I_y\ ,\ I_z\}$: maximum moment of inertia in kg·m^2.
- θ: maximum angular deviation between the longitudinal axis and the nadir-zenit direction.

Solar Radiation Pressure (SRP) Torque

In asymmetric satellites (almost all of them), the incident electromagnetic radiation causes a pressure and a shear stress over the intercepted surface. This radiation comes mostly from the Sun, therefore the solar irradiance flux is used to estimate this disturbance effect and it depends mostly on the distance between the vehicle and the Sun. It is a cyclic torque for Earth-oriented vehicles and constant for solar-oriented ones. It is influenced primarily by spacecraft geometry, spacecraft surface reflectivity and the position of the centre of gravity.

In Earth's orbit, there is an annual variation associated with the distance to the Sun since Earth's orbit has a little eccentricity. The annual maximum and minimum values of the solar irradiance flux are 1412 and 1321 W/m^2, respectively.

$$T_{sp} = \frac{\Phi}{c} A_{sp}(1+q)\cos(i)||\mathbf{c}_{sp} - \mathbf{c}_g||$$

- Φ: solar flux. At Earth's orbit is around 1367 W/m^2, with a 3% variation.
- c: speed of light, around 3·10^8 m/s.
- q: reflectivity coefficient.
- i: sun light incident angle.
- A_{sp}: projected area in Sun direction, in m^2.
- c_g: centre of gravity position in metres.
- c_{sp}: solar pressure centre position in metres.

Aerodynamic Torque

For satellites in LEO, the interaction of their surface with the residual atmosphere should not be ignored. It is constant for Earth-oriented vehicles and variable for inertially oriented ones. Orbit altitude, spacecraft geometry and the position of the centre of gravity are the main parameters that influence in this torque.

This is the most important disturbance for orbits with an altitude below 400 km.

$$T_a = \frac{1}{2}\rho_{\text{atm},p} C_D A_a V_p^2 ||\mathbf{c}_a - \mathbf{c}_g||$$

- $\rho_{\text{atm},p}$: atmospheric density at the periapsis, in kg/m^3.
- C_D: drag coefficient, dimensionless, usually between 2 and 2.5.
- A_a: projected area in the velocity direction, in m^2.
- $\mathbf{c}_a$: pressure centre position in metres.
- $\mathbf{c}_g$: centre of gravity position in metres.
- V_p: vehicle velocity at the periapsis, in m/s.

Magnetic Torque

This torque is the result of the interaction between Earth's magnetic field (or other central body) and the residual magnetic dipole of the vehicle, which is usually minimised in the final design phases of the electrical harness and other components but it is rarely null. This torque is cyclic and depends mostly on orbit altitude, the residual magnetic dipole and orbit inclination.

As a first calculation, we can use a simple model: Earth's magnetic field can be modelled as a dipole

$$T_m \approx D_m \frac{\lambda M_m}{r_p^3}$$

- M_m: magnetic moment of the central body. For the Earth, the value is 7.96·10^{15} T·m^3. For other celestial bodies, this value can be computed from the spherical harmonic representation of the magnetic field of that body. Wertz describes the procedure in Appendix H of *[Wer78]* (page 74) for the Earth, reaching a similar value in equation (H-18).
- r_p: periapsis radius, in metres.
- D_m: vehicle residual magnetic dipole, in m^{2}A. Usually in the range 0.1-20 A·m^2
- λ: unitless function of the magnetic latitude that ranges from 1 at the magnetic equator to 2 at the magnetic poles

Other disturbance torques

- Uncertainty in centre of gravity
- Thruster misalignment
- Mismatch of thruster outputs
- Reaction wheel friction and electromotive force
- Rotating machinery (pumps, filter wheels)
- Liquid slosh
- Dynamics of flexible bodies
- Thermal shocks on flexible appendages

3.2.6 Selection and Sizing of ADCS components

Actuators

Reaction wheels

Reaction wheels are torque motors with high-inertia rotors. They can spin in either direction and provide one-axis control for each wheel. To compensate the disturbance torques these actuators increase or decrease their angular rate, so it is important to *desaturate* them (process known as *momentum dumping*) once they reach their maximum angular rate. This is usually done with thrusters or magnetorquers. Also, we should keep in mind that some of these disturbances are cyclic so our design should take that to size our components.

To estimate the maximum torque that the reaction wheel should be able to generate to compensate disturbances ($_d$) or for a slewing manoeuvre, ($_s$), we can use (3.1), where T_D is the total disturbance torque, SM is a margin of security (usually 20%), θ is the maximum angle of rotation in degrees of the slew manoeuvres, I is the inertia momentum (kg·m^2) about the axis of the slew manoeuvre (maximum for a conservative estimation) and t_s is the minimum duration of the slew manoeuvre in minutes.

$$\begin{aligned} T_{RW} &= \max\{T_{RW,d}, T_{RW,s}\} \\ T_{RW,d} &= T_D(1+SM) \\ T_{RW,s} &= 4\theta\frac{\pi}{180}\frac{I}{(60\cdot t_s)^2} \end{aligned} \tag{3.1}$$

Another important value for reaction wheels is the stored angular momentum H_{RW} (N·m·s), which can be estimated with (3.2), where P is the orbital period in seconds.

$$H_{RW} = T_D \cdot P \tag{3.2}$$

Magnetorquers

Magnetorquers are devices capable of changing its magnetic dipole so they can interact with the surrounding magnetic field. This means that they only work if there is a strong magnetic field, like Earth's, nearby (like LEO).

Briefly, when a voltage is applied, the electrical current generates a magnetic dipole and, as a result, the winding tries to align itself with the local magnetic field. This magnetic dipole $D_{m,MT}$ (A·m^2) is estimated with (3.3).

$$D_{m,MT} = T_D\frac{r_p^3}{2M_m} \tag{3.3}$$

where M_m is the magnetic moment of the central body in T·m^3 (see Magnetic Torque) and r_p is the radius of the periapsis in metres.

Momentum wheels

Momentum wheels are reaction wheels with nominal spin rate above zero to give spin-stabilisation around their rotation axis since they provide a nearly constant angular momentum.

$$\begin{aligned} T_D \frac{P}{4} &= H_{MW} \cdot \delta \frac{\pi}{180} \\ H_{MW} &= \omega_{MW} \frac{2\pi}{60} \cdot I_{MW} \\ m_{MW} &= \frac{2H_{MW}}{\omega_{MW} \frac{2\pi}{60} \cdot R_{MW}^2} \end{aligned} \tag{3.4}$$

where

- P: orbital period (s).
- T_D: disturbance torque (N·m).
- δ: maximum angular deviation allowed in degrees.
- ω_{MW}: angular velocity in rpm.
- $I_{MW} = m_{MW} R_{MW}^2$: inertia momentum of the wheel (kg·m^2)
- m_{MW}: wheel mass in kg
- R_{MW}: wheel radius in m.

Thrusters

Thrusters provide a thrust in their pointing direction and therefore they can provide a torque if positioned properly. They require propellant so their lifetime is limited.

Thrusters can be used to compensate the environmental disturbances, so for a pair of thrusters separated L (m) from the centre of mass, the required thrust would be:

$$F_D = \frac{T_D}{2L}$$

In a zero-momentum system (a system with reaction wheels), the torque that a couple of thrusters should provide for a slew manoeuvre is:

$$\begin{aligned} T = 2F_{RW} L = I\ddot{\theta} = I \frac{\dot{\theta}}{t_{ac}} &= I \frac{\theta}{t_b t_s} \\ F_{RW} &= \frac{I}{2L} \frac{\theta\pi/180}{t_b \cdot 60 \cdot t_s} \end{aligned}$$

where

- I: inertial momentum about the rotation axis of the slew manoeuvre (kg·m^2).
- θ: slew angle in degrees.
- t_b: burn time in seconds.
- t_s: manoeuvre duration in minutes.

In the case of a bias-momentum system (with momentum wheels), the torque would be

$$T = F_{MW} \cdot 2L\frac{t_b}{t_s} = H_{MW}\dot{\theta}$$

$$F_{MW} = \frac{H_{MW}\dot{\theta}}{2L}\frac{t_b}{60 \cdot t_s}$$

where

- H_{MW}‘: angular moment of the system (N·m·s).
- $\dot{\theta}$: angular rate of the momentum wheel (rad/s).
- t_b: burn time (s).
- t_s: manoeuvre duration (min).

Another important use case of thrusters is momentum detumbling for reaction wheels and momentum wheels, since the the effect of secular disturbances make reaction wheels to reach their maximum angular rate and therefore they cannot store more angular momentum.

$$F_S = \frac{H_{RW}}{2Lt_b}$$

Others

Control momentum gyros consist of a spinning rotor and one or more motorized gimbals that tilt the rotor’s angular momentum. As the rotor tilts, the changing angular momentum causes a gyroscopic torque that rotates the spacecraft. These devices are big and expensive, so they are used in big space vehicles and where they are truly necessary.

Sensors

They are used for the attitude determination part.

- **Sun sensors** can detect if the Sun is in its field of view or not, so you usually need more than one to be able to determine the attitude with respect to the Sun.
- **Earth/horizon sensors** are able to determine if the Earth (or Earth’s horizon) is in its field of view, similar to sun sensors.
- A *star tracker** will give the relative attitude to the celestial sphere. It works comparing a “picture” of the stars with a catalogue.
- **Magnetometers** measure the local magnetic field in 1 axis (or more if there are more magnetometers)
- **Gyroscopes** (as MEMS rate sensors) measure the angular rate, which can be integrated to obtain the evolution of the attitude. They are independent of the environment but tend to drift, so they must be recalibrated with the information from other sensors for an accurate measurement of the attitude. One can find MEMS rate sensors on IMUs (inertial measurements units, that usually include accelerometers, gyroscopes and magnetometers).
- **GNSS**: we can determine the attitude measuring the time difference between two antennae when communicating with a GNSS constellation (whose position is known with great precision).

Here we must highlight that a good attitude determination system uses several sensors and combines its information to reduce the uncertainties and increase the accuracy of the estimation.

3.2.7 References and Other Resources

- *[WL99]* (page 74)
- *[Wer78]* (page 74)
- *[DRDF12]* (page 74)

Online Resources

- *Attitude control* entry on Wikipedia: https://en.wikipedia.org/wiki/Attitude_control
- *Kinetics: Studying Spacecraft Motion* course in Coursera, https://www.coursera.org/learn/spacecraft-dynamics-kinetics
- *Kinematics: Describing the Motions of Spacecraft* course in Coursera, https://www.coursera.org/learn/spacecraft-dynamics-kinematics
- *Control of Nonlinear Spacecraft Attitude Motion* course in Coursera, https://www.coursera.org/learn/nonlinear-spacecraft-attitude-control
- *ADCS*, Lecture from the Satellite Engineering course at MIT in Fall 2003: https://ocw.mit.edu/courses/aeronautics-and-astronautics/16-851-satellite-engineering-fall-2003/lecture-notes/l9_acs.pdf

3.3 Communications Subsystem

The communications subsystem is responsible for **ensuring telecommunication between the satellite and another system**, which may be either another satellite or a ground station. The signals used to interchange data are nothing but electromagnetic pulses molded or manipulated by the transmitter in such a way that contains information that the receiver can understand. It provides the interface between the spacecraft and ground systems.

Payload mission data and spacecraft housekeeping data pass from the spacecraft through this subsystem to operators and users at the operations center. Operator commands also pass to the spacecraft through this subsystem to control the spacecraft and to operate the payload. As for the communication subsystem itself, it is formed by a set of antennae and transceivers to be able to communicate with the monitoring stations, sending collected data and receiving instructions from them. These instructions are processed by the control system, which could be the satellite's main computer, or certain components from the OBDH (On-board Data Handling) Subsystem. The communications subsystem receives and demodulates uplink signals and modulates and transmits downlink signals.

The subsystem also allows us to track spacecraft by retransmitting received range tones or by providing coherence between received and transmitted signals, so we can measure Doppler shift. Next three bullet points summarizes the main system considerations which drive the design of communications subsystems.

- **Access**: Ability to communicate with the spacecraft requires clear field of view to the receiving antenna and appropriate antenna gain.
- **Frequency**: Selection based on bands approved for spacecraft use by international agreement. Standard bands are S (2 GHz), X (8 GHz) and Ku (12 GHz). UHF band (0.4 GHz) is also used.
- **Baseband Data Characteristics**: Data bandwidth and allowable error rate determine RF power level for communications.

