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Space systems engineering and operation

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Abstract

The purpose of this text is to combine the slides from Professor Lavagna with the content discussed during lectures or presented in other handouts. It is still advisable to accompany this text with the slides, as there will be references to images that are not included in the text (due to my lack of motivation), but which refer to those present on the slides of that specific topic. This PDF contains only the theoretical parts, not the exercises. At the and of every chapter there are questions that were present in the exams of the previous years. Consider that these questions are not the entirety of the ones that have actually come out, so I do not recommend studying only these.

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Introduction

1.1 System Engineering

System Engineering is an interdisciplinary approach that oversees the technical efforts required to transform a requirement into a system solution. It ensures that customer needs are met throughout the entire life cycle of the system. System Engineering involves formalizing program, project, and system management approaches, learning from past experiences, and providing tested procedures for managers. System engineers are responsible for the overall design of the system and make decisions on subsystem combinations to achieve the best design results. They possess extensive knowledge of all fundamental features of the project and understand the interdependencies among subsystems. Additionally, they define mission budgets and ensure consistency across various aspects of the project.
In other words: **is an interdisciplinary process that ensures that the customer's needs are satisfied throughout a system's entire life cycle**.
System engineers are responsible for the overall lifecycle of a (space) system. They have a good knowledge of all fundamental features characterising each part (or subsystem) a project is made of. They have an holistic view of all interdependencies and interconnections among the parts/disciplines. Thus, bearing in mind the final objective of the project, they have to:

- monitor the process evolution
- drive the development of each part to limit risks and maximise consistencies with expectations and make decision on which combination of solutions is to be preferred to achieve the best answer to the given objectives
- address and lead testing for verification of products.

So the System engineers have to:

- define all product budgets (mass, power, link, cost, schedule, etc...) and check their consistency during and at the end of the design process
- Ensures the integration of various disciplines all over the whole life cycle by:
  - Managing the system engineering activities, such as monitoring the project progress, making design decisions by means of trade-offs, models, simulators, breadboard, ensuring the product availability, ensuring design and validation

loop to be performed and ensuring the past experience is considered in the process

– Managing the project planning
– Managing an engineering database to facilitate inter-disciplines exchange
– Managing the documentation production
– Taking care of interfaces both internal and external, of all disciplines contribute to the requirement definition, of the budgets levels and the considered margin.
– Being in charge of considering different technologies together with the TRL and related qualification/tests required
– Tracking the cost effectiveness
– Assessing the data involved in the risk management

1.2 Phases

In the top-down process, objectives and requirements are defined, leading to the identification of different functionalities and their respective requirements. The bottom-up process starts with designing and testing individual parts in a lab, then gradually building and testing the complete system. The ultimate goal is to have well-interfaced clusters of software and hardware that fulfill all functionalities. The top-down process provides a detailed and quantified design, while the bottom-up process involves buying and testing the manufactured parts. Throughout the process, compliance with requirements is checked, and necessary revisions are made to ensure overall compliance. Proper planning and definition of tests, facilities, and infrastructures are essential during the top-down process for the successful execution of the bottom-up process.

1.2.1 Phase 0

In Phase 0 or Pre-Phase A the high-level goals and feasibility of the project are analyzed. This phase helps identify bottlenecks, criticalities, and risks, guiding the decision-making process for the project's direction. ESA, for example, frequently performs such studies to propose missions to the community. Each design phase involves looping rectangles, indicating iterative processes to ensure convergence before proceeding. Milestones and reviews are present at each phase, allowing decisions to continue, stop, or postpone based on progress and analysis. The documentation of each phase is standardized, focusing on specialized analysis and necessary outputs to pass that phase. The management aspect, including cost, risk, planning, and development, is also crucial and requires consideration from the initial rough analysis.After this phase, it follows the MDR (Mission Definition Review)

1.2.2 Phase A

Phase A follows Phase 0 in the top-down process of system engineering. Phase A involves the preliminary mission design, mission requirements, disposal regulation considerations, and feasibility assessment. In this phase, requirements and constraints are established, and a general system-level solution is proposed, such as potential architectures or mission analysis alternatives. However, the design is not fully solved at this stage; instead, the focus is on defining the path for the design solution. Phase A is a conceptual phase that quantifies requirements and discusses potential solutions. Decision points within the phase allow for internal and external (client) discussions, and reviews become more crucial as the phase progresses. The phase concludes with the preliminary requirements review (PRR). The agency conducts a check represented by the orange triangle, determining whether the project is accepted or requires further refinement. If rejected, the process restarts from the beginning of Phase A until the agency approves the project to proceed to the next phase. A final evaluation of the feasibility of the user requirements PRR (Preliminary Requirements Review) ends this phase.

1.2.3 Phase B

Phase B involves detailed design work where different alternatives for systems and subsystems are sized and analyzed. Heavy analysis is performed, such as finite elements for structures, multimodal analysis for thermal and control systems, and refinement of the astrodynamics model. The phase confirms the alternatives selected in Phase A and finalizes the requirements at the subsystems level. A preferred baseline for systems of systems architecture is established, supported by tech and budget analysis. Phase B includes the systems requirements review and the preliminary design review, which must be approved by the agency to proceed with implementation. At this stage, the technology should be at least at Technology Readiness Level 6 (TRL 6), indicating that the mission is capable of flying. Once approved, Phase C begins, involving building facilities, purchasing hardware, and implementing the mission. The duration of each phase varies, with Phase 0 typically lasting 3 months, Phase A up to 1 year, and Phase B up to 1 year, depending on the mission's complexity and cost. Phase C can take 2 to 3 years, depending on the mission. During Phase B approval, all mission features, budget, selected technologies for subsystems, and other critical details are presented to the agency before moving forward with the proper mission design.

1.2.4 Phase C

Phase C is the "product realization process applied to each product up and across system structures." It's a hybrid phase that consolidates the design, and once completed, no changes to components are allowed. Engineering prototypes may be created to test functionalities that numerical analysis alone cannot verify. The critical design review is a crucial step in this phase, where the complete design of subsystems, software, ground support equipment, development plan, procurement plan, and various subsystem designs are presented to the client. The critical design review is a detailed phase that involves analyzing everything in fine detail, including feasibility testing in laboratories to strengthen the proof of concept. Once the critical design review is passed, the construction phase can begin. The CDR (Critical Design Review) marks the approval of the product design specifications.

1.2.5 Phase D

In phase D manufacturing, building, and testing take place. During this phase, there may still be some looping, but the "if condition" for further progress depends on the test results rather than design modifications. The goal of Phase D is to complete all necessary tests, ensuring the system's readiness for flight. Tests during Phase D are conducted in the actual environment that the system will encounter. They follow standardized procedures, covering aspects such as acceptance of mass for launch, temperature, dynamics, and electromagnetic compatibility, especially for telecommunication systems or any components with electromagnetic interactions. Everything is thoroughly tested and qualified, and once completed, the system is delivered to the launch pad or user. In this phase there are:

- The qualification review (QR) to confirm the verification process ended successfully.
- the acceptance review (AR) to confirm the verification process demonstrated the product is free of workmanship errors and ready for operational use.
- the operational readiness review (ORR), to verify the readiness of operational procedures and compatibility with the flight system done on the GS

1.2.6 Phase E

Phase E encompasses the in-space operations of the mission. During this phase, there may still be looping with respect to resolving non-nominalities, which are deviations from the expected or normal conditions. If there are no non-nominalities, the mission can proceed to Phase F, which marks the end of the mission's life. In this phase there are:

- The flight readiness review (FRR) is held prior to launch for last verification of systems and IFs
- The launch readiness review (LRR), held immediately prior to launch.
- The commissioning result review (CRR), held after completion of the on orbit commissioning activities to state the readiness to enter in routine operations
- The end-of-life review (ELR) held at the completion of the mission to ensure and verify the mission completed and systems are read for safe disposal

1.2.7 Phase F

It's the disposal phase that includes all the events from the end of life till the final disposal of the product. MCR (Mission Close-out Review) ensures that all mission disposal activities are adequately completed.

1.3 Life cycle models

To select the most suitable approach for a specific project, various criteria are considered, such as the solution development scheme, timeline for deployment, robustness and consolidation of requirements, risk sensitivity, complexity of the system, and stability of the environment. Common life cycle models that can be used to manage the project are:

- **Sequential/waterfall**: A linear approach where each phase is completed before moving to the next.