Communications access to a spacecraft requires a clear field of view for the spacecraft antenna. It also requires sufficient received power to detect the signal with acceptable error rate. Table 3.3 shows how we size the communications subsystem. To do so, we must identify the data bandwidths of the uplink and downlink, select communication frequencies, prepare RF power budgets for both links and select equipment. The basic communications subsystem consisted of a **transmitter**, a **receiver**, an **antenna** and a **RF diplexer**

Table 3.3: Steps to prelimnary size the communication subsystem

Step	Definition	What's involve
1. Identify Data Rate	Selec a Data Rate Value	Payload commands and data, Spacecraft bus commands and telemetry
2. Select frequency Bands	Decide which of the allowed bands to use	Data Rate and the proper international institution
3. Prepare RF Power Budget	Analyze characteristics of RF links	Some RF properties to define the communications subsystem
4. Select the equipment	Define equipment properties and needs	Configuration, Power subsystem and others

3.3.1 Identify Data Rate

The data rate is defined as the information flow rate, which means the information bits or bytes per unit of time that the satellite and the ground station exchange. This information may come from three different origins, on which its properties depend:

- **Command data**: All the directives from the satellite operation. Usually in a range between 2000 and 8000 bps and with a typical value of 4000 bps.
- **Health & Status telemetry**: After a concrete command, the satellite send to the ground station the conditions of all his parameters. The typical value is about 8000 bps, with a range from 40 to 10000 bps.
- **Mission/science**: All the payload and mission data rate. It depends on the purpose of the mission and on the payload itself. It be divided into Low (<32 bps), Medium (32 bps to 1 Mbps) or High Data Rate (>1 Mbps).

3.3.2 Select frequency Bands

Regulatory constrains exist on the selection of frequency band, transmission bandwidth and power flux density. International agreements have allocated frequency bands for space communications, as listed in Table 3.3.2. These agreements originated with the International Telecommunications Union (ITU) and the World Administrative Radio Conference (WARC). The system designer must apply for and receive permission from the appropriate agency to operate at a specified frequency with the specified orbit and ground locations.

Frequency band	Frequency Range (GHz)		Service
	Uplink	Downlink	
UHF	0.2 - 0.45	0.2 - 0.45	Military
L	1.635 - 1.66	1.535 - 1.56	Maritime, Telephone
S	2.65 - 2.69	2.5 - 2.54	Broadcast, Telephone
C	5.9 - 6.4	3.7 - 4.2	Domestic, Comsat
X	7.9 - 8.4	7.25 - 7.75	Military, Comsat
Ku	14.0 - 14.5	12.5 - 12.75	Domestic, Comsat
Ka	27.5 - 31.0	17.7 - 19.7	Domestic, Comsat
SHF/EHF	43.5 - 45.5	19.7 - 20.7	Military, Comsat
SHF/EHF	49	38	Internet Data, Telephone, Trunking
V	60		Satellite Crosslinks

A criterion for frequency band allocation is the potential for one link to interfere with another. Two geostationary satellites in approximately the same orbit location servicing the same ground area may share the same frequency band by: **(1)** separating adjacent satellites by an angle (typically 2 deg), which is larger than the ground station's beamwidth, and **(2)** polarizing transmitting and receiving carriers orthogonally, which allows two carriers to be received at the same frequency without significant mutual interference. Right-hand and left-hand circular polarization are orthogonal, as are horizontal and vertical linear polarization. Commercial systems use these frequency-sharing techniques extensively.

The election of the frequency also affect to the design of the communication link itself.

The bandwidth of the frequency selected determine the Data rate capacity by using the Nyquist theorem (3.5)

$$R \leq 2C \tag{3.5}$$

3.3.3 RF Link Budget

In order to prepare the RF budget, it is necessary to define all the components which may affect the RF. A definition of the unit used in the RF calculations, the decibels, can be found here: http://www.animations.physics.unsw.edu.au/jw/dB.htm. The steps to define the link budget are the following:

1. Select the satellite transmitter power, according to the size and power of other satellites.
2. Estimate RF losses between transmitter and satellite antennas $L_{TM-ant,SAT}$ (usually between -1 and -3 dB).
3. Determine the required beamwidth for the satellite antenna, depending on the satellite orbit, satellite stabilization, and ground coverage area (SMAD, Cap. 7).
4. Estimate the maximum antenna pointing offset angle, based on coverage angle, satellite stabilization error, and stationkeeping accuracy.
5. Calculate the satellite antenna transmission gain (emitting toward the ground station), G(dBi), using (3.6) (3.7) (3.8). You might also want to check the antenna diameter to see if it will fit on the satellite. A non-circular antenna has an elliptical beam with the half-power beamwidth along the major axis equal to θ_x and the half-power beamwidth along the minor axis equal to θ_y.

$$G = 44.3 - 10\log(\theta_x \theta_y) \tag{3.6}$$

   Where θ_x and θ_y are in deg.

$$L_{\theta_x} = -12(e/\theta)^2 \tag{3.7}$$

Where θ is the antenna half-power beamwidth and e is the pointing error.

$$\theta = 21/f_{GHz}D \tag{3.8}$$

Where $f_{(GHz)}$ is the carrier frequency in GHz, and D is the antenna diameter in m.

6. Calculate the space loss (3.9). This is determined by satellite orbit and ground-station location.

$$L_s = 20\log(3E8) - 20\log(4\pi) - 20\log(S) - 20\log(f) \tag{3.9}$$

7. Estimate propagation absorption loss L_a due to the atmosphere using Fig. 3.2, dividing the zenith attenuation by the sine of the minimum elevation angle (e.g. 10 deg) from the ground station to the satellite. (Consider rain attenuation later). Also add a loss of 0.3 dB to account for polarization mismatch for large ground antennas. Using a radome adds another 1 dB loss.

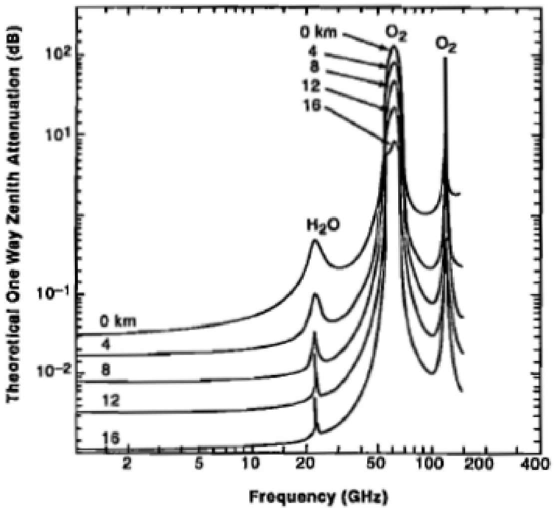

Fig. 3.2: Propagation absorption loss due to the atmosphere and the link frequencies

8. Select the ground station antenna diameter and estimate pointing error. If autotracking is used, let the pointing error be 10% of the beamwidth.
9. Calculate the ground station antenna reception gain (receiving the satellite signal). Typical values are from 0.4 to 0.7, average 0.55; but it depends of other parameters as seen in (3.6).
10. Estimate the system noise temperature (in clear weather), using Table 3.3.3.

Noise Temperature	Frequency (GHz)					
	Downlink			Crosslink	Uplink	
	0.2	2 - 12	20	60	0.2 - 20	40
Antenna Noise (K)	150	25	100	20	290	290
Line Loss (dB)	0.5	0.5	0.5	0.5	0.5	0.5
Line Loss Noise (K)	35	35	35	35	35	35
Receiver Noise Figure (dB)	0.5	1.0	3.0	5.0	3.0	4.0
Receiver Noise (K)	36	75	289	627	289	438
System Noise (K)	221	135	424	682	614	763
System Noise (dB-K)	23.4	21.3	26.3	28.3	27.9	28.8

11. Calculate E_b/N_0 for the required data rate (3.10) (3.11) (3.12) (3.13) (3.14) (i.e. for a downlink).

$$EIRP\ (dBW) = G_{TM,SAT}\ (dBW)^{*} - L_{TM-ant,SAT}\ (dB) + G_{ant,SAT}\ (dBi) \tag{3.10}$$

$$L_{Tot}\ (dB) = L_s\ (dB) + L_a\ (dB) + L_{\theta_x}\ (dB) \tag{3.11}$$

$$G/T\ (dB/K) = G_{ant,GS} - T_{sys,GS} \tag{3.12}$$

$$C/N_0\ (dBHz) = EIRP\ (dBW) + G/T\ (dB/K) - k\ (dBW/K/Hz) - L_{Tot}\ (dB) \tag{3.13}$$

Where k is the boltzmann constant.

$$E_b/N_0\ (dB) = C/N_0\ (dBHz) - R\ (dBHz) \tag{3.14}$$

Where R is the Data Rate.

- To convert dBW to dBm: $G\ (dBm) = G\ (dBW) + 30$

12. Using Fig. 3.3, look up Eb/No required to achieve desired BER for the selected modulation and coding technique.

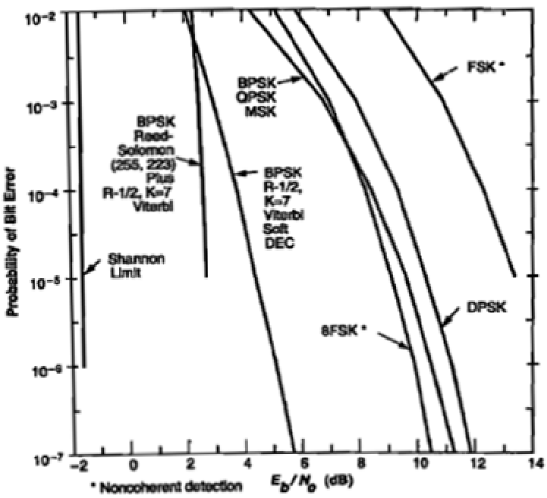

Fig. 3.3: The link BER respect to the calculated E_b/N_0

13. Add 1 to 2 dB to the theoretical value given in the last step for implementation losses.
14. Calculate the link margin, the difference between the expected value of E_b/N_0 calculated and the E_b/N_0 required (including implementation loss).
15. Adjust imput parameters until the margin is at least 3 dB greater than the estimated value, depending on confidence in the parameter estimation.

It should be highlighted that sometimes the process is the reverse one: You must find the satellite power to achieve a given data rate "R". In these cases, the equations used are the same but you must change the order of calculation.

3.3.4 Select the equipment

At the end, available equipment must be selected considering the mission requirements and the values obtained from the Link Budget. COTS are highly preferred.

3.4 Configuration and Preliminary Sizing

The configuration of a satellite refers to **the positioning of the different components** on the primary structure. Most of the mechanical requirements derived from the configuration options, which may conditionate design options as fundamental as the election of the launcher, depending on the satellite's Center of Gravity (C.o.G) or minimum size. Cubesats also have some restrictions to the C.o.G. positioning or the volume of the external components.

The first step in designing a satellite starts once the high-level requirements have been indentified and defined by the system engineer. It consists on defining, at least in a preliminary way, the mission orbit and the functions that the payload must perform to fulfill the requirements. It is useful to prepare a list of these requirements and constrains and identify those concerning the satellite configuration (i.e. the thermal requirements of a electronic equipment or the positioning requirement of a camera).

The second step is to configure a space vehicle to carry the payload equipment and provide the functions necessary for mission sucess. The design properties required from the payload are the **mass**, **size**, **power consumption** and the **field of view**. These parameters can be statistical estimate based on a collection of data from similar missions or collected for the principal mission requirements. Based on the statistical estimation, unmanned spacecraft with the same mass usually have the same energy consumption, especially with small satellites or spacecrafts with a similar mission usually have the same orientation requirements.

Before defining a preliminary configuration, it is necessary to resolve some issues related to the design process of the other subsystems involved (e.g. the satellite control method, the type of communication to be integrated or the need for a propulsion system). Once the rest of the subsystems begin to be defined, the parameters that affect the configuration subsystem can be completed, such as the mass of the components or their position requirements. For each subsystem defined in the satellite, the most critical components for the configuration must be defined first. Those that have less effect on the configuration, either by weight or size, must be defined later. Many of the necessary design parameters can not be completed in a first iteration, since the configuration of the satellite forms a closed loop with the design of the other subsystems. To define these parameters, similar missions can be used until the definition of the subsystems is more detailed. Some practical tips are sumarized in Table 3.4, including some directives in order to estimate the payload weight, power or the solar array area.

Table 3.4: Estimations and tips for the definition of the satellite configuration

Configuration Drivers	Effect	Rule of Thumb
Payload Weight	Spacecraft dry weight	Payload weight is between 17% and 50% of spacecraft dry weight. Average Is 30%
Payload Size and Shape	Spacecraft size	Spacecraft dimensions must accomodate payload dimensions
Payload Power	Spacecraft power	Spacecraft power is equal to payload power plus an allowance for the spacecraft bus and battery recharging
Spacecraft Weight	Spacecraft size	Spacecraft density will be between 20 kg/m^3 and 172 kg/m^3. Average is 79 kg/m^3
Spacecraft Power	Solar Array Area	The solar array will produce approximately 100 W/m^2 of projected area
Solar Array Area	Solar Array Type	If required solar array area is larger than area avaliable on equipment compartment, then external panels are required
Booster Diameter	Spacecraft diameter	Spacecraft diameter is generally less than the booster diameter
Pointing requirements	Spacecraft Orientation	Two axes of control are required for each article to be pointed. Attitude control of the spacecraft body provides 3 axes of control

3.5 Ground Segment

Next table (Table 3.5) summarizes the ground system design process and references discussions pertaining to each step. This process is iterative because the steps interrelate and we must strike a balance in complexity between the spacecraft and the ground system. Each iteration must address:

- Ground station locations, based on spacecraft coverage and data user needs, balanced against cost, accessibility, and available communications. You will need new sites for a dedicated ground system, and suitable existing stations when using established ground systems.
- Link data rates, which establish the required gain-to-noise temperature ratios (G/T), and effective isotropic radiated powers (EIRPs). For dedicated ground stations, defer details of antenna and RF systems until you have establisbed these two parameters. For an existing system, determine whether RF links are adequate and ajust data rates if necessary.
- Requirements for data handling, so you can determine where it will occur. Once you know the location, you can meet almost any need with avaliable hardware and software, and at reasonable cost. A decision to perform data handling at a central facility instead of the ground station influences the location of the control centers.
- Appropriate communications between ground system elements and data users, for a dedicated system. When using an existing ground system, where most decisions are already made, confirm that data handling and bandwidth are adecuate.

Table 3.5: Ground system design process and references

Steps	Description	Origin
1	Establish number and locations of ground stations	Ground segment
2	Establish space-to-ground data rates	Payload and Communication subsystem
3	Determine required G/T and EIRP	Communication subsystem
4	Determine required data handling	C&DH
5	Establish data handling location	C&DH
6	Decide location of spacecraft operation control center, payload operations control centers and mission control center	Ground segment
7	Determine and select communications links	Communication subsystem

Major Earth stations and Earth terminal complexes are sumarized in Table 3.6:

Table 3.6: Principal ground stations and Earth terminal complexes

Ground station	Location
Bukit Timah Satellite Earth Station	Singapore
Canberra Deep Space Communication Complex	Australia
Esrange Satellite Station	Sweden
Goldstone Deep Space Communications Complex	California, US
Goonhilly Satellite Earth Station	UK
Honeysuckle Creek Tracking Station	Australia
Madley Communications Centre	UK
Madrid Deep Space Communication Complex	Spain
Makarios Earth Station	Cyprus
Suparco Satellite Center	Pakistan
Svalbard Satellite Station	Norway
Kaena Point Satellite Tracking Station	Hawaii, US
The New Norcia station	Australia
Cebreros	Spain
Malargüe	Argentina
Kiruna station	Sweden
Kourou station	French Guiana
Redu station	Belgium
Santa Maria Island station	Azores, Portugal
Maspalomas Station	Gran Canaria, Spain

3.6 Launcher

A launch system consists of a basic launch vehicle incorporating one or more stages and an infrastructure for ground support. It alters velocity to place the spacecraft in orbit and protects it from the ascent surrounding. Below we discuss some of the fundamental considerations when trying to select a launch system.

- **Collect requirements and constraints of the mission:** The first step in the launch system selection process is to establish the mission needs and objectives, since they dictate the performance, trajectory, and the family of vehicles which can operate from suitable sites. Some of the requirements that must be considered in this case are related to mission orbit, mission timeline, spacecraft dimensions, spacecraft dry weight, etc.

- **Identify and collect information for the launch systems:** Include the following information for each considered launcher: performance capability to boost the necessary weight, orbit insertion, maximum satellite dimensions, reliability, cost, availability, compatibility with the satellite, etc. All these parameter can be found at the different manuals of each launcher. For some general aspects of several launchers please refer to *[WL99]* (page 74) (Section 18.2).
- **Select launch systems for spacecraft design:** During conceptual design of the satellite, identify several potential launch systems to make the launch more likely. The selected launchers should satisfy the mission performance requirements and minimize program risks. This selection shall be based on the parameters collected in the previous step.
- **Determine spacecraft design and environment dictated by the selected launch systems:** Characterize the worst-case environment for all launcher options: Maximum accelerations, vibration frequencies, temperature, orbital insertion accuracy, launcher-satellite interfaces, etc.
- **Iterate to meet constraints on performance risk and schedule:** Document and maintain the criteria, decision progress and data to support program changes.

3.6.1 Principal CubeSat Launch Services

- Albaorbital: http://www.albaorbital.com/
- Exolaunch: https://www.exolaunch.com/
- D-orbit: https://www.deorbitaldevices.com/
- GAUSS Srl: https://www.gaussteam.com/
- ISIS Space: https://www.isispace.nl/
- KSF Space: https://www.ksf.space/
- Spaceflight: https://spaceflight.com/
- TriSept Corporation: https://trisept.com/space/
- Tyvak: https://www.tyvak.com/

3.6.2 Cubesat Deployers

P-PODs (Poly-PicoSatellite Orbital Deployers) were designed with CubeSats to provide a common platform for secondary payloads.[19] P-PODs are mounted to a launch vehicle and carry CubeSats into orbit and deploy them once the proper signal is received from the launch vehicle.

The P-POD Mk III *[Uni07]* (page 74) has capacity for three 1U CubeSats, or other 0.5U, 1U, 1.5U, 2U, or 3U CubeSats combination up to a maximum volume of 3U. Other CubeSat deployers exist, with the NanoRacks CubeSat Deployer (NRCSD) on the International Space Station being the most popular method of CubeSat deployment as of 2014. Some CubeSat deployers are created by companies, such as the ISIPOD (Innovative Solutions In Space BV) or SPL (Astro und Feinwerktechnik Adlershof GmbH), while some have been created by governments or other non-profit institutions such as the X-POD (University of Toronto), T-POD (University of Tokyo), or the J-SSOD (JAXA) on the International Space

Station. While the P-POD is limited to launching a 3U CubeSat at most, the NRCSD can launch a 6U (10×10×68.1 cm) CubeSat and the ISIPOD can launch a different form of 6U CubeSat (10×22.63×34.05 cm).

Fig. 3.4: Mark II P-POD.

Source: https://en.wikipedia.org/wiki/CubeSat

3.7 Mission Analysis

In this section a general description of the spacecraft orbit design is provided. A mathematical description of keplerian orbits, simplified equations for orbital transfers as well as a general process for choosing the different orbits of the mission is stated here.

3.7.1 Keplerian orbits

Orbits represent the satellite path through space. We can completely describe the orbit of a spacecraft with five constants and one quantity which varies with time. These quantities, called classical orbital elements, are defined below and are shown in Fig. 3.5.

- a: Semimajor axis. Describe the size of the ellipse.
- e: Eccentricity. Describes the shape of the ellipse.
- i: Inclination. The angle between a reference plane (usually the Equatorial plane) and the orbital plane.
- Ω: Longitude of the ascending node. It is the angle from a reference direction (usually the Vernal Equinox), to the direction of the ascending node, measured on the reference plane.
- ω: Argument of periapsis. It is the angle from the ascending node to the periapsis, measured on the orbital plane.
- ν: True anomaly. It is the angle between the direction of periapsis and the current position of the body. This parameter defines the position of a body moving along a Keplerian orbit.

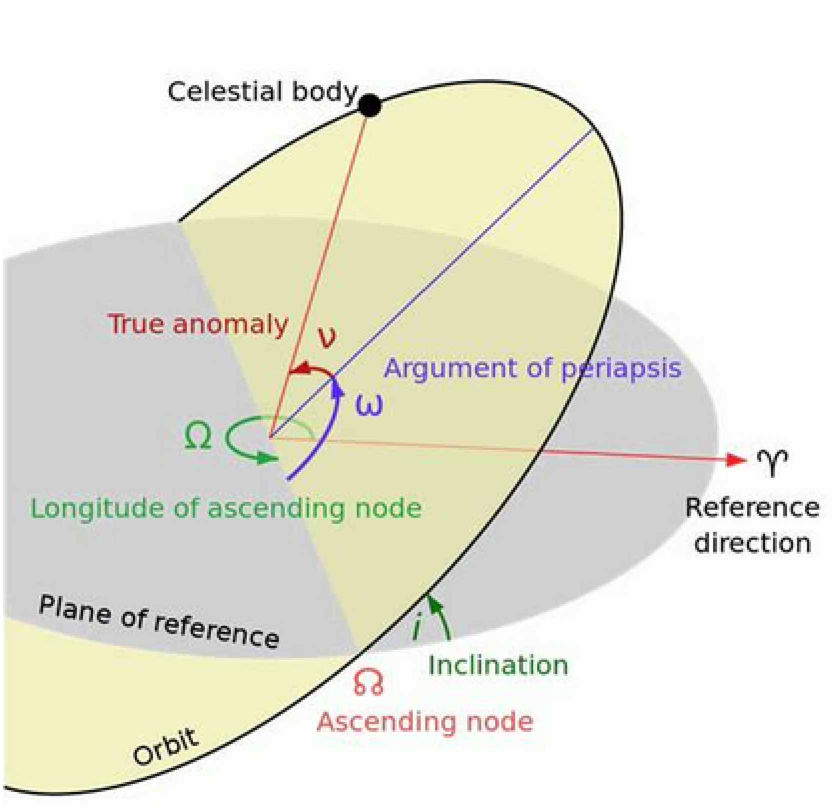

Fig. 3.5: Keplerian elements diagram. Retrieved from *https://en.wikipedia.org/wiki/Orbital_elements*.

The movement of a satellite along its orbit is due to the attraction caused by a main body located at the focus of the ellipse. The equation of the motion of a satellite position vector, $\mathbf{r}$, with respect to the planet is called the (simplified) *two-body equation of motion*:

$$\frac{d\mathbf{r}}{dt} + (\mu r^{-3})\mathbf{r} = 0, \tag{3.15}$$

where μ is the planet gravitational constant (μ_{Earth} = 398600.5 km^3/s^2). A solution of equation (3.15) gives the magnitude of the position vector in terms of the location in the orbit:

$$r = \frac{a(1-e^2)}{1+e\cdot\cos(\nu)}.$$

On the other hand, the orbital speed for an elliptical orbit can also be determined:

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}, \tag{3.16}$$

being always tangent to the satellite trajectory. For more useful equations and further details, refer to *[WL99]* (page 74) (Section 6.1).

3.7.2 The orbit design process

Orbit design has no absolute rules. The method described below, based on the one described in *[WL99]* (page 74), gives a starting point for the process for a single satellite mission.