- **Incremental**: The project is divided into smaller increments, and each increment is developed and tested separately.
- **Evolutionary**: The project evolves through iterations, with each iteration adding new features or improvements.
- **Spiral**: The project progresses in cycles, with each cycle refining the system based on feedback and risk analysis.
- **V-diagram**: A model that shows the relationship between testing and development phases, emphasizing the importance of verification and validation.

1.3.1 Sequetial/Waterfall model

The project activity follows a top-to-bottom approach, progressing through discrete, sequential, and linear phases. In the original model, feedback loops were not considered, which led to unplanned changes once the requirements were written. However, later modifications incorporated feedback loops to address this issue. This approach is suitable for simple systems with a limited number of alternatives and low risks. The requirements are well-defined from the beginning, and the cost, technical, and schedule baselines remain stable throughout the project. Additionally, the delivery of one product at a time is a key characteristic of this model.

1.3.2 Incremental model

The system architecture and requirements are well-defined and remain unchanged throughout the project. The design is implemented in increments, where each increment provides useful operational capability. Successive increments add greater functionality, resulting in a partial solution that may cover part of the functionality, provide all functionality with limited performance, or be implemented at a limited number of operational locations. An example of this approach is seen in projects like EGNOS−Galileo or Starlink constellation. In such cases, the system is well−defined from the beginning, but full finance is not immediately available, and time−to−market is crucial. As a result, the system is delivered in multiple stages, with each delivery adding incrementally more functionality to achieve the final desired system.

1.3.3 Evolutionary model

The final design is not necessarily well-defined in the early stages of the process. The basic lifecycle is repeated to deliver successive versions of the product, gradually increasing its functionality over time. Throughout the operational period, the system is reviewed, and feedback is incorporated to improve its performance. To facilitate this approach, the initial versions of the system are kept small and introduced into use to gather valuable user feedback. The system evolves gradually towards its final design, allowing for adaptations to changing technology, requirements, and the environment. Examples of projects following this approach include the Sentinel fleet and Artemis missions. In such cases, the requirements are not fully known or defined at the beginning, and users need to experience early versions of the system to better understand their needs. This approach is particularly beneficial when dealing with complex interfaces, rapidly changing environments, or pioneering systems. The project involves successive product deliveries, each offering increased capability over time as the system evolves and matures.

1.3.4 Spiral model

The Spiral Model combines the evolutionary model with risk assessment. The philosophy behind this model is to acknowledge that system development involves inherent risks and to thoroughly understand these risks. To manage risks effectively, the Spiral Model incorporates prototyping as a risk reduction strategy. The model follows a basic five-step iterative sequence:

- Identify the project's objectives, alternatives, constraints, and requirements.
- Evaluate and assess different alternatives, identifying and resolving risks associated with the project.
- Design, develop, and verify the product.
- Plan for the next cycle of development.
- Review and analyze the results of the current development cycle.

The Spiral Model is commonly used in software development, particularly when the product contains high risks. The model specifies incremental capability levels, and multiple product deliveries occur over time as the system evolves and matures.

1.3.5 V−Diagram

The simple life cycle can be re-organized as a V-diagram to highlight specific aspects:

- Verification defined at the design level, ensuring that the design meets the specified requirements.
- Verification between phases, checking each phase's deliverables against their respective requirements.
- Validation as end-to-end verification, ensuring that the complete system fulfills the user needs.
- Decomposition and definition of what is to be built, breaking down the system into manageable components.
- Integration and verification of what has been built, ensuring that the assembled system functions correctly as a whole.

The V-diagram emphasizes the importance of verification and validation throughout the development process, ensuring that the system meets its intended requirements and performs as expected.

1.3.6 Robust design

A robust design output from a comprehensive trade space analysis is fundamental to protect the next expensive phases from unavoidable changes and variations.
Left Graph: The graph illustrates the development timeline, highlighting key milestones, and the expected investment in the system. In the initial phases, the cost is relatively low, but as the development and testing progress, it exponentially increases. During the early stage of the project, it is crucial to establish a well-defined plan for subsequent steps, as this bears significant responsibility. It is essential to set clear requirements and carefully identify the most suitable architectures, considering various solutions and trade-offs.
Right Graph: The graph on the right shows the cost distribution for phases A and B in relation to the overall cost, accounting for up to 30

1.4 Functional analysis

Functional analysis is a systematic approach used to identify, describe, and establish the relationships between essential functions that a system must perform to achieve its high-level objectives successfully. It focuses on determining the top-level functions required for the system to accomplish its goals during the early stages of the project life cycle. This analysis considers the locations, frequency, operational concept, and environmental conditions under which these functions need to operate. By addressing these aspects, functional analysis provides a foundation for detailed planning and system development. On the other hand, functional decomposition is another method that complements functional analysis (defines **what** functions need to be done). It involves identifying actions related to functionalities and organizing them in sequence or in parallel to achieve the final objective. This process allows you to define the engineering feasibility of your goals even before actual implementation begins. Functional decomposition generates important documentation and tools like the Work Breakdown Structure and Product Tree, which serve as valuable guides throughout the project's real-life execution. To better understand the significance of functional decomposition, a comparison between Icarus and the Wright brothers is provided. Icarus attempted to replicate birds' flight, focusing on their form, while the Wright brothers concentrated on different functionalities, like take-off, landing, and horizontal movement. This example illustrates the importance of analyzing functionalities individually to find more efficient and optimal solutions. The text stresses the value of avoiding hasty solutions to maintain flexibility and attain better outcomes. Furthermore, the text includes two more examples of functional decomposition, specifically related to a real mission involving debris inspection. The mission aims to perform a removal action on debris, and the analysis highlights the need for careful assessment of functionalities like close-up visual inspection and guideline navigation and control. Understanding the required velocity for clear images is a critical part of this analysis. In summary, both functional analysis and functional decomposition play crucial roles in system development. Functional analysis focuses on identifying essential functions and their relationships, while functional decomposition organizes functionalities for efficient implementation. Together, these methods provide a robust framework for planning and executing successful projects.

1.4.1 PBS

Product Breakdown Structure (PBS) is important to understand the components of a system and identifying the required expertise in a team. The PBS is built once the requirements are fixed and helps keep track of all elements, from the flight segment level to specific physical components. In contrast, the Work Breakdown Structure (WBS) defines the system engineering activities needed during each project phase and assigns responsibility to the individuals or entities handling those activities. The "work packages" represent the specific activities to be completed, and the colors indicate the entities responsible for them. Overall, the PBS and WBS play crucial roles in project planning, ensuring all components are accounted for and that the right expertise is available to accomplish the necessary tasks.

1.5 Conops

While doing the functional analysis, there is another important step: Conceptual Operations. Functionalities are related with operations. Functional decomposition/analysis provides a logical sequence of essential actions that must occur without specifying their time duration or interdependencies. On the other hand, Conceptual Operations (ConOps) plays a vital role in defining the work by:

- Identifying time-critical requirements: ConOps highlights the crucial elements that need to be addressed within specific timeframes.
- Fixing functions on a timeline and defining their concurrency, overlapping, and sequential relationships: ConOps establishes the temporal arrangement of functions, determining whether they occur simultaneously, sequentially, or with overlapping timeframes.
- Generating alternative architectures for mission accomplishment: ConOps explores various possible structures and arrangements to achieve mission objectives efficiently.
- Identifying/defining phases and modes: ConOps delineates the different phases (occuring once along the mission) and modes (system states occurring more than once along the mission timeline) of the mission, such as launch, disposal, communication mode, sun pointing mode, etc.

By leveraging ConOps, mission planners can gain a comprehensive understanding of the mission's temporal and functional aspects, aiding in the development of effective strategies and architectures to accomplish mission goals successfully.

1.5.1 Definition of operation

- **Integration and test operations** are crucial phases in a project's life cycle, where the system is thoroughly tested to ensure its functionality and performance meet project requirements. During this period, operational simulations are conducted, evaluating both functional and environmental aspects. These simulations include end-to-end command and data system exercises, verifying the system's performance against simulated project scenarios. There is also the launch integration that involves the repetition of integration and test operations in the launch−integrated configuration.

- **Launch operation** encompass the launch countdown, ascent, and orbit injection. Critical event telemetry plays a significant role during this phase. Following orbit injection, spacecraft deployment operations configure the spacecraft to its orbital setup, involving critical events like solar array, antenna, and other deployments, as well as orbit trim maneuvers. In-orbit checkout is conducted to verify the health of all systems, followed by on-orbit alignment, calibration, and parameterization to prepare for science operations.

- **Science operation** The primary operational phase is dedicated to conducting science operations, making the most of the spacecraft's operational lifetime.

- **Safe-hold operations** are activated in response to on-board fault detection or ground command. This mode maintains the spacecraft in a power positive and thermally stable state until the fault is resolved and science operations can resume.

- Throughout the mission, **anomaly resolution and maintenance operations** may be required. These operations might demand additional resources beyond the established operational ones.

- At the end of the project life, **disposal operations** take place. The spacecraft is either controlledly re-entered or repositioned to a disposal orbit. In the latter case, the dissipation of stored fuel and electrical energy is necessary. These disposal operations ensure the proper and safe conclusion of the project.

1.5.2 Definition of architecture

The association between different functionalities and the responsible parties should be thoroughly established in the ground architecture. Although not necessary at the early stages of a project, this level of detail becomes essential in the final stage of mission design.

1.6 Setting criteria

It's important of having clear criteria and measurable quantities when making decisions and selecting alternatives for a project. After being launched, the first task is to establish communication with Earth, which is achieved through a simple signal known as the beacon, containing the spacecraft's identification. In the context of the project, it is crucial to define criteria that will guide the decision-making process throughout its entirety. The choice of criteria should align with the mission's objectives. For instance, criteria such as mass, radiation, cost, or time to flight can be relevant depending on the nature of the mission. For scientific missions, factors like cost and time to flight might not carry as much significance as they would in commercial missions. Ultimately, having well-defined criteria and relevant measurable quantities is vital for making informed decisions during the project's execution.

1.6.1 Requirements

The requirements presented here serve as the foundation for the initial analysis of the assigned mission's objectives. To approach the problem, it is essential to identify the functionalities that need to be addressed. As the understanding of these functionalities grows, the requirements are shaped through sentences with or without quantification. Requirements are categorized as follows:

- Functional Requirements (**What** has to be done): These outline what the system or its subsystems are expected to do. For example, "the mission shall land on the moon's surface." These functional requirements can be specified at various hierarchical levels throughout the project's phases, ultimately being frozen by Phase B.
- Operational and Performance Requirements (**How** has to be done and **How well** shall be satisfied): These address how the tasks are performed and their expected performance levels. For instance, "the system shall land softly," with the performance criterion being "the system shall land with a minimum vertical velocity of 1 m/s." The process of setting these requirements may occur iteratively, involving brainstorming and refinement.
- Verification (**How** should be verified): This category pertains to how the system's adherence to the requirements will be validated. Methods such as numerical analysis or real-world testing campaigns are specified to cross-check that the system meets the defined requirements during design and implementation.

There are two categories of requirements for a mission:

Mandatory Requirements: These requirements are crucial and must be met to fulfill the customer's needs. They clearly outline the necessary and sufficient conditions for the system to be considered acceptable, using the term "shall." Mandatory requirements are pass-fail criteria, and scoring functions are not applicable. Additionally, they are not subject to trade-offs between different requirements, making them non-negotiable. The quantification process for these requirements involves thorough analysis, reasoning, and discussions to determine the final values. This analytical approach aids in making well-informed decisions and effectively setting up the mission.

Preference Requirements (Nice to Have): Once compliance with mandatory requirements is ensured, preference requirements are considered to identify the most favorable designs. These requirements aim to enhance customer satisfaction and are often expressed using terms like "should," "want," or "nice to have." Preference requirements are evaluated using scoring functions to measure figures of merit. Since there might be trade-offs among different criteria and no single optimal solution, a multicriteria decision aiding technique is employed for evaluation. These requirements are not mandatory for mission success but can be included if feasible during the design phase. They present opportunities for improvement, but if they are not achievable due to technical or energy constraints, they can be omitted. Preference requirements are indicated using the verb "should."

The choice of verbs used, "shall" for mandatory requirements and "should" for preference requirements, distinguishes between these categories. This standardization helps define essential elements necessary to achieve the mission's goals while allowing flexibility to consider additional features if possible. The process of quantification and setting requirements ensures that the mission is well-planned and capable of effectively meeting its objectives. Essential characteristics and principles for developing effective requirements are:

- Achievable: Requirements should be technically and economically feasible based on feasibility analysis specific to the mission's targets and characteristics. Checking achievability helps understand alternatives.
- Affordable: Requirements should not only be feasible but also economically viable. It's crucial to consider practical constraints, such as availability of rare materials or excessive human resources.
- Justified: Each requirement must have a rational basis, and the reasoning behind the quantification should be transparent.
- Classified and Clear: Requirements must be categorized by type and should be unambiguous, ensuring that all readers interpret them consistently.
- Traceability: Requirements should have clear sources, and their logical relationships, such as parent-child relationships, should be identifiable for traceability.
- No Contradictions: The requirements should not contradict one another. A thorough review of all requirements as a whole is necessary to avoid inconsistencies.
- Specification of Verification: Verification methods must be specified for each requirement to ensure that the system is built correctly according to the scope.
- Avoid Negative Statements: Requirements should be written in positive sentences rather than using negative phrasing to facilitate verification.
- Responsible Entity: Each requirement must be clearly ascribed to a specific entity, and there should be a responsible person in charge of its writing and verification.

These principles help in crafting well-defined, coherent, and verifiable requirements that lay the foundation for a successful mission design.

1.6.2 Path to follow

The process of defining requirements follows a logical schematic that begins with external goals and high-level qualitative information. During this initial phase, a thorough understanding of the problem is developed by analyzing the environment and relevant activities in the field. This information helps identify meaningful aspects. As the process continues, the requirements are organized, classified, and justified using quantifiable measures. More aspects are specified while keeping the problem in focus, leading to the establishment of the first loop of requirements. Once this loop is defined and preliminary analysis is completed, attention shifts to exploring potential solutions. In this phase, aspects such as trajectory options, propulsion systems, communication frequencies, and conceptual operational scenarios are considered. However, this is not the stage for detailed system design; instead, it involves refining the analysis while also examining the availability and feasibility of relevant technologies. The requirements at this point should effectively constrain and describe the problem, laying the groundwork for the subsequent stages of mission development. The focus remains on aligning the project with its objectives and ensuring the most suitable approach is taken to address the identified challenges.

1.6.3 Verification and Validation

Prescribing tests is an early step in the system life cycle to ensure that the final system complies with its requirements. This involves both verifying and validating the requirements. Verifying requirements involves ensuring that they are consistent and that a real-world solution can be built and tested to demonstrate compliance. On the other hand, validating requirements ensures that they align with the mission objectives. As you formulate the requirements, it's essential to consider the feasibility of verifying them. You should question whether suitable verification methods, such as experiments, software, models, or instruments, exist to measure the specified quantities. If verification is not feasible, efforts should be made to find solutions or remove the requirement altogether. While this task may initially be challenging and uncertain, it will gradually evolve and become more detailed as the project progresses. Keeping this aspect in mind from the outset will contribute to a well-structured and successful system development process.

1.7 Drivers

Requirements play a crucial role in driving the engineering process, and they need to be continuously and critically assessed throughout the project's lifecycle. Among the requirements, there are specific ones known as drivers, which have a significant impact on the system design process. These drivers consist of a set of mandatory requirements that highly constrain and influence decision-making at all levels of the project. Examples of drivers include environmental requirements for missions to destinations like Mercury or a comet, as well as mechanical interface requirements for a constellation launch. Drivers are not just regular requirements; they hold greater importance, and their resolution becomes a top priority right from the beginning of the project. Addressing drivers first is essential, as they set the foundation for the subsequent stages of development. Failing to address them in a sequential manner could lead to chaos and hinder the overall progress of the project. Therefore, these powerful requirements must be handled with utmost attention and urgency to ensure a successful and well-organized engineering process.

1.8 Interfaces

The "interface requirements" category is crucial for any interaction with external entities, and engineers must define the necessary interfaces to ensure compliance. Interfaces can be classified into two types:

- Interfaces between subsystems
- Interfaces between the main system and the external world (e.g., launcher, ground station, test facilities, etc.), which require constraints or requirements.