- **Establish orbit types:** To design orbits we first divide the space mission into segments and classify each segment by its overall function. Each orbit segment has different selection criteria, so we evaluate it separately, placing it into one of the three basic types: Parking orbit, Transfer orbit and Operational orbit.
- **Establish orbit-related mission requirements:** For each mission segment, we define the orbit-related requirements. They may include orbital limits, individual requirements such as the altitude needed for specific observations or a range of values constraining any of the orbital parameters.
- **Assess specialized orbits:** In selecting the orbit for any mission phase, we must first determine if a specialized orbit applies. Specialized orbits are those with unique characteristics, such as the geostationary ring in which satellites can remain nearly stationary over a given point on the Earth's equator.
- **Mission orbit design trades:** The next step is to select the mission orbit by evaluating how orbital parameters affect each of the mission requirements. Orbit design depends mainly on altitude. Because of that, the easiest way to begin is by assuming a circular orbit and then conducting altitude and inclination trades.This process establishes a range of potential altitudes and inclinations from which we can select one or more alternatives.
- **Assess launchers and disposal options:** The launch vehicle contributes strongly to mission costs, and ultimately will limit the amount of mass that can be placed in an orbit of any given altitude. We must provide enough launch margin to allow for later changes in launch vehicles or spacecraft weight. On the other hand a disposal plan must developed to be carried out at the end of the satellite useful life so it is not hazardous to other spacecrafts.
- **Create a ΔV budget:** The total ΔV necessary for the mission is directly related to the type of orbit. It must include orbital transfer impulses (described below), the orbit maintenance as well as the ones necessary to fulfill the mission goals. To see how to create a ΔV budget refer to *[WL99]* (page 74) (Section 7.3).
- **Document and iterate:** A key component of orbit design is documenting the mission requirements used to define the orbit, the reasons for selecting the orbit, and the numerical values of the selected orbit parameters. This baseline can be reevaluated from time to time as mission conditions change.

3.7.3 Orbit maneuvering

At some point during the lifetime of most satellites, we must change one or more orbital elements. For example, we may need to transfer the spacecraft from an initial parking orbit to the final mission orbit, rendezvous with another spacecraft, or correct some orbital elements due to the orbital perturbations. To change the orbit of a satellite, we have to change the satellite velocity vector in magnitude and/or direction using a thruster. Most propulsion systems operate for only a short time compared to the orbital period, so we can treat the maneuver as an impulsive change in the velocity while the position remains fixed. For this reason, any maneuver changing the orbit of a satellite must occur at a point where the old orbit intersects the new orbit. If the two orbits do not intersect, we must use an intermediate orbit that intersects both.

In general, the change in the velocity vector to go from one orbit to another is given by

$$\Delta V = V_{NEED} - V_{CURRENT}$$

Coplanar Orbit Transfer

The most common type of in-plane maneuver changes the size and energy of the orbit, usually from a low-altitude parking orbit to a higher-altitude mission orbit such as a geosynchronous orbit. Because the initial and final orbits do not intersect the maneuver requires a transfer orbit.

One of the most common orbit transfer is called "Hohmann Transfer", shown in Fig. 3.6, and it represents the most fuel-efficient transfer between two circular coplanar orbits.

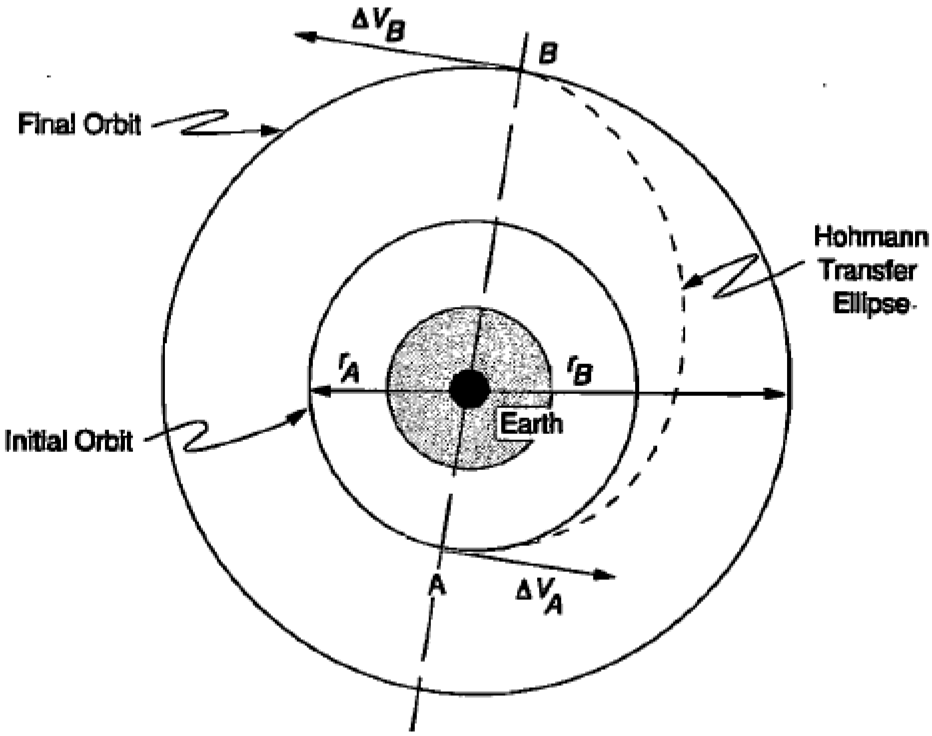

Fig. 3.6: Hohmann transfer scheme. Retrieved from *[WL99]* (page 74).

In this case, the transfer orbit ellipse is tangent to both the initial and final circular orbits at the transfer orbit perigee and apogee, respectively, so the velocity vectors are collinear at the intersection points. The total velocity change required for the transfer is the sum of the velocity changes, using equation (3.16), at perigee and apogee of the transfer ellipse:

$$\Delta V_{TOTAL} = \Delta V_A + \Delta V_B = \sqrt{\mu} \cdot \left[\left| \sqrt{\frac{2}{r_A} - \frac{1}{a_t}} - \sqrt{\frac{1}{r_A}} \right| + \left| \sqrt{\frac{2}{r_B} - \frac{1}{a_t}} - \sqrt{\frac{1}{r_B}} \right| \right],$$

where $a_t = (r_A + r_B)/2$.

Orbit plane change

To change the orientation of the satellite orbital plane, typically the inclination, we must change the direction of the velocity vector. This maneuver requires a component of ΔV to be perpendicular to the orbital plane and, therefore, perpendicular to the initial velocity vector. If the size of the orbit remains constant, we call the maneuver a "simple plane change".

The necessary ΔV to carry out a change in inclination of θ degrees is simply

$$\Delta V = 2V_i \sin(\theta/2),$$

where V_i is the velocity before and after the burn, given by equation (3.16). Because the ΔV required to change the plane is directly proportional to satellite velocity, plane changes are easier at high altitudes where the satellite velocity is lower. That is why most of the plane change in geosynchronous transfer orbit is done at apogee rather than perigee.

For further details about orbit transfers, refer to *[WL99]* (page 74) (Section 6.3).

3.8 Payload

The Payload is defined as the total complement of equipment carried by a spacecraft that interacts with the subject in performance of a particular mission. Payloads are typically unique to each mission and are the fundamental reason that the spacecraft is flown. The purpose of the rest of the spacecraft is to keep the payload working and in suitable conditions (temperature, orientation, etc).

The objective of a space mission is typically to detect, communicate or interact. The subject, as an element of the space mission, is the specific thing that the spacecraft will detect, communicate or interact with.

Spacecraft missions can serve many purposes, each one with its own set of unique requirements. Nevertheless, it is possible to classify most space missions and payloads into the following broad categories: communications, remote sensing, navigation, in situ science and other. In scientific mission, the most common type is remote sensing payloads. Any observation that a spacecraft makes without directly contacting the object in question is considered remote sensing. Fundamentally we focus on measurements in the electromagnetic spectrum to determine the nature, state, or features of some physical object or phenomenon. The electromagnetic signal can be produced by the subject or reflected and it provides information about a certain feature of such subject.

Designing a payload (or any other subsystem) consists in fixing a certain set of parameters that will define it. According to the level of detail of such design, the number of parameters will vary. These design parameters allow to estimate the main characteristics and size of the payload. These main attributes, as can be the mass, power, weight, resolution, data rate, etc, allow to compare different options. They are also forwarded to the other subsystem designers so they can take into account the presence of the payload.

3.9 Electrical Power Subsystem

The electrical power subsystem (EPS) provides, stores, distributes and control spacecraft electrical power. In order to size each component of this subsystem we must identify the electrical power loads for mission operations at the beginning-of-life, BOL, and end-of-life, EOL. For many missions, the end-of-life power demands must be reduced to compensate for solar array performance degradation. The average electrical power needed at EOL determines the size of the power source so a detailed power budget, at different stages of the mission, must be done. In Table 3.7 a sample power budget that may be used to begin the sizing process is shown but for further stages a more accurate analyses will be necessary.

Table 3.7: Typical power consumption by subsystem for small satellites (less than 100 W).

Subsystem	Consumption
Payload	20-50 W
Propulsion	0 W
Attitude Control	0 W
Communications	15 W
Data Handling	5 W
Thermal	0 W
Power	10-30 W
Structure	0 W

In addition to the values shown in the previous table, an extra margin (from 5 % to 25 %) of power might be added based on design maturity. We usually multiply average power by 2 or 3 to obtain peak power requirements for attitude control, payload, thermal, and EPS (when charging the batteries).

Once the power budget is done, we need to size the different elements of the EPS. Below, the basic steps to size these components are explained. Although these steps represent a simple process, it is enough to cover a preliminary design of the electrical power subsystem.

3.9.1 Power source

The power source generates electrical power within the spacecraft. Photovoltaic solar cells are the most common power source in most missions so we will focus on the design of this element. They convert incident solar radiation directly to electrical energy and are famous for being well-known and reliable.

The starting point of the photovoltaic solar cells design is defining the mission life and the average power requirement. The first one is often known before starting the preliminary design of any subsystem and the second one must be determined from the power budget, given in Table 3.7. We size a photovoltaic system to meet power requirements at EOL, with the resulting solar array often oversized for power requirements at BOL. This excess power at BOL requires coordinated systems engineering to avoid thermal problems. The longer the mission life, the larger the difference between power requirements at EOL and BOL.

Once these two key parameters have been established, the total power that the solar array must provide during daylight (P_{sa}) can be determined using

$$P_{sa} = \frac{\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}}{T_d}$$

where P_e and P_d are the spacecraft's power requirements (excluding regulation and battery charging losses) during eclipse and daylight, respectively, and T_e and T_d are the lengths of these periods per orbit. X_e and X_d the efficiency of the paths from the battery to the individual loads and the path directly from the arrays to the loads, respectively. The previous formula shows the total power that must be provided for one orbit to fulfill the spacecraft power requirements. However, nothing has been said until now about how to determine the total power that the solar arrays really provide.

There are three many types of cells (Sillicon, Gallium Arsenide, Multijuction, etc.) and each one of them has an efficiency (η) and degradation coefficients. Gallium arsenide has the advantage of higher efficiencies, whereas indium phosphide reduces the degrading effects of radiation. Silicon solar cell technology is mature and has the advantage of lower cost per watt for most applications. Gallium arsenide and indium phosphide cost about 3 times more than silicon. In Table 3.8 the performance of several solar cells are shown. For a more detailed analysis, please refer to the datasheet of each particular case.

Table 3.8: Performance comparison for photovoltaic solar cells.

Cell type	Silicone	Gallium Arsenide	Indium Phosphide	Multijunction
Theoretical efficiency	20.8 %	23.5 %	22.8 %	25.8 %
Achieved (production)	14.8 %	18.5 %	18 %	22 %

Next, we must determine the resultant power production capability of the manufactured solar array. An assembled solar array is less efficient than single cells due to design inefficiencies, shadowing and temperature variations, collectively referred as inherent degradation, I_d. A typical value for this coefficient is 0.77 although it might vary from 0.49 to 0.88.

Once these two parameters (efficiency and inherent degradation) have been defined, we can compute the solar array power at the beginning of life, per unit of area,

$$P_{BOL} = P_{SUN} \eta I_d \cos(\theta),$$

where P_{SUN} is the solar illumination intensity (1367 W/m^2), which depends on the distance Sun-satellite, and $\cos(\theta)$ is referred as the cosine loss. We measure the Sun incidence angle θ between the vector normal to the surface of the array and the Sun line. So if the Sun rays are perpendicular to the solar array surface, we get maximum power. Obviously, the geometry between the array and the Sun changes throughout the mission and different solar array panels will have different geometry. The worst-case Sun incidence angle is used for our calculations.