Well-designed subsystems transmit completed products to other subsystems, and clear, unambiguous, and limited interfaces are essential. Subsystems should be defined in a way

that minimizes the amount of information exchange required between them, streamlining the communication process and ensuring efficient collaboration. Interfaces can be established towards the external world, involving entities such as the launcher, mothership, and ground stations. For instance, if different frequencies are used from those of the ground station, it would create an incorrect interface. Therefore, specifying that ”ground station shall operate in X frequency” becomes an interface requirement concerning the interaction with ground stations from your vehicle. Moreover, numerous interfaces must be considered among and between subsystems. For example, if all members are sizing elements with a specific mass, a configuration specialist needs to control the inertia matrix. It is crucial to clearly define the precise location of elements within the spacecraft and any associated limitations to ensure compliance with the interface requirements.
The development process of spacecraft and space instruments involves several stages to ensure a successful and space-qualified product. This staged development schedule is widely adopted across space engineering companies. The model philosophy comprises the following phases:

- Breadboard stage: This stage aims to prove the functionality of the instrument concept and its ability to provide the intended measurements. The breadboard may not resemble the final instrument but operates on the same basic principles.
- Structural and Thermal model: This model replicates representative mechanical and thermal properties of the final instrument.
- Electrical model: The electrical model demonstrates the integration of the instrument design with the spacecraft’s data and power buses.
- Qualification model: This fully representative model uses components of the same type and specification as those to be used on the spacecraft. It is specially designed to withstand the harsh space environment. This model undergoes extensive testing under specified operational and non-operational environmental conditions.
- Flight Model: The Flight Model is the instrument that will be used on the spacecraft. It shares the same design and specification components as the Qualification Model but is rigorously tested under Acceptance Level environmental conditions, which are broader than expected operating conditions but not as extreme as Qualification levels.

By progressing through these stages, the development process mitigates risks, ensuring a space-qualified instrument with proven performance for its intended mission.

1.9 Trading off alternatives

Exploring different design alternatives that meet the specified requirements is essential. To identify the most favorable options based on performance and cost figures of merit, multicriteria decision aiding techniques are employed. To compute the figures of merit, a step-by-step process is followed:

- Initially, design engineers provide estimates for the figures of merit.
- Models are then constructed and evaluated to gain insights into the performance and cost aspects.

- Simulation data is derived from the models, providing more accurate assessments of the alternatives.
- Prototypes are created and measured, allowing for further refinement of the figures of merit.
- Finally, tests are conducted on the final system to validate and fine-tune the figures of merit.

This comprehensive approach ensures a thorough evaluation of the design alternatives, leading to the identification of the most suitable solution that aligns with the specified requirements and project objectives.
The engineering process is a multi-stage endeavor that requires the generation of alternatives and careful trade-offs. To begin, engineers define functionalities and goals while establishing criteria to aid decision-making. Brainstorming sessions are then conducted to explore potential solutions, ensuring they meet the requirements. These identified alternatives undergo detailed verification through simulation, prototyping, and testing. Some options are retained while others are discarded based on their feasibility and performance. Eventually, a set of options, known as Phase B, is selected as the final baseline for the mission, aligning with objectives and addressing challenges. The iterative process of generating alternatives and conducting trade-offs ensures robustness and adaptability during the design phase. By exploring various scenarios and considering every possible condition, the decision-making process becomes thorough and well-informed. Even after the mission is operational, ongoing trade-offs may be necessary based on evolving conditions and challenges faced.
The concept of "drivers" is vital in prioritizing critical requirements that significantly influence the mission's design process. Identifying these key requirements early on allows engineers to address them first, ensuring their impact on the project's outcome is adequately managed. The engineering process involves constant analysis and trade-offs at various stages, with careful consideration of functionalities, requirements, and alternatives. This comprehensive and iterative approach empowers engineers to confidently proceed with their chosen solutions, leading to a well-planned and effective mission.

1.10 Design process

In space missions, considering reusability is crucial. Rather than always designing something new, it's essential to explore what has been done before. When designing a new component, one must consider whether it is dedicated to a specific mission or can be customized for future applications. For instance, comparing active debris removal solutions like net tethered systems and robotic arms, while the net solution may be unique, the robotic arm can be used in other types of missions as well. Another critical point to keep in mind is the decision to make or buy certain components during the design phase. This decision affects the evolution of the project in terms of time and cost. For each component, there must be an analysis to explore potential alternatives, focusing on obtaining a deep understanding of what is needed for the mission and what best fits its functionalities. Flexibility is more important than striving for optimality, as there is no one-size-fits-all perfect solution. The engineering process follows milestones, ensuring that the design and requirements are frozen at each stage. After arriving at the bottom left of the process with fixed requirements and a suitable solution, the implementation phase begins. This involves acquiring or making the selected components, conducting functional tests, assembling the parts, and performing tests on the entire product. Overall, the process involves careful consideration of reusability, decision-making regarding make or buy options, and step-by-step implementation to achieve a successful space mission.

1.10.1 Product implementation

The focus is on hardware trade-offs and decision-making in the development of a space system. Emphasis is placed on defining criteria and requirements for all subsystems and the entire system. There are various alternatives for each component on board, starting from low-fidelity models like breadboards and advancing towards the final flight unit. As the hardware progresses from low-fidelity models to the flight unit, manufacturing complexity and sensitivity to cleanliness increase, requiring management in a clean room environment. Key decision points in this process are the qualification and flight units. The qualification unit is identical to the flight unit in terms of materials and performance but undergoes extreme stress testing without being flown. Its survival indicates that the design and construction have passed the necessary tests, providing confidence for building the actual flight unit. The flight unit also undergoes testing, with stresses tailored to the mission. The text underscores the importance of testing critical aspects and making informed decisions to ensure the system's success while considering factors such as cost and time. The readiness of the system is contingent on thorough testing and carefully considered trade-offs made during the hardware development process. Here a list of the units:

- Breadboard: This is a low-fidelity unit designed solely to demonstrate basic functionality. It often utilizes commercial or improvised components and does not provide definitive information about operational performance.

- Brassboard: A medium-fidelity functional unit that incorporates a significant amount of operational hardware and software. It is structured to operate in simulated operational environments to assess the performance of critical functions.

- Engineering Unit: A high-fidelity unit that showcases crucial aspects of the engineering processes involved in developing the operational unit. Engineering test units closely resemble the final product (hardware/software) as much as possible. They are built and tested to establish confidence in the design's functionality in expected environments.

- Prototype Unit: The prototype unit demonstrates the form, fit, and function of the final product on a representative scale within its operational environment.

- Qualification Unit: This unit is identical to the flight unit in terms of form, fit, function, components, etc. It undergoes testing under extreme environmental conditions, such as thermal and vibration stresses. It is usually not flown due to these off-nominal stresses.

- Protoflight Unit: In projects where a qualification unit is not developed, the flight unit may be designated as a protoflight unit. It undergoes a limited version of qualification test ranges and will be flown.

- Flight Unit: This is the end product that will be flown and typically undergoes acceptance level testing before deployment.

1.10.2 TRL

In order to determine which models to implement and when to implement them the Technology Readiness Level (TRL) has an important role. A clear understanding of TRLs is crucial for decision-making regarding model usage and testing. TRL serves as a valuable criterion in navigating the complexity of selecting subsystem implementations and tests. To determine the TRL level, questions like the current TRL status and the expected achievement level are helpful. TRL levels are indicative of technology maturity, and proposing a novel solution with low TRL highlights the need for further development and testing. Conversely, if there are precedents, like breadboard tests, it can motivate progress to higher TRL levels, such as Engineering Model (EM) testing. It is essential to consider that making changes or variations to an existing unit may affect the TRL level. For instance, introducing different configurations or architectures could lower the TRL level, necessitating reevaluation and additional testing.
The technology's level of development is classified using a scale called the Technology Readiness Level (TRL). A lower TRL indicates less developed technology. For a system to be flight-worthy, all its technologies must be at least at TRL 6. There are 9 TRL levels:

- TRL 1: Basic working principles are observed.
- TRL 2: Technology concepts are formulated.
- TRL 3: Proof of concept through experience.
- TRL 4: Technology is validated in a laboratory setting.
- TRL 5: Technology is validated in a relevant environment.
- TRL 6: Technology is demonstrated in a relevant environment.
- TRL 7: System prototype is demonstrated in a relevant environment.
- TRL 8: System is completed and qualified.
- TRL 9: Actual system is proven to be effective in an operational environment.

1.11 Verification methods and IV/V stages

We are currently in the finalization phase of our detailed design, where we have a clear understanding of what we want to build, and we need to ensure that the design aligns with the mission goals and fulfills all the requirements and performance criteria. To verify the requirements and needs during the building phase, we rely on the Engineering Standards of System Engineering (ECSS-E-ST-10-02C), which provide standardized specifications.
There are four alternatives for verifying the requirements, and it's essential to determine not only how but also when to verify them, although the latter is not always a project requirement: inspection, analysis, Review−of−design and test. Verification takes place at different stages of the project:
a) Qualification: Ensuring that the requirements are met, leading to an increase in the Technology Readiness Level (TRL).
b) Acceptance: Confirming that there are no workmanship errors before deployment.
c) Pre-launch: Ensuring the system is properly configured for launch activities.

d) In-orbit (including commissioning): Checking for any degradation that may have occurred during launch and verifying proper functioning in space.
e) Post-landing: Assessing the product's integrity and performance after completing the mission.
By following these verification steps, we can ensure that our design meets all the necessary requirements and performs optimally throughout the mission.

1.11.1 Inspection

Inspection involves a visual examination of project documentation, drawings, vendor specifications, or the physical item itself. This method is used to ensure compliance with specified requirements by checking physical attributes like dimensions, weight, physical characteristics, color, markings, etc. It is particularly useful for requirements that cannot be assessed through tests, making it an important early-stage verification process.

1.11.2 Analysis

Analysis entails evaluating data using established analytical techniques to determine if the system meets specified requirements. The analysis techniques include systems engineering analysis, statistics, qualitative analysis, analog modeling, similarity, and computer and hardware simulation. It is selected as the verification method when test or demonstration techniques may not adequately address all conditions or when demonstrating compliance is more challenging than performing analysis. Mathematical modeling, often with the aid of software simulation, is commonly used to analyze complex system behaviors and assess compliance with requirements.

1.11.3 Demonstration

Demonstration involves assessing system conformance by operating, adjusting, or reconfiguring a test article. This method is particularly used to verify system characteristics such as human engineering features, services, access features, and transportability. Unlike analysis or tests, demonstrations do not require elaborate instrumentation or specialized test equipment. They rely on observing and recording functional operations to validate that the system meets specified requirements.

1.11.4 Tests

Testing is a crucial verification method that involves evaluating the system or system components under controlled conditions to determine compliance with quantitative design or performance requirements. Tests may be conducted at different levels of assembly within the system assembly hierarchy. They may require specialized equipment and elaborate instrumentation to measure specific parameters relevant to the requirements. Testing is the preferred method for requirement verification and is used when analytical techniques do not yield adequate results, failure modes might compromise safety or mission objectives, or for components associated with critical system interfaces. Testing is standardized and typically includes verification of system performance under relevant environmental conditions. It also identifies potential failures that may not have been considered during the design and development process. However, conducting tests during the final mission phase is generally avoided to prevent last-minute critical issues that could jeopardize the success of the project.

1.12 Exam question

1) Describe the product lifecycle approach based on the phases
2) Sequential/Waterfall model
3) Incremental model
4) Evolutionary model
5) Spiral model
6) V diagram
7) Explain the phylosophy that is behind the margins
8) What is the model philosophy used for in a space equipment/mission development
9) Protoflight model
10) Engineering unit
11) What are the goals of the verification and validation processes
12) Define the requirement concept, the features it must have, and the process to generate it
13) Constraints of the launchers
14) Product Tree
15) Work Breakdown Structure
16) Functional Tree
17) Conops
18) TRL

Chapter 2

Space environment

The space environment encompasses various factors, and our analysis will consider these aspects based on the mission's location:

- The Sun
- The Earth's magnetic field
- Thermal conditions
- The neutral atmosphere
- The plasma zone
- Radiation levels
- Microgravity conditions
- Vacuum conditions
- Presence of debris/micrometeorites
- Planetary-specific environmental factors
- Launchers

2.1 The Sun

The Sun acts as a colossal thermonuclear reactor, releasing a staggering amount of power, approximately $3.8510^{26}W$. Its behavior resembles that of a blackbody, with a surface temperature of 5800 K and a peak emission wavelength of 460 nm (visible light). The Sun completes one rotation every 28 days. Emitted particles predominantly consist of protons and electrons, accounting for 95% of the total, with the remaining being heavy ions. The Sun boasts a magnetic field of about 15 Gauss, roughly 30 times stronger than Earth's magnetic field, which ranges between 0.2 to 0.7 Gauss. Solar activity follows a cycle of approximately 11 years, influenced by the number of sunspots and flares. Sunspots and flares result from deformations in the Sun's magnetic field, leading to significant electromagnetic and mass emissions during periods of increased sunspot activity. Mass emissions also occur during the Corona Mass Ejection (CMEs) events. The Sun's plasma, moving through space, gives rise to the solar wind, reaching Earth at velocities ranging from 200 to 900 km/s and generating the Interplanetary Magnetic Field (IMF). This IMF profoundly influences the space environment.

2.1.1 Sun activity

Sun activity is centered around two primary aspects: radiation and emitted particles. Depending on the mission's timeline, several elements require consideration.
Electromagnetic (EM) Radiation: This type of radiation poses challenges for payloads relying on EM, such as telecom and deep space observation. During the Sun's passage, these payloads may need temporary shutdowns to prevent interference.
Emissions and Effects on Earth and Spacecraft: The Sun emits low to medium-energy particles, including X-rays, EUV, and radio bursts, which can lead to geomagnetic storms. Such events can disrupt satellite communications, spacecraft orientation, radar, radio signals, and cause satellite drag.
High-Energy Particles: The second group comprises high-energy ions and protons emitted by the Sun. When these energetic particles interact with the environment, they can induce changes in atmospheric conditions and magnetic field intensity. Significant emissions of these particles may adversely affect instrument reliability, like GPS. Fortunately, detection of such phenomena is possible hours before they reach Earth.

2.1.2 Elettromagnetic radiation and Solar activity modelling

To study the effects of solar activity on the solar system, Earth, and space activities, a crucial aspect is the modeling of solar activity. The primary indices used to represent the 11-year solar cycle are the sunspot number (R) and the 10.7 cm wavelength radio flux (F10,7) emitted mainly from the Corona. These solar activity values can be measured at the ground and have a strong correlation with the Sun's UV radiation, which significantly influences Earth's atmosphere. By detecting the frequency at a wavelength of approximately 11 cm, which passes through the atmosphere, the behavior of the Sun can be monitored and measured from the ground. Observing the number of sunspots, which plays a crucial role in solar activity, helps in deriving models for the Sun's life cycles.
Graphs displaying the periodic behavior of sunspot numbers, irradiance, and mathematical models derived from data are essential in locating mission windows intelligently, considering the one-year cycles. Understanding the Sun's emission in the UV spectrum is vital when selecting materials for shielding, as UV rays can deteriorate the structural properties of many materials. Additionally, studying the Far-Infrared (FIR) spectrum is crucial for thermal control. The Sun's emission ranges from Extreme Ultraviolet (EUV) at 10-124 nm to Far-Infrared (FIR) at 15 µm-1 mm. The solar constant represents the radiation falling on a unit area of the surface, normal to the line from the Sun, per unit time, outside the atmosphere, at an average Earth-Sun distance (1 AU).
The following values for electromagnetic radiation are relevant:

- Solar constant at 1 AU: $1366.1 W/m^2$
- Maximum solar energy flux (winter solstice): $1412.9 W/m^2$
- Minimum solar energy flux (summer solstice): $1321.6 W/m^2$
- Solar radiation pressure (100% reflecting plate): $9.0210^{-}6 N/m^2$

While the visible spectrum contains the most significant power density, it's crucial to consider the infrared and UV spectra due to their effects on materials, as mentioned earlier. Understanding and modeling solar activity play a pivotal role in various space-related endeavors.