Next, we must consider the factors that degrade the solar array's performance during the mission. Life degradation, L_d occurs because of thermal cycling in and out of eclipses, micrometeoroid strikes, plume impingement from thrusters, and material outgassing for the duration of the mission. In general, for a silicon solar array in LEO, the degradation is about 3.75 % per year. For gallium-arsenide cells in LEO, power production can decrease by as much as 2.75 % per year. The actual lifetime degradation can be estimated using

$$L_d = (1 - c)^{t_f},$$

where c is the degradation per year and t_f is the satellite lifetime. The array performance per unit of area at the end of life is

$$P_{EOL} = P_{BOL} L_d.$$

Finally, the solar array area, A_{sa} required to support the spacecraft power requirements is given by

$$A_{sa} = \frac{P_{sa}}{P_{EOL}}$$

Solar-array sizing is more difficult than it appears from the above discussion. Typically, we must consider several arrays with varying geometry. Also, the angle of incidence on the array surface is constantly changing. We must predict that angle continuously or at least determine the worst-case angle to develop an estimate of P_{EOL}.

Energy Storage

Any spacecraft that uses photovoltaics or solar thermal dynamics as a power source requires a system to store energy for peak-power demands and eclipse periods. Energy storage typically occurs in a battery, although systems such as flywheels and fuel cells have been considered in some particular cases.

A battery consists of individual cells connected in series. The number of cells required is determined by the bus-voltage. The amount of energy stored within the battery is the ampere-hour capacity or watt-hour (ampere-hour times operating voltage) capacity. The design capacity of the battery derives from the energy storage requirements. Batteries can be connected in series to increase the voltage or in parallel to increase the current.

One of the most important parameters which characterize a discharge period is known as the Depth of Discharge (DoD). It is simply the percentage of total battery capacity consumed during a discharge period. Once we know the average depth of discharge, determining the different currents that each load needs, we can determine the total capacity of the batteries (C_r) by using the following expression

$$C_r = \frac{P_e \cdot T_e}{DoD \cdot N \cdot X_e},$$

where N is the number of non-redundant batteries. To obtain the battery capacity in ampere-hour, the previous result must be divided by the bus voltage (usually 28 V).

Once the total capacity has been determined, finding a suitable battery or batteries is the task of the electrical power engineer.

Power distribution

A spacecraft power distribution system consists of cabling, fault protection, and switching gear to turn power on and off to the spacecraft loads. Power distribution designs for various power systems depend on source characteristics, load requirements and subsystem functions. In selecting a type of power distribution, we focus on keeping power losses and mass at a minimum while attending to survivability, cost, reliability, and power quality.

In this section no detailed analysis will be described. We only insist on the fact that we must account for the cable and harness mass when designing the EPS. Operating low current (less than 30 A) devices helps to keep this mass low. In Fig. 3.7 , an estimation of the mass of wire per meter with respect to the current is shown.

The harness or cabling that interconnects the spacecraft subsystems is a large part (10-25 %) of the electrical power system mass. We must keep harness as short as possible to reduce voltage drops and to keep the total spacecraft mass low.

3.10 Propulsion

3.10.1 Introduction

Space propulsion is used to three things: lift the launch vehicle from the surface into low-Earth orbit (LEO), transfer payloads from LEO to higher orbits or interplanetary trajectories and provide attitude control and orbit corrections.

In this section, we are going to cover how to choose a thruster for a satellite given some performance indicators like the Delta-V. This guide is oriented towards the use of commercial off-the-shelf components instead of being a preliminary design guide of chemical or electrical thrusters (which is a very complex science), and therefore will not cover launch propulsion systems.

The most important data that the engineer in charge of the propulsion subsystem should provide to the design team at the end of each iteration are:

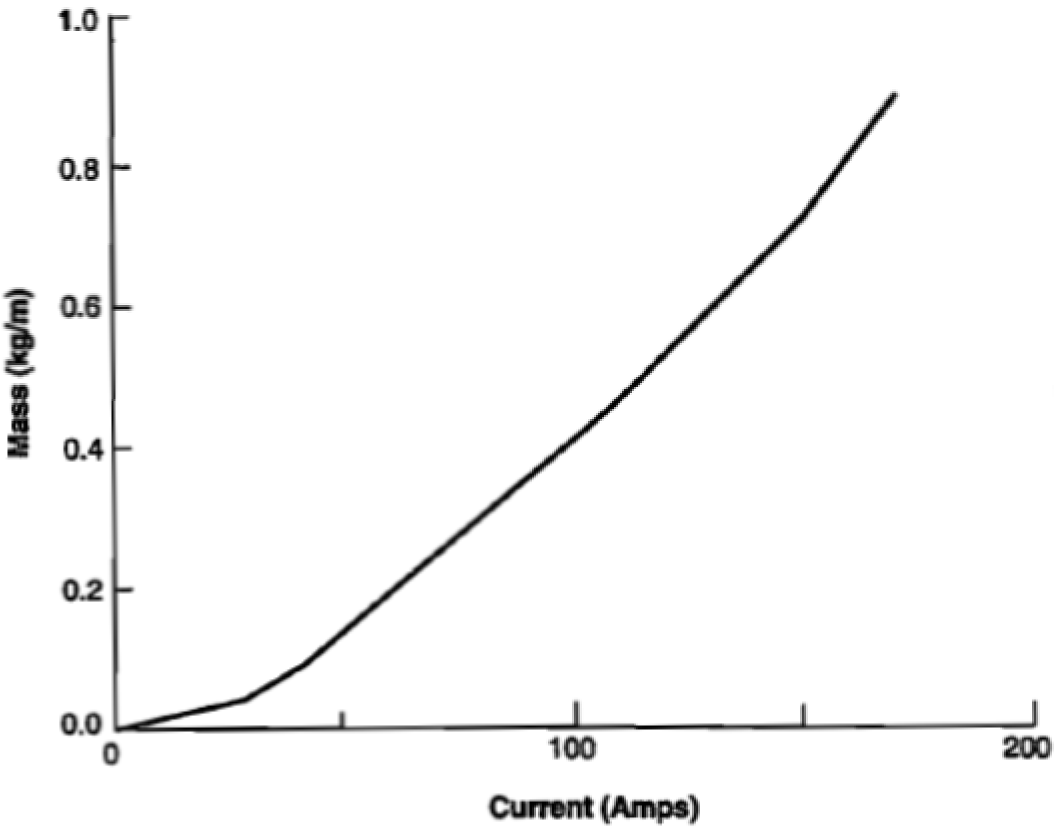

Fig. 3.7: Cable mass per meter over current. Retrieved from *[WL99]* (page 74) (Section 11.4.3).

- Propellant budget, detailing how much mass is reserved for orbit transfers, orbit corrections and attitude control
- Mass of the thruster(s)
- Power consumed by the thruster in stand-by and in service
- Total size of the thruster(s)

Other components that are part of the propulsion subsystem are the tanks and the lines and pressure-regulating equipment that connects the tank to the thrusters. In this guide, we are going to suppose that the thruster is a component off-the-shelf and, therefore, we are not going to design that connecting system.

The propulsion subsystem has two purposes: translational control manoeuvres and attitude control manoeuvres. Depending on its purposes, its requirements and location on the spacecraft are different. For example, a translational control thruster should be aligned with the centre of mass, whereas an attitude control one should be as far as possible from it to increase the lever arm and therefore reduce the required thrust.

In terms of the impact of this subsystem to the rest of the vehicle, its power consumption is low unless it requires heated propellant or it is an electric thruster. Regarding the mass budget, its impact is big since the propellant must be burn and expelled to create thrust. Finally, thermally speaking, it depends on the type of thruster but in principle we can say that at least we should prevent the propellant and the lines from freezing, so heaters may be required.

In Table 3.9 are summarised the principal propulsion technologies and where they can be applied.

Table 3.9: Principal options for spacecraft propulsion systems, from [WL99]. P: perigee, A: apogee

Propulsion technology	Orbit insertion	Orbit Maintenance and Manoeuvring	Attitude Control	Typical Isp (s)
Cold gas	PA	X	X	30-70
Solid	PA			280-300
Liquid: monopropellant	PA	X	X	220-240
Liquid: bipropellant	PA	X	X	305-310
Liquid: dual mode	PA	X	X	313-322
Hybrid	PA	X		250-340
Electric	A	X		300-3000

3.10.2 Design Process

1. List applicable spacecraft propulsion functions, like orbit insertion, orbit maintenance, attitude control and controlled de-orbit or reentry.
2. Determine ΔV budget and thrust level constraints for orbit insertion and maintenance.
3. Determine total impulse for attitude control, thrust levels for control authority, duty cycles (% on/off, total number of cycles) and mission life requirements.
4. Determine propulsion system options: combined or separate propulsion systems for orbit and attitude control, high vs. low thrust, liquid vs. solid vs. electric propulsion technology.
5. Estimate key parameters for each option: effective specific impulse for orbit and attitude control, propellant mass, propellant and pressurant volume, configure the subsystem and create equipment list.
6. Estime the total mass and power for each option.
7. Establish baseline propulsion system.
8. Document results and iterate as required.

3.10.3 The Rocket Equation

Also known as Tsiolkovsky equation (3.17), it is the most important equation in mission analysis and therefore it is important too for the propulsion subsystem: given a required ΔV (Delta-V, m/s) we can estimate the required propellant mass if we know the specific impulse I_{sp} (s), which is the most characteristic parameter of a thruster.

$$\Delta V = I_{sp} g_0 \log \frac{M_0}{M_f} \tag{3.17}$$

In Equation (3.17) it is assumed that the manoeuvre is applied instantaneously, so it is only valid for short bursts. M_0 and M_f are the pre- and post-manoeuvre masses (kg) of the vehicle, and g_0 is 9.81 m/s^2. From this equation, we can compute the required propellant mass for our ΔV, since that would be the change in the total mass of the vehicle:

$$\Delta m_p = M_0 \left(1 - e^{-\Delta V / I_{sp} g_0}\right)$$

3.10.4 Types of Rockets

This is a brief description of the types of rockets or propulsion systems that we saw in Table 3.9. The principal propulsion technologies are: *cold gas*, *chemical* and *electrical*.

Cold gas

Cold gas propulsion is just a controlled, pressurised gas source and a nozzle. There is no combustion and there is only a gas expelled. Its the simplest form of a rocket engine.

Chemical

Chemical thrusters work redirecting the resultant gases from the combustion of the propellants. Depending on the nature of the propellants, we can have liquid or solid chemical propulsion, with important changes in the internal architecture of the thrusters. The internal architecture of the thrusters is a topic that is not going to be covered due to its complexity and lack of interest when using COTS.

The terminology *solid*, *liquid* or *hybrid* refers to the initial state of the stored propellant.

Liquid rocket systems use liquid propellants that are fed to the combustion chamber by gas pressurisation or a pump. Depending on the number of components, these rockets are called *bipropellant* if they use two propellants or *monopropellant* if only one.

On the one hand, bipropellant systems are more complex since they usually have a fuel and an oxidiser that chemically react in a combustion process, but they can provide a higher specific impulse. On the other hand, monopropellant systems are simpler and therefore more reliable; the propellant that they use can ignite and provide the required energy without more components. A very common monopropellant is hydrazine, since it is stable under normal storage conditions, has a clean decomposition process and is very easy to handle, although it must be done with care due to its toxicity. It is the most common type of propulsion for spacecraft attitude and velocity control.

Hybrid systems have the propellants in different states, usually the fuel is a solid and the oxidiser is a gas or a liquid. They are not very common.

Finally, **solid rockets** have their propellant in solid form. They have lower performance than liquid rockets but they are simpler. Their main limitation is that, once they have been ignited, they cannot be shut down, that is to say, they can only be used once.

Electric

Electric propulsion accelerates the working fluid using electrical power obtained from the Sun as solar energy, nuclear or thermal. This acceleration is the responsible of the creation of thrust once the fluid is expelled. There are different ways to use electric power to create this thrust, and depending on the way we can establish three main classes of electric propulsion systems: *electrothermal*, *electrostatic* and *electromagnetic*.