Planet/Sun EM radiation effects

- **TCS effects**: The Sun serves as the primary source of thermal perturbation in space missions. Besides the direct radiation emitted by the star, other factors come into play, such as Albedo (reflected radiation from the Sun on a third body) and Infrared (IR) radiation emanating from the planet's temperature. These effects are influenced by the proximity of the Sun to the specific planet or body of interest. In the space environment, thermal exchange primarily occurs through radiation. As a result, the Sun plays a fundamental role in two key aspects:

  – Electromagnetic radiation: The Sun's energy impacts spacecraft directly, affecting their thermal conditions.

  – Thermal effects on natural bodies: The Sun's energy emission influences the temperature of nearby celestial bodies.

  External thermal sources in space include:
  a) Solar flux from the Sun, which depends on the square of the distance between the spacecraft and the Sun.
  b) Albedo, which represents the reflected radiation from the planets in the UV band.
  c)Infrared (IR) radiation emitted by the planets, which remains above $0°C$.
  d) Cold space, with a temperature of approximately $3°C$, acts as a sink for thermal energy.
  Graphs depicting planet radiance concerning different wavelengths (and energy content) are essential in understanding thermal effects. The left group of graphs illustrates radiation, while the right group represents albedo. The planets are assumed to be grey bodies, with an emissivity of 1. The charts show values of radiation and albedo on Earth's surface. Notably, albedo peaks occur at the poles, where ice reflects a significant amount of incoming radiation. Conversely, in regions with water, radiation is absorbed. In contrast, the IR chart highlights regions of interest at the equator.

- **UV band effects**: he Sun emits UV radiation in the wavelength range of 10-400 nm, with approximately 90% of this radiation being absorbed by the Earth's atmosphere. In the space environment, UV radiation has significant effects on organic materials. Polymeric materials, such as Teflon, Dacron, and Mylar, experience erosion and increased brittleness with exposure to UV radiation. For paints and polymerics, their optical and mechanical properties, including absorbance, reflectivity, and emissivity, change with exposure so the darker the material, the more the absorbance is. Given these effects, it becomes crucial to study the impact of the Sun and the space environment on the mission. This study encompasses material selection, coatings, and considering the mission's overall lifespan. The chart illustrates the degradation of material properties over time due to UV irradiation. To mitigate these effects, one possible solution is to protect the system by covering it with a layer of aluminum or a similar protective material. The photo provided demonstrates the visible effect of irradiation on a payload, underscoring the importance of addressing this issue to ensure the success and durability of the mission.

2.2 Earth's magnetic field

Around Earth, two main magnetic fields exist: the Internal Magnetic Field (Main) and the External Magnetic Field. The Internal Magnetic Field results from geodynamic interactions between the core and the surrounding liquid composed of iron and nickel. On the other hand, the External Magnetic Field arises from interactions between the outer atmosphere (ionosphere/magnetosphere) and the solar wind. In the case of Earth, the Internal Magnetic Field also interacts with the outer atmosphere. The External Magnetic Field is induced externally and isn't related to the planet's composition, unlike the Internal Magnetic Field (e.g., Venus). This field experiences daily and seasonal variations. To rely on this external field for protection against Solar wind or Galactic Cosmic Rays, it's essential to consider that its orientation changes periodically. While this level of design might not be necessary for our project, it's crucial to keep it in mind. Two indices, Kp and Ap, shown in the tables, offer insights into the relationship between solar activity and the active atmosphere in terms of electrical intensity. These indices are updated every 3 hours and play a significant role in space weather analysis. The Earth's magnetic field has non-uniformity, known as the Southern Atlantic Anomaly (SAA), which represents a hazardous region for electronic devices. It contains trapped protons and electrons from the Sun due to the weakened magnetic field in this area. The SAA needs consideration during the design phase, and if a spacecraft passes through it, one solution may be to switch off the onboard electronics temporarily.

2.2.1 Magnetic field of other planet

In the Solar System, the most relevant magnetic fields are associated with Jupiter and Saturn, while the other celestial bodies, including Mercury, the Moon, Mars, and Uranus, have almost inactive magnetic fields. Venus lacks an internal magnetic field similar to Earth but exhibits an external magnetic field due to solar radiation interaction with its atmosphere, though less intense than Earth's.

2.3 Atmosphere

The atmosphere is a crucial element to study in space missions. It can be broadly divided into three zones:

- The homosphere (0-90 km) includes:
  Troposphere (0 to 21 km altitude)
  Stratosphere (12 to 50 km altitude)
  Mesosphere (50 to 90 km altitude)

- The thermosphere extends from about 90 km altitude to 250 km or 700 km (depending on solar and geomagnetic activity levels).

- The exosphere begins at the top of the thermosphere.

The atmosphere's thickness and the presence of ionized particles are highly influenced by the Sun's life cycles. Its main effects on space vehicles include drag, contamination, and erosion caused by atomic oxygen, which is a highly reactive element. For instance, star tracker lenses should be positioned opposite to the velocity to avoid adverse effects. The atmosphere's influence on space vehicles is evident in terms of drag coefficients, induced torques affecting attitude, and trajectory changes. Standardized models are available for many missions, and adherence to these conventions is essential, even if you cannot use the specific model you desire. The atmosphere also causes contamination of the vehicle surface when gases emitted by the vehicle collide with atmospheric particles, leading to the deposition of materials on optical surfaces. Aerodynamic heating occurs around 150 km, reducing solar radiation absorption by approximately 1%.
The left chart displays temperature versus altitude, showing how it varies based on the Sun's activity. Notably, a significant temperature gradient exists during the transition zone into the eclipse, resulting in differential expansion of materials. The right chart illustrates temperature variation due to solar activity. Conduction and convective heat transport distribute energy within the atmosphere, but to a limited extent. The temperature gradient is particularly pronounced from the transition zone into the eclipse. In the exosphere, the gradient can increase to over 200 K, ranging from 840 K on the nightside to 1060 K on the dayside.
Similar temperature trends can be observed qualitatively for other planets. Neptune, being mostly gaseous, experiences relatively little change in temperature trends, partly due to its distance from the Sun.
The density of the atmosphere is heavily influenced by solar activity. This has crucial implications for launching time and the entire mission duration around a planet, considering the effects of drag. Additionally, when determining the mission time window, it's essential to consider the impact of solar activity on the atmosphere's size, which can affect lowering the apogee or perigee with air drag. The first chart illustrates how high or low solar activity influences orbit decay, leading to a significant time gap for deorbiting modifications due to solar effects, potentially spanning years. Solar activity significantly impacts the density level at a given altitude. Consequently, planning the mission's operational time window must account for solar activity, as it can cause substantial changes in drag, affecting the fuel budget and mission lifetime.

2.3.1 The ionosphere

The Earth's atmosphere includes the ionosphere, where high-energy UV radiation dissociates oxygen in its upper part. For instance, the ISS operates within this intense region, where UV radiation and dissociated oxygen pose significant considerations. The ionosphere acts as a barrier for radio signals below 100 MHz, preventing direct communication between space vehicles and Earth. However, this region can be utilized for reaching non-visible Earth locations through signal reflection. Below an altitude of approximately 86 km, the lower atmosphere is turbulent and a homogeneous gas mixture, with pressure decreasing exponentially. Above this level, photochemical processes disrupt homogeneity, particularly UV absorption from the Sun leading to oxygen dissociation.

Atomic Oxigen

Atomic oxygen (AO) plays a crucial role in Earth's atmosphere, particularly at altitudes above 150 km. At around 200 km and beyond, the mean free path of atomic oxygen

becomes significantly greater than most material dimensions. This impacts material erosion and optical properties of space vehicles in this region. Heat exchange in this area is primarily radiative, and the aerothermo-dynamics exhibit properties based on free molecular flow. The density of atomic oxygen depends on Solar Activity (SA) and increases with higher solar activity. At geostationary altitudes (around 36,000 km), the density of the atmosphere is similar to that in the interplanetary medium, with AO density varying according to SA levels. As a result, it is essential to consider solar activity when planning missions, as the presence of atomic oxygen can influence a mission's lifetime and performance. For instance, in Low Earth Orbits (LEO), the typical mission lifetime is about four years, while the SA period lasts one year. Therefore, mission planners need to carefully consider the operational time window with respect to solar activity to maximize the mission's success and longevity.
Selecting suitable materials for spacecraft surfaces is crucial, considering the presence of atomic oxygen and its effects on material erosion and optical properties. Specific materials exposed to atomic oxygen may experience significant weight loss. For example, Mvlar material's outer surface exposed to AO could lose about 35% of its weight after just three days of exposure. Additionally, a composite truss based on graphite could experience over 30% weight loss due to AO erosion, leading to structural issues rather than thermal concerns. Regardless of the planet you orbit, if there is oxygen in the atmosphere, you will encounter AO presence due to the high-energy radiation emitted by the Sun. Hence, it's essential to consider these effects even when planning missions to other planets. Slides (L5-Environment) provide a list of materials affected by AO and how Optical Properties have changed in previous missions due to variations in absorptivity (high-energy radiation (UV) absorption coefficient) and emissivity (low-energy radiation (IF, thermal) absorption coefficient). These coefficients' values determine whether your system will experience an increase in temperature or not.
Apart from erosion, the recombination of AO must also be taken into account. The most effective way to counteract it is to adopt protective oxide layers or silicone-based coatings. Oxide layers prevent reactions with AO, ensuring material integrity. However, caution should be exercised with the presence of silver on solar arrays, as it could threaten power production if it reacts with AO. Another important consideration is the graphite fibers in composites. They should either be placed inside the spacecraft, away from AO exposure, or adequately protected. The intensity of AO flux is closely related to Solar Activity (SA), and the table below shows surface recession rates for specific materials based on SA levels, AO flux, and altitude. While initially selecting materials based on thermal control and structural design, a final check considering the specific environmental conditions encountered during the mission is crucial. This may require modifying the design configuration or adjusting the attitude of flight to protect exposed surfaces accordingly. It is important to find a balance between mitigating AO effects and maintaining thermal control during the mission.

2.4 Plasma

A plasma is a gas that is either partially or fully ionized, causing its particles to respond collectively to magnetic and electric fields. This collective motion is a result of the electrostatic Coulomb force between charged particles. When spacecraft orbit Earth, they can encounter several distinct plasma regimes: the ionosphere, which is the region of cold plasma (plasmasphere) at the top of the atmosphere; the magnetosphere, an area domi-

nated by Earth's magnetic field above the ionosphere and bounded by the magnetopause; and the solar wind, which surrounds the magnetosphere and originates from the Sun, blowing throughout interplanetary space with an average velocity of 468 km/s.
When the solar flux interacts with the atmosphere, plasma is created and directly interacts with the magnetosphere in the case of Earth. The evolution of the plasma is influenced by the direction of the solar wind, with its tail following the direction of the Sun, and the strength of the surrounding magnetic field. The magnetic field traps the plasma around the Earth. This interaction with plasma can lead to various effects, such as parasitic currents generated at the ionosphere level, electrostatic charge buildup on the system (requiring discharge on the ground) due to hot plasma at higher altitudes, and potential damage to electronic components from high-charged particles. Moreover, at Lagrange points, plasma effects may also be experienced.

2.4.1 Problems

- Charging: Plasma flying in space can lead to surface charging, which poses a significant hazard. The free movement of electric charges (electrons and ions) in the plasma can cause arcs and result in the deposition of particles on material surfaces, leading to different surfaces on the spacecraft charging to varying potentials. This can create strong local electric fields that interfere with instruments, damage electronic systems, and affect scientific measurements. Surface charging drawbacks include electronic system damage and interference with measurements. To mitigate these effects, spacecraft materials on the outer surface, such as thermal blankets, solar cell cover glass, sensor optics, and exposed cables, must be carefully checked and controlled. Additionally, if an electrical propulsion system's plume is not entirely neutral, it can interact with the plasma, impacting the exit velocity of the propellant.

- Contamination: Certain thin films, like thermal blankets, may not effectively shield the materials beneath them from high-energy electrons (ranging from 10 keV to 50 keV). This can lead to surface charging issues affecting various spacecraft components covered by thin films, where the film's thickness is less than 100 micro$-$meters, such as Aluminium (0.03 g/cm^2) and multi$-$layer insulation. Flowing ions can impact the surface materials, causing sputtering. These ions carry significant kinetic energy, typically around 1 keV for protons and 4 keV for Helium ions. Additionally, neutral atoms generated by the spacecraft can be ionized by sunlight or interact with other ions, resulting in a low-energy ion population (below 10 eV). These ions can be attracted to negatively charged surfaces, leading to surface contamination issues.

- Solar array: In the ionosphere, ions and electrons neutralize high potential surfaces on spacecraft, except for solar panels, which maintain high potentials due to sunlight. The incident ions and electrons collected by the solar array generate a current through the ionospheric plasma, but this current cannot be used to power spacecraft systems, leading to reduced solar panel efficiency. Electrons are collected more easily, causing the positive end of the array to move towards space potential and leaving the negative end at a high negative potential. The equilibrium potential depends on various factors. The electrical potential difference of the panels induces the plasma to reduce the generated current, resulting in induced differential potential (spurious current) through the panels. To mitigate this, the panels must be decoupled from the spacecraft to avoid system interference and limit power production issues. A conductive path on the spacecraft's outer surfaces can prevent the buildup of differential potential that attracts plasma. However, introducing such a path may impact thermal blankets' performance, which aim to reflect radiation. Evaluation is needed to balance resolving the differential charge issue and potential thermal deficits. Grounding requirements based on established standards (ECTSS) should be followed.

2.4.2 Design requirements

The primary control for managing spacecraft charging levels is achieved through the design of a conduction path from each surface to the ground and other parts of the spacecraft. Therefore, the main focus lies on grounding and ensuring electrical continuity throughout the spacecraft structure, subsystems, and electronics.
The requirements for spacecraft charging are as follows:

- Maximum permitted voltage: A maximum differential voltage of -1000 V between a dielectric and an adjacent conductor, and a maximum differential inverted voltage of +100 V.
- Maximum electric field within the dielectric: Limited to 107 V/m.
- All structural and mechanical parts must be electrically bonded to each other, with a resistance of less than 106 Ω at each bond.
- A risk assessment of potential discharges from sharp edges and tips must be conducted.
- All external and internal metallic layers of a thermal blanket should be grounded to the structure (although this may conflict with thermal insulation goals).

In the magnetosphere, beyond the plasmapause and within the magnetopause, the plasma environment is characterized by high temperatures and low densities, primarily of solar wind origin. Magnetospheric electrons accumulate on exposed spacecraft surfaces, leading to a net current that negatively charges the surface. While opposing currents usually prevent significant charging levels, insufficient currents can result in spacecraft charging to hundreds or thousands of volts. This difference in charging levels among various spacecraft surfaces increases the risk of damaging electrostatic discharges.
The concern and mitigation process involve several steps:

- Consideration of the spacecraft's orbit (LEO, MEO, GEO, or extra magnetosphere) and the mission's electrical characteristics, including the use of electric propulsion, scientific instruments for field or plasma measurements, and the presence of high voltage systems.
- Analysis, verification, and testing to assess vulnerabilities and interactions between generated plasma, ambient plasma, and the ionosphere, as well as potential power leakage, discharges, and sputtering.

Overall, managing spacecraft charging is a critical aspect of ensuring the safety and proper functioning of space missions, and careful consideration and testing are essential to mitigate potential issues.

2.5 Particles and radiation

The design of your spacecraft will be significantly influenced by the space environment, particularly concerning energetic particles and radiations. There are two key aspects to consider: the flux, which represents the quantity of particles per unit of time and volume or surface, and the energy carried by these particles.
The graph illustrates the mapping of flux and energy based on the class of particles:

- Generally, when dealing with highly energetic particles, the flux tends to be low, and conversely, high flux corresponds to lower particle energies.

- Ionizing radiation can be categorized into three main groups based on their sources:

  - Galactic Cosmic Rays (GCR): These rays are present throughout space and are relatively aggressive, but their flux is low. Interestingly, their impact on the environment is inversely related to the solar cycle. During periods of high solar activity (peak of the sun cycle with increased particle emission), GCRs are more rejected and deviated, resulting in a relatively small contribution to the overall radiation exposure.
  - Radiation Belt Particles: These consist of protons and electrons, originating from the sun and being trapped by the magnetic field of the celestial object around which your spacecraft orbits.
  - Free Solar Particles from the Solar Wind: These are particles emitted by the sun and travel freely in space.

These factors play a crucial role in the design considerations for your spacecraft, as they can affect its performance, shielding requirements, and overall safety during the mission. Properly accounting for the space environment and radiation exposure is essential for successful space missions.