Since these propulsion systems rely heavily on electric power and propellant mass, it is important to measure their efficiency to make a good comparison. The relationship between the power P (W), the thrust F (N), the specific impulse I_{sp}, the mass flow rate $\dot{m}$ (kg/s) and the efficiency η is:

$$P = \frac{F^2}{2\dot{m}\eta} = \frac{F I_{sp} g}{2\eta}$$

Table 3.10: Main characteristics of electric propulsion systems

Class	Name	Propellant	Power	Specific impulse (s)	Efficiency	Thrust
Electrothermal	Resistojet	Hydrazine, ammoniac	0.5-1.5 kW	300	80%	0.1-0.5N
Electrothermal	Arcjet	Hydrazine, hydrogen	0.3-100 kW	500-2,000	35%	0.2-2N
Electrostatic	Ion	Xenon	0.5-2.5 kW	3,000	60%-80%	10-200mN
Electrostatic	Hall	Xenon	1.5-5kW	1,500-2,000	50%	80-200mN
Electrostatic	FEED	Indium, caesium	10-150W	6,000-10,000	30%-90%	0.001-2mN
Electrostatic	Coloidal	Glycerine	5-50 W	500-1,500	(-)	0.001-1mN
Electromagnetic	PPT	Teflón	1-200 W	1,000	5%	1-100mN
Electromagnetic	MPD	Hydrogen, ammoniac, lithium	1-4000kW	2,000-5,000	25%	1-200N
Electromagnetic	VASIMR	Hydrogen	1-10 MW	3,000-30,000	20%-60%	1-2kN

3.10.5 References and Other Resources

- *[WL99]* (page 74)
- *[WEP11]* (page 74)
- *[SB16]* (page 74)
- *[MB14]* (page 74)

3.11 Structure

The spacecraft structure carries and protects the spacecraft and payload equipment through the launch environment and during its lifetime from external conditions. The load-carrying structure of a spacecraft is *primary structure*, whereas brackets, closeout panels, and most deployable components are *secondary structure*.

We size primary structure based on the launch loads, with strength and stiffness dominating its design. The size of secondary structure depends on on-orbit factors rather than boost-phase loads. Secondary structure only has to survive but not function during boost, and we can usually cage and protect deployables throughout this phase. As a reference value, the structure of a satellite usually represents the 20% of the spacecraft total mass. Although we should also add approximately 25% for weight growth to account for program additions, underestimating, and inadequate understanding of requirements.

3.11.1 Design Philosophy

To develop a structure light enough for flight, and to keep spacecraft affordable, we must accept some risk of failure. Material strengths vary because of random, undetectable flaws and process variations, and loads depend on unpredictable environments. Because of loads uncertainty, we cannot accurately quantify the structural reliability of a spacecraft. We can approximate it, however, and we can develop design criteria that will provide acceptable reliability.

- Use a design-allowable load (the highest load that a structure or material can withstand without failure) for the selected material that we expect 99% of all specimens will equal or exceed.
- If possible, from available environmental data from previous missions, derive a design limit load (the maximum load expecting during the mission) equal to the mean value plus three standard deviations. This means there will be 99.87% probability that the limit load will not be exceeded during the mission.
- Multiply the design limit load by a factor of safety, then show that the stress level at this load does not exceed the corresponding allowable load.

Nowadays, resorting to prefabricated structures when designing nanosatellites or cubesatellites is the common practice. There are several possibilities in the industry and each one of them shall be analyzed to determine whether it is suitable for your mission. Preliminary analyses might be done using the formulas described below and combining some FEM model to ensure the proper functioning of the satellite until its end-of-life.

3.11.2 Design options

When designing a structure, we consider several materials, types of structure, and methods of construction. To select from these options, we compare each option analyzing its weight, cost, and risk.

A typical spacecraft structure contains metallic and nonmetallic materials. By far the most commonly used metal for spacecraft structure is aluminum alloy, of which there are many types and tempers. Aluminum is relatively lightweight, strong, readily available, easy to machine, and low in raw material cost. Nonmetals are usually formed with composites. Although aluminum and composites are widely used in satellite structures there are a lot of different options that shall be considered. The main advantages and disadvantages of some of them are summarized in Table 3.11.2.

Material	Advantages	Disadvantages
Aluminum	• High strength vs weight • Ductile and low density • Easy to machine	• Relatively low strenh vs volume • Low hardness • High coefficient of themal expansion
Steel	• High strength • Wide range of strength, hardness and ductility	• High density • Hard to machine • Magnetic
Magnesium	• Low density	• Susceptible to corrosion • Low stregth vs volume
Titanium	• High strength vs weight • Low coefficient of thermal expansion	• Hard to machine • Poor fracture toughness if treated and aged
Beryllium	• High stiffness vs density	• Low ductile and fracture toughness • Low short transverse properties • Toxic
Composite	• High stiffness, strength and low thermal expansion coefficient • Low density • Good in tension	• Expensive for low production volume • Brittle • Dependant strength; usually requires proof testing

3.11.3 Preliminary design

The launch vehicle is the most obvious source of structural requirements, dictating the spacecraft weight, geometry, rigidity, and strength. Each of the launch boosters provides maximum acceleration levels to be used for design. These acceleration levels or load factors are typically 6 g's maximum axial acceleration and 3 g's maximum lateral acceleration. The actual force is obtained multiplying these load factors (LF) by the satellite weight:

$$P_{ax} = mgLF_{ax}; \quad P_{lat} = mgLF_{lat}.$$

To approximate the required thickness of the structure we can resort to simplified geometries and formulas. Regardless of the actual shape of the satellite, we can suppose it to be a hollow cantilevered cylinder with uniform thickness, t, and height, L. The transversal area, A, and its moment of inertia, I, are related to its internal radius, r, and the mentioned thickness:

$$A = 2\pi rt, \quad I = \pi r^3 t.$$

Supposing that the center of mass location is at the cylinder mid-length ($L/2$), we can find the equivalent axial load taking into account that the lateral acceleration will produce a momentum on the base of the structure (in the interface with the launcher). This momentum will increase the axial equivalent force:

$$P_{eq} = P_{axial} + \frac{2M}{r} = P_{axial} + \frac{2P_{lat}L}{2r}$$

Now the structure shall be checked for ultimate and yield conditions. P_{eq} is multiplied by a factor of safety, f, which has a value of 1.25 for ultimate conditions and 1.1 for yield and then compared with the material allowable stress. The thickness must be capable of resisting the axial loads:

$$\frac{P_{eq}f_u}{A} < \sigma_u; \quad \frac{P_{eq}f_y}{A} < \sigma_y,$$

where σ_u and σ_y represent the material allowable stress for ultimate and yield conditions, respectively. To see different materials σ_u and σ_y refer to Table 11-52 in *[WL99]* (page 74).

Furthermore, a launch-vehicle structure has certain natural frequencies that respond to forces from both internal (engine oscillations) and external (aerodynamic effects) sources. The launch vehicle contractor lists known natural frequencies for each launch vehicle. The spacecraft structure tailored to avoid the launch vehicle natural frequencies will experience much lower loads and this is one of the most important requirements when designing the spacecraft structure.

With the launcher frequencies (axial and lateral) known, we can find a value of the thickness for which the natural frequencies of the structure will be larger than the launcher frequencies. For axial natural frequency we have:

$$f_{ax} = \frac{1}{2\pi}\sqrt{\frac{EA}{mL}} = \frac{1}{2\pi}\sqrt{\frac{2E\pi rt}{mL}},$$

and for the lateral natural frequency

$$f_{lat} = \frac{1}{2\pi}\sqrt{\frac{3EI}{mL}} = \frac{1}{2\pi}\sqrt{\frac{3E\pi r^3 t}{mL}}.$$

The final thickness will be the more conservative value of the ones obtained before (combining the values obtained in the equivalent loads and frequency analyses).

Although the previous formulas can provide an approximated value for the structure thickness, the actual analysis of a spacecraft structure is much more complex. It possesses a lot of internal elements which will affect to the natural frequencies of the structure. Moreover, there are more requirements to meet caused by different perturbation sources. For example, random vibration from engines and other sources is a critical source of load as well. It must be proved that the satellite is capable of resisting these kind of loads (and any other). In actual design, this is proved by means of some numerical result using FEM tools and then checked during the test campaign.

3.12 Systems Engineering

3.12.1 Budgets

Once the requirements have been defined, the next step is the allocation of resources to each subsystem. In other words, we must decide exactly how much of the spacecraft resources will need each subsystem to do its job. This allocation will depend on the subsystem requirements and will be an iterative process. The initial estimates are provided by the system engineer and are based on experience on previous missions. Subsystem designers will use these estimates as a starting point and will provide more accurate estimations as the design progresses.

The fundamental resource of a spacecraft is mass. Because of the high cost of launch vehicles and the step function in cost when you outgrow a vehicle, the system design must stay within established mass limits. Power becomes another limited resource once you choose the solar array size (a choice that must generally occur early in the development). There systems engineers can track multiple resources budgets although it may not be necessary to track them all.

Margins

In order to ensure that the system stays within the mass capability of the launch vehicle, we set aside margin, which is extra mass that is not assigned to any particular subsystem. We also carry margin for all the other spacecraft resources, like power, processor utilization, and memory. One of the biggest challenges for a spacecraft systems engineer is properly managing margin. If you hold too much margin, the subsystem designs become overly constrained and the cost of the system rises, or you lose out on capability you could have had in your system. If you hold too little margin, you will find yourself with an assembled spacecraft that doesn't work (it is too heavy for the launch vehicle, its solar arrays don't generate enough power to support its function, there is insufficient fuel, or some other such crisis). The consequences of exceeding capability are severe. Margin covers the uncertainty of the design, so you can reduce it as the design matures.

On top of the allocation, system engineers carry addition system margin to handle unforeseen situations.

Mass

To create a first mass budget:

1. Start with the target launch mass
2. Subtract the propellant mass
3. Determine a reasonable allocation for the payload. This allocation must include a margin for contingencies that will depend on how new is the design of the payload
4. Set aside system margin
5. Allocate mass for each subsystem based on previous missions
6. Iterate. Subsystem designers provide new estimations of the subsystem mass based on their calculations. They must aim at the target mass established in the budget for their subsystem
7. Adjust the budget as appropriate

In Table 3.11, the average mass by subsystem for 4 types of spacecrafts is listed. It has been extracted from *[WL99]* (page 74), consult the appendix A of the reference for more information.

Table 3.11: Typical percentages per subsystem in a mass budget.

Subsystem (% of Dry Mass)	No Prop	LEO with Prop	High Earth	Planetary
Payload	41%	31%	32%	15%
Structure and Mechanism	20%	27%	24%	25%
Thermal Control	2%	2%	4%	6%
Power (incl. harness)	19%	21%	17%	21%
TT&C	2%	2%	4%	7%
On-Board Processing	5%	5%	3%	4%
Attitude Determination and Control	8%	6%	6%	6%
Propulsion	0%	3%	7%	13%
Other (balance + launch)	3%	3%	3%	3%
Total	100%	100%	100%	100%
Propellant	0%	27%	72%	110%

Propellant budget

The propellant budget follows directly from the ΔV budget. Using the rocket equation

$$\Delta V = I_{sp} ln \frac{M_0}{M_f}$$

we can calculate the propellant mass necessary for a maneuver given the ΔV and the I_{sp} of such maneuver.

$$M_p = M_f (\exp^{\Delta V/(I_{sp} g_0)} - 1)$$

where M_f is the spacecraft mass after the maneuver. Using this equation for each maneuver, the propellant budget is built.

A typical propellant budget contains four elements:

1. Velocity-control propellant
2. Attitude-control propellant
3. Margin (a percentage of the total)
4. Residual (unavailable propellant)

Power budget

The initial estimate for the solar array size should be based on an estimate of the power consumed by the spacecraft. Just like with mass, you should itemize the subsystems and eventually the individual components in a subsystem and add up the total power consumption based on type of spacecraft and on historical data. Some rule of thumb on power consumption:

- The faster a data system operates, the more power it will consumed
- Communications systems trade power for antenna size
- Generally, the spacecraft will need more heater power if the environment has large variations

The estimation of the spacecraft power requirements is carried out in three steps:

1. Preparation of an initial operatin power budget by estimating the power required by the payload an the spacecraft subsystems. In case the spacecraft has several operating modes that differ in powe requirements, we must budget separately for each mode (survival mode, nominal mode, science mode, etc)
2. Selecting the battery capacity appropriate to the spacecraft power requirements and battery cycle life.
3. Accounting of the power-subsystem degradation over the mission life by computing radiation damage to the solar array.

In Table 3.12, it is listed the average power by subsystem for 4 types of spacecrafts. It has been extracted from *[WL99]* (page 74), consult the appendix A of the reference for more information.

Table 3.12: Typical percentages per subsystem in a power budget.

Subsystem (% of Total power)	No Prop	LEO with Prop	High Earth	Planetary
Payload	43%	48%	35%	22%
Structure and Mechanism	0%	1%	0%	1%
Thermal Control	5%	10%	14%	15%
Power (incl. harness)	10%	9%	7%	10%
TT&C	13%	12%	16%	18%
On-Board Processing	13%	12%	10%	11%
Attitude Determination and Control	18%	10%	16%	12%
Propulsion	0%	0%	2%	11%

3.12.2 References and Other Resources

- *[WL99]* (page 74)
- *[FSS11]* (page 74)
- *[MB14]* (page 74)
- [esa2017margins]

3.13 Thermal Control

3.13.1 Introduction

The role of thermal control subsystem is to maintain all spacecraft and payload components and subsystems within their required temperature limits for each mission phase. Temperature limits include a cold temperature which the component must not go below and a hot temperature that it must not exceed. Two limits are frequently defined: operational limits that the component must remain within while operating and survival limits that the component must remain within at all times, even not powered. Exceeding survival temperature limits can result in permanent equipment damage as opposed to out-of-tolerance performance when operational limits are exceeded.

Thermal control techniques are broadly divided into two categories. Passive thermal control makes use of materials, coatings, or surface finishes (such as blankets or second surface mirrors) to maintain temperature limits. Active thermal control, which is generally more complex and expensive, maintains the temperature by some active means, suah as heaters or thermo-electric cooleers.

The desired temperatures are achieved by balancing the flow of heat energy across spacecraft interfaces. The power dissipation of the system combined with the orbit's thermal environment will drive the thermal design. The system must conduct internally-generated heat to the outside of the spacecraft where it can be radiated to space. All spacecraft components are designed to operate over a defined temperature range. While most components operate at or near room temperatures, others have narrow temperature requirements which demand operation over a wide temperature range.

Therefore, to begin the design process, we need to know the temperature requirements of the components, the power dissipated by them and the heating environment to which the spacecraft will be exposed.

The spacecraft and its elements are required to perform various functions during the mission. These can include pointing instruments, collecting energy from solar arrays, or communicating with ground stations. These functional requirements usually dictate a general configuration and a spacecraft location and orientation in space that will enable the functions. The physical configuration, its locations, and orientation will define the radiant energy that falls on the spacecraft surfaces. This energy can be absorbed or reflected as necessary to control the flow of heat into the spacecraft and set the heat balance to achieve the required temperatures. The balance of the heat in, out through the spacecraft sets temperatures to meet requirements during the mission.

Ideally, the radiators will face deep space, with its 3 K background temperature. For low-Earth orbit, the Earth is about 300 K and fills half the sky, making it more difficult to find room on the spacecraft for radiators that need to run below this temperature. Since the Sun is only a half-degree across from Earth distance and it is surrounded by deep space, radiators can face it, but they need white or shiny coatings which reflect optical energies and still radiate in the infrared. Smaller spacecraft generally have plenty of surface area to radiate the heat that can be generated by the few components that fit in their small volume. On larger spacecraft, it can be a challenge to move heat from its source out of a radiator.

Example of typical spacecraft element operating temperature requirements are shown in the table below. Components that are sensitive to temperature, (e.g., batteries, star trackers) can be located in positions on the spacecraft that provide thermal stability and heaters can be allocated to provide additional control. Components that must be exposed directly to the space environment to perform their function (antennas, solar arrays) have to be designed to withstand a very wide range of temperatures.

Table 3.13: Typical temperature ranges for each type of component.

Component	Typical Temperature Ranges (ºC)	
	Operational	Survival
Batteries	0 to 15	-10 to 25
Power Box Baseplates	-10 to 50	-20 to 60
Reaction Wheels	-10 to 40	-20 to 50
Gyros/IMUs	0 to 40	-10 to 50
Star Trackers	0 to 30	-10 to 40
C&DH Box Baseplates	-20 to 60	-40 to 75
Hydrazine Tanks and Lines	15 to 40	5 to 50
Antenna Gimbals	-40 to 80	-50 to 90
Antennas	-100 to 100	-120 to 120
Solar Panels	-150 to 110	-200 to 130

Radiant Energy Heat Balance in Spacecraft

The external surfaces of a spacecraft are the boundary conditions which ultimately provide the energy balance with the environment and therefore control the average temperature. The thermal properties of these surfaces are chosen to balance the energy transfer over the required spacecraft temperature. This is possible because the radiant energy absorbed from the environment is independent of the surface temperature, but the energy radiated from the surface is a strong function of surface temperature (T^4).

Example

Consider a simple flat plate with one side normal to the Sun at 1 AU, and the other side insulated (adiabatic). We can compute the temperature of this object by writing the heat energy balance and solving for T. At equilibrium the heat energy in, Q_{in}, equals the heat energy out, Q_{out}:

$$Q_{in} = Q_{out}$$
$$Q_{in} = SA_p\alpha$$
$$Q_{out} = A_r\varepsilon\sigma T^4$$

where S is the solar constant (1366 W/m:math:^2 at 1 AU), A_p is the projected area of the plate toward the Sun, α is the absortivity of the surface, A_r is the radiating surface area of the plate, ε is the emissivity of the surface, σ is the Stefan-Boltzmann constant (5,67051 x 10:$^-$8 W/m:math:^2*'K:math:'^4*), and T is the absolute temperature in Kelvin. Therefore:

$$T = \left(\frac{S}{\sigma}\left(\frac{\alpha}{\varepsilon}\right)\left(\frac{A_p}{A_r}\right)\right)^{1/4}$$

For the insulated flat plate geometry, the projected area, A_p equals the radiating area, A_r.

Sources of Radiant Energy in Spacecraft

- Direct solar energy flux is a function of the distance from the Sun. The radiant energy from the Sun is usually the most significant heat source in spacecraft thermal control. The Sun radiates its energy equally in all directions. However, from the Earth, the Sun appears as a small disc subtending an angle of only ½ deg. This geometry causes the Suns energy to impinge on a surface normal to the Sun as parallel rays. There are two important characteristics of the Sun's radiant energy for spacecraft thermal control. These are the total energy density, or flux, incident on a surface and the spectral content of that flux. The flux at 1 AU is called the solar constant and is calculated at the Earth's average distance from the Sun. As we change distances from the Sun the flux changes as the square of the distance. For deep space missions, or missions to other planets, the solar flux can vary dramatically. The spectral content of the solar flux emulates the black body (perfect emitter) distribution for a source at 5780 K. This concentrates the energy in the visible wavelengths with significant amounts near visible in the ultraviolet and infrared wavelengths. This concentration of energy at certain wavelengths allows the selective absorption of the energy (i.e., surface properties).
- Reflected Sun (albedo) is a function of orbit and orientation. The solar energy reflected by a planet, albedo, is also a significant source of radiant energy when the spacecraft is near the planet. The Earth diffusely reflects from 25% to 55% of the incident solar energy depending on the Earth surface properties (land, sea, clouds and ice all have different reflectance). For higher orbits, the Earth begins to fill less the field of view of the satellite until the albedo at geosynchronous orbits is negligible. The spectral content of the reflected solar energy is approximately the same as energy directly from the Sun. However, it arrives as diffuse energy and not as parallel rays directly from the Sun. Computing the albedo incident on the various surfaces requires knowledge of spacecraft

orientation and location relative to a sunlit Earth. Surfaces facing away from the Earth will receive no solar energy reflected from Earth.

- Planetary infrared energy is a function of the planet's temperature and the spacecraft's orientation to the planet. Planets, moons and other large space bodies can also provide infrared heat sources to spacecraft. These sources are warm compared to space because they have absorbed energy from the Sun and re-radiate it as infrared energy. Most of these sources can be treated as uniform in temperature if they are rotating in the sunlight or have an atmosphere to spread the heat over the surface of the body. The Moon is a notable exception since it does not have an atmosphere and takes 28 Earth days for one revolution. Lunar designs must consider the wide temperature variation on the Moon's surface.

Absorbed Energy; Surface Properties

The quantity of heat incident on a surface depends on the temperature of the source, and the geometric relationship between the surface and the source. The incident energy on each surface can be absorbed or reflected by the surface. The surface properties that define the absorbed energy are the absorptance, (α for solar energy) and emissivity (ε for infrared energy). α is the surface property which defines the percent of the incident solar energy that is absorbed by the surface. Solar energy is primarily in the visible part of the electromagnetic spectrum and is absorbed differently from energy in the infrared part of the spectrum. The percent of energy absorbed or emitted at infrared wavelengths is called emissivity. Emissivity is used to determine both the incident infrared energy absorbed and the infrared energy emitted by a surface at the surface temperature.

Applied Radiant Heat Transfer and Temperature Control in Spacecraft

A typical thermal control problem must consider complex geometries and radiant thermal environments. Following the spacecraft through its orbit and collecting all the radiant energy on each surface for the range of orientations is a task best left to computer programs. However, with a few reference tables and some interpolation, much of the information needed to do thermal analysis and design in a initial design phase can be estimated. These estimates and assumptions can be "tested" analytically to see if they are critical drivers in meeting requirements. If they are, they can become the hight priority items for precision definition, or become requirements placed on the rest of the system.

Steps for design of thermal control subsystem

1. A preliminary baseline mission profile is identified. This sets the thermal environment (heat sources for the spacecraft exterior). Variations are expected within the estimated limits. A major change in the mission profile could present a significantly different thermal environment for the flight system and therefore may require a very different design.
2. Location and orientation: (Trajectories, orbits, attitude) are known or baselined; radiant properties of heat sources (solar intensity, planetary reflectance and planetary temperature) are known:

- Trajectory elements
- Orbital parameters altitude, eccentricity, inclination
- Orientation with respect to planet and Sun is known
- Heat source characteristics are known

3. Power dissipation must be known or estimated for the spacecraft equipment (heat sources for the spacecraft interior)
4. Operational and non-operational temperature requirements are established
5. Select simplified geometric shape: cube, sphere, several flat surfaces with an operational orientation representative of the mission flight system (spacecraft)

Once all this information has been collected, the thermal analysis and design can start.

6. Compute the thermal heating environment: Move the geometric shape through the mission profile with the established orientations relative to the heat sources. For each surface and for each incident heat source, identify the maximum and average incident heat flux over time, for each mission phase. Except for special cases, all energy into a spacecraft arrives by radiant energy.
7. Compute the absorbed environmental heat, Q_{env}, on each surface by selecting surface radiant properties (α for solar source and ε for an infrared source) from the following, where S is the solar irrandiance at the planet; R is the percentage of solar irradiance diffusely reflected from the planet: IR is the irradiance of infrared energy from planet; A_p is the projected area toward the sun; A_R is teh area exposed to diffusely reflected solar energy from the planet; and A_{IR} is the area exposed to the infrared energy emitted from the planet.

$$Q_{env} = \alpha S(A_p + RA_R) + \varepsilon IRA_{IR}$$

8. Compute the total IR dissipation capability of the spacecraft geometry by summing all the external surfaces, according to this equation, where ε_n is the emissivity at surface n, and A_n is the radiating area at surface n.

$$Q_{out} = \sigma T^4 \sum_1^n \varepsilon_n A_n$$

9. Use the steady state heat balance equation for a isothermal object, include the electrical power from Step 3 and compute the steady state temperature for the object. Q_{env} is the heat absorbed from environment loads, and Q_{in} is the internal power.

$$Q_{env} + Q_{in} = \sigma T^4 \sum_1^n \varepsilon_n A_n$$

10. Compute temperatures for the maximum and minimum heat loads using the maximum and minimum internal power dissipation.
11. Solve for T and compare with the requirements.
12. Compare the temperature results of all these cases and evaluate the adequacy of a passive thermal design.
13. Adjust the surface properties so the steady state temperature, for the average heat load, is near the middle of the required temperature range.
14. Iterate results using the avaiable independent variables (e.g., areas, finishes, power dissipation).

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3.14.2 Online courses

- UC3M. *The Conquest of Space: Space Exploration and Rocket Science* - Explore the history of space travel and learn the basics of aerospace engineering. Course in edX, course notes in pdf

CHAPTER FOUR

ECSS

The European Cooperation for Space Standardization is an initiative established to develop a coherent, single set of user-friendly standards for use in all European space activities. The following agencies/companies are actively supporting ECSS: Agenzia Spaziale Italiana (ASI), UK Space Agency, Centre National d'Etudes Spatiales (CNES), Deutsches Zentrum für Luft- und Raumfahrt (DLR), European Space Agency (ESA), Netherlands Space Office (NSO) and Norwegian Space Centre.

A complete version of the actual and the older versions of the ECCS can be found at the official page: https://ecss.nl/

4.1 Space project management branch (M)

The general purpose of project management is to implement a process that successfully completes the project in terms of costs, schedule and technical performance. The project management is carried out following a structured approach through all stages of the life cycle and at all levels of the client-provider chain. It integrates all the functions of management, engineering and product assurance required for the execution of the project.

4.1.1 M-10 Project planning and implementation

The discipline of project planning and implementation provides a series of coherent processes to minimize technical, schedule and economic risk of the project. It is carried forward with the following tools:

- **Introducing phases and formal milestones allowing the progress of the project being controlled with respect to cost, schedule and technical objectives**
- **Defining structures that constitute the unique reference system for project management to:**
    - Identify the task and responsibilities of each person.
    - Ensure consistency between technical, administrative documentation and financial activities of the project as a whole.
    - Prepare the schedule and cost activities.
    - Set up a project organization to implement and structure a complete approach to perform all activities needed in the project.

4.1.2 M-40 Configuration and information management

This discipline focused on:

- **Identify, describe and control the technical description of the system in a logical and consistent way throughout the life cycle of the system.**
- **Ensure that the necessary information for an effective execution of all the management processes are recorded, recovered, distributed and modified in a traceable way.**

4.1.3 M-60 Cost and schedule management

The discipline of cost management and schedules provides a coherent series of processes to verify compliance with the planning and organization of the project and thereby ensure the consistent use of human resources, facilities, materials and funds to successfully complete the space project within the established objectives of costs, schedule and performance. It also provides alerts to execute necessary adaptations (for example: re-planning, relocation of resources, etc).

4.1.4 M-70 Integrated logistic support

The integrated logistics support discipline covers the necessary requirements for logistics support throughout the life cycle of the system.

4.1.5 M-80 Risk management

The risk management discipline identifies all risks (including new opportunities) and keep those risks within defined margins and accepted by the project's risk policy. Risk management is addressed to all aspects of the program, including technical performance and quality, programmatic (example: financing, environmental policy), costs (example: type of contract, project cost), schedule and operation (example: logistic support, security).

In particular it includes:

- **The systematic identification, evaluation and classification of all the causes of risks and consequences for the definition and implementation of the decision to accept, monitor or take an action. The risk assessment supports the decision-making process, including the consideration of uncertainties about the risk involved. The independent verification of the assessed risk ensures its objectivity.**
- **The systematic definition, implementation, control and verification of appropriate actions for the elimination or reduction of risk to an acceptable level.**

4.2 Space product assurance branch (Q)

The main objective of Product Assurance is to ensure that space products meet the defined mission objectives, and more specifically, that they are safe, available and reliable. As a support for risk management, PA (Product Assurance) ensures adequate identification, evaluation, prevention and control of technical risks within the limits of the project.

4.2.1 Q-10 Product assurance management

The product assurance management is a multidisciplinary activity that ensures that: a product assurance plan is implemented and managed through all phases of the project and coordinated with all the related stakeholders. It corresponds to the Product Assurance Plan to define all PA activities consistent with the project objectives, requirements, critical points and limits. In particular it includes:

- Search and availability of adequate resources, personnel and facilities to carry out PA tasks.
- Monitoring of processes, reports and visibility of everything related to PA, in particular that related to alerts, critical items, non-conformities, changes, deviations, actions or recommendations resulting from reviews, inspections and audits, qualification, verification and acceptance.

4.2.2 Q-20 Quality assurance

The discipline of quality assurance covers all aspects to ensure that the quality of the product is in accordance with the specifications and that it has been designed, built, verified and maintained with associated documentation throughout its life cycle. In particular it includes:

- Quality assurance for testing facilities.
- Components and equipment of the COTS type (Commercial Off The Shelf).

4.2.3 Q-30 Dependability

The discipline of reliability covers all aspects to ensure that reliability (availability and the factors that influence the reliability, performance and maintenance support) from space products including systems implemented in software and the interaction between hardware and software. In particular it includes:

- Design rules (example: derating, EOL parameters)
- Reliability analysis (example: worst-case circuit performance, failure mode, and effects)

4.2.4 Q-40 Safety

The security discipline covers all aspects in order to ensure that all risks associated with the design, development, production and operation of space products are identified, evaluated, minimized, controlled and finally accepted through the implementation of a program of product assurance. In particular it includes:

The evaluation of risks based on qualitative and quantitative analysis (e.g. hazards, fault tree, etc).

4.2.5 Q-60 EEE components

The discipline of EEE components defines requirements for the selection, control and acquisition of EEE components for space projects, to ensure that they meet the mission performance requirements throughout the life cycle of the products. In particular it includes:

- Program Management component.
- Selection, evaluation and approval component.
- Acquisition component.

- Component of handling, storage and restitution.
- Quality assurance component.

4.2.6 Q-70 Materials, mechanical parts and processes

The discipline of Materials, Mechanical Parts and Processes defines requirements for the selection, control and acquisition of materials, mechanical parts and processes for space projects, to ensure that these satisfy the performance of the mission and the performance requirements of the mission throughout the Products lifecycle. In particular it includes:

- Requirements and selection processes (eg characterization, evaluation, qualification for the intention of use) and the supply of materials and mechanical parts (eg degassing, thermal cycles, radiation, welding, cracking)
- Requirements and processes to avoid planetary contamination (eg sterilization, hygiene and cleaning)

4.2.7 Q-80 Software product assurance

The discipline of Software Product Assurance defines requirements that ensure developed or reused software, as well as software services, perform with property and security in their operating environments. This includes requirements for the development of non-deliverable software, which affect the quality of the products or services provided by a spatial system (for example, tests and software verification).

4.3 Space engineering branch (E)

The disciplines of the “E” branch of the ECSS cover the engineering aspects of space systems and products, including:

- Process engineering applied as scout systems and their elements or functions.
- Technical aspects of the products used or associated with space missions.

4.3.1 E-10 System engineering

The discipline of systems engineering is a multidisciplinary activity that aims coordinating technically the different engineering activities to ensure consistency and general technical integrity within the project. In particular it includes:

- Analysis of definition of technical requirements of the system, integration and control and verification interfaces.
- System activities providing limits to other disciplines such as environment spatial, human factors, celestial mechanics and reference coordinates.

4.3.2 E-20 Electrical and optical engineering

The discipline of optical and electrical engineering covers all aspects related to the design of electrical, electronic, electromagnetic, microwave and optical space products. In particular it includes:

- Management, storage, conversion and distribution of power.
- Analysis and design of requirements as multipactor, satellite load, electromagnetic compatibility, electrical connections and interfaces.
- Technological aspects of communication interfaces.

4.3.3 E-30 Mechanical engineering

The discipline of mechanical engineering covers aspects related to the mechanical design of space products. In particular it includes:

- Thermal control.
- Structures including structural materials.
- Mechanisms and pyrotechnics.
- Environmental control and life support (ECLS).
- Propulsion (launchers and satellites).

4.3.4 E-40 Software engineering

The discipline of software engineering covers the life cycle processes of software products (for example: definition of requirements, architecture design, development, operation and maintenance). In particular it includes:

- The different types of software: on board (embedded), on land and softwares for qualification, testing and verification.

4.3.5 E-50 Communications

The communications discipline covers all aspects related to communications: external interfaces for telemetry, remote controls, scope and data for: satellite-terran station, satellite-satellite, earth station-earth station; and internal interfaces between equipment on board. In particular it includes:

- «Link budget» and protocols at various communication levels.

4.3.6 E-60 Control engineering

The discipline of control engineering covers aspects related to automatic control in space systems. In particular it includes:

- Dynamic and control requirements (for example: attitude control and orbit determination, robotics, overflights and couplings).
- Requirements for sensors and actuators (for example: gyroscopes, sensors).

4.3.7 E-70 Ground systems and operations

The field and operations segment discipline covers the aspects related to mission and ground segment operations, which are an integral part of a complete system implementing a spatial project. In particular it includes:

- Procedures and languages for operations.
- Requirements for the operation of the ground segment.
- Definition of telemetry and telecommand services.
- Requirements of the interface between the satellites and the equipment of the ground segment.

CHAPTER

FIVE

TOOLS FOR A COLLABORATIVE ENVIRONMENT

It is interesting to have a repository where everything is accessible for the team members and where it is possible to have an overview of the evolution of the project. Another suggestion that we make is to use a communication tool like Slack or others, since it is very easy to have access to all conversations and to, for example, keet the conversations about each subsystem separated, therefore keeping everything clean and organised.

The main requirement for these tools is that they must be free to use.

5.1 Remote repository

In a complex project carried out by a small team it is important that all documentation is accessible for every team member, and that a versioning system is in place to be able to review old designs and proposals.

For that purpose, there are two different services that could be of use to the students: file hosting services and source code hosting services. The former provides remote access to any file in a very easy and accessible way, not requiring any new knowledge. On the other side, the latter has very powerful version control system that facilitates collaborative work, but its use requires knowledge in this kind of tools, since they are not very intuitive for beginners.

5.1.1 File hosting services

Regarding file hosting, there are several options available. Some of the most known services in this area are:

- **Google Drive**: it includes an office suite that permits live collaboration, free, 15 Gb of online storage. More information at https://drive.google.com/
- **Microsoft Onedrive**: very good integration with the Microsoft Windows and the Microsoft Office software suite (with an online version available), free, 5 Gb of online storage. More information at https://onedrive.live.com/
- **Dropbox**: very good integration with other services, free, 2 Gb of online storage. More information at: https://www.dropbox.com/
- **Box**: less known service, free, 10 Gb of online storage. More information at https://www.box.com/
- **Mega**: free, 50 Gb of storage. More information at https://mega.nz/

Other interesting options are ownCloud, NextCloud or Seafile, which are open source services that can be self-hosted if wanted.

5.1.2 Source code hosting services

Three options can be recommended for a source code hosting service:

- **GitHub**: https://www.github.com/
- **GitLab**: https://about.gitlab.com/
- **Bitbucket**: https://www.bitbucket.org/

All of them provide similar capabilies:

- Public and private repositories, although GitHub restricts to a maximum of three collaborators
- Version control
- Issue tracking
- Creation of branche of the design and then merging

These services provide very interesing features, but they have a learning curve that not all student will be willing to or can assume.

Resources:

- Git tutorial (Youtube): https://www.youtube.com/watch?v=SWYqp7iY_Tc
- https://jwiegley.github.io/git-from-the-bottom-up/
- https://rachelcarmena.github.io/2018/12/12/how-to-teach-git.html
- https://www.atlassian.com/git/tutorials/learn-git-with-bitbucket-cloud

5.2 Communication tools

For communication inside the team, it is important to keep track of all of has been said and have the different topics compartmentalised using channels. Some interesing tools are compatible with these requirements:

- **Slack**: it is the most widespread application from the list, so it has the higher number of integrations with other services. It is free to use. More information at https://www.slack.com/
- **Microsoft Teams**: very good integration with other Microsoft services like Onedrive. Free to use. More information at https://products.office.com/es-es/microsoft-teams/group-chat-software
- **Flock**: another alternative with interesing features in comparison with Slack, like tasks management or group videocalls. Free to use. More information at https://www.flock.com/
- **Riot.im**: open source alternative, less features. Free to use. More information at https://about.riot.im/
- **Zulip**: open source, possibility of topic-based threads inside channels. Free to use. More information at https://zulipchat.com/
- **Mattermost**: open source, unlimited search history. Free to use. More information at https://mattermost.com/

Most of them include some online storage space and a limitation on the number of the saved messages.

CHAPTER SIX

TEMPLATES

Final report template (.docx) with comments, proposal of contents and structure,... (Preview in .pdf)

Word template (.dotx, EXAMPLE)

Powerpoint template (.potx, EXAMPLE)

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