2.5.1 Galactic cosmic rays

Their composition consist of 85% protons, 14% alpha particles, and 1% nuclei with Z>4. Energy levels range from 0 to more than 10 GeV.
Compared to trapped particles, GCRs have a low flux. These rays possess high energy, especially during solar minima. The rate of energy deposition is measured by their "Linear Energy Transfer" (LET) rate. LET Rate denotes the rate of energy transferred to material by an ionizing particle traveling through it.

2.5.2 Radiation belt particles

Van Allen

The combination of external and internal magnetic fields generates an electromagnetic field that traps charged particles. This phenomenon is best observed in the regions around Earth known as the Van Allen belts, discovered by Van Allen in the late 1950s. The presence of these belts was verified by the USA science spacecraft, Explorer I. The graph depicts a section of Earth, with distance represented in terms of Earth's radii on the horizontal and vertical axes. On the left side, the spatial distribution of protons inside

the magnetic field is shown, while the right side displays the electron distribution. These regions indicate where electronics are most vulnerable to potential dangers.
For protons, the region to avoid is the MEO (Medium Earth Orbit), whereas for electrons, it's just before the GEO (Geostationary Earth Orbit). It's worth mentioning that particle accumulation is higher near the south Atlantic magnetic anomaly, as particles can penetrate more deeply and reach nearer regions.
The graph also shows the energy of protons and electrons as a function of flux and distance in terms of Earth's radii. For electrons, two peaks are evident: one coinciding with the proton peak in the belt, and a second one at medium-high altitude. The maximum isolines for protons are higher than those for electrons, indicating that protons pose a greater danger.

2.5.3 South Atlantic Anomaly

The South Atlantic anomaly creates a concentrated area of radiation, which is the main source of significant radiation encountered in low Earth orbits with altitudes below approximately 800 km and inclinations between 30°-70°. It is crucial for my spacecraft to avoid these specific inclination and altitude ranges. If I inadvertently pass through this region, I must take measures to enhance the robustness of my system and ensure that my electronics remain unaffected by the radiation exposure.

2.5.4 Difference in planet

The issue of radiation exposure occurs on planets with their own magnetic fields, not just on Earth. Let's compare the radiation environments of Jupiter and Saturn systems with that of Earth. Notably, practically all of Jupiter and Saturn's moons are within these radiation regions, which is a critical factor to consider when planning a mission to fly around moons like Io, Europa, or Enceladus. However, Earth's moon lies outside Earth's radiation belt, though other aspects must be taken into account. To accurately study the radiation environment, analysis and simulations are essential. The comparison between Jupiter and Earth's environments can be observed in the graphs, showing the flux and energy levels of protons and electrons. For Earth, protons are more energetic, while for Jupiter, electrons have higher energy, yet both exhibit a similar behavior: higher energy results in lower flux. Consequently, we must primarily address the charged particles that are most relevant in the environment we are operating.

2.5.5 Effect and damage of radiation

Radiation can cause damage in various ways, involving three fundamental mechanisms: ionization, atomic displacement, and prompt effects.

- Ionization: This occurs when radiation directly interacts with the nucleus or atoms of matter. It involves two types of atom interactions:
  - Inelastic collision with atomic electrons: Energy transfer removes electrons from atoms (ionization) or excites them.
  - Elastic collision: Incident particles are deflected elastically, with insufficient energy to remove electrons from atoms.

- Atomic Displacement: This non-ionizing effect causes atoms to move from their usual positions in crystal lattices. Energetic protons are the main source of atomic displacement, which can lead to significant damage to solar cells and power reduction.

- Prompt Effects: These are anomalies resulting from a single energetic particle striking a device, creating an ionized track of electron-hole pairs along the particle path through the semiconductor. There are different types of prompt effects:

  - SEU (Single Event Upset): In this transient effect, a particle striking a digital device causes a change in its logic state (e.g., from 1 to 0), resulting in a modification of commands or data. However, the device can be restored to its original logical state.
  - SEL (Single Event Latchup): Another type of SEU where a particle hitting a digital device alters its logic state. In this case, the effect is more severe as it can lead to a permanent change in the device's behavior. Unless the current to the affected device is turned off, it may experience burnout, causing irreparable damage.
  - SEB (Single Event Burnout): This is a permanent effect where large currents generated by particle interactions can destroy the devices beyond repair.

The Total Ionizing Dose (TID) is measured in [rad] or [krads], representing the flux per unit of time and unit of surface.
To better understand the correlation between onboard components and the radiation environment, consider a classification based on subsystems and electronic components. Energetic ions, mainly originating from cosmic rays and solar particle events, lose their energy rapidly when interacting with materials, primarily through ionization. This energy transfer can have disruptive or damaging effects on various targets, such as living cells or memory elements, resulting in Single Event Upsets (SEUs) in components or detectors (causing radiation background). SEUs and biological effects can also occur due to nuclear interactions between highly energetic trapped protons and materials, particularly sensitive parts of components, biological experiments, and detectors. In such cases, the proton breaks the nucleus apart, leading to highly-localized ionization.
Energetic particles also cause interference with payloads, notably affecting detectors on astronomy and observation missions. They can produce a "background" signal that is indistinguishable from the photon signal being counted or even overload the detector system. Furthermore, energetic electrons have the capability to penetrate thin shields and accumulate static charge in internal dielectric materials, such as cable and other insulation, circuit boards, and ungrounded metallic parts. This buildup can subsequently lead to discharges, resulting in electromagnetic interference.

2.5.6 Design

Documentation always provides guidelines and numerical values that you can use to set the requirements based on the desired level of performance. One critical aspect to consider is energetic particle radiation.

Shielding

- **Energetic particles radiation**:Whit the graph we are able to discusses the effects of radiation on components without LTE (threshold) greater than 60 and how different orbits influence the impact of protons, electrons, and other particles. The graph presented shows the total ionizing dose on the y−axis, representing the system's ionization that can cause malfunctions, and various orbits on the x−axis, each affected by different particles. The shield size is a crucial design parameter, and thicker shields result in lower radiation levels. Protons have the most significant impact on the total ionizing dose, especially in low pericentre orbits like GTO with pericentre as low as 250 km. Increasing the shield thickness better protects electronic devices from ions, reducing the total ionizing dose. A dose of around 10^4 rads/year is generally acceptable for spacecraft.
  Placing electronic devices at the center of the spacecraft is beneficial as the metallic cage of the spacecraft acts as an additional shield, reducing the flux of energy reaching the devices. This approach eliminates the need for excessively thick shields, offering an effective solution to mitigate radiation effects.
  The graph illustrates dose output based on shield thickness from a design perspective, considering different types of orbits. TO orbit is the most challenging due to the accumulation of protons and electrons in the Van Allen belts. MIR orbit experiences the lowest dose as it is well-protected. The Polar orbit, passing through the belts briefly, is exposed to auroral electrons, which are less harmful than protons but more energetic than regular electrons. Choosing the appropriate shield material affects the system's weight, with lower material density being preferable. The graphs are also parametrized based on the flux of electrons and protons, which are more aggressive in LEO and MEO. Comparisons between Earth and Europa's environments show differences in magnitude and shield thickness due to various materials and orbital inclinations. Europa's environment requires shielding from Jupiter's energetic and numerous electrons.
  In conclusion, understanding the radiation effects on different orbits and selecting the appropriate shield thickness and material are crucial in spacecraft design to protect electronic components and ensure successful space missions.

- **Energetic particles in belts**: The cumulative graph illustrates the dose per year at different altitudes, considering various contributions of energetic particle radiations. The Bremsstrahlung effect, responsible for generating particles through ionizations within the material and initiating a cascade effect, is shown in this graph. However, since the data pertains to Earth, we observe that the primary effects come from external electrons and protons, rather than the Bremsstrahlung effect.
  This graph serves as a valuable reference for selecting suitable components for the spacecraft. Data sheets of satellites, such as CubeSat shop or Skylab, provide information about the krads that the devices can withstand. This information is essential for understanding the spacecraft's survival capabilities, especially at different altitudes and durations of exposure. It helps identify critical points where certain components may face risks, such as semi-hard parts or commercial off-the-shelf (COTS) components. For instance, cameras and electronics in CubeSats currently operate within the range of Dose = 10A krads.

- **Cosmic ray**: The shielding principles discussed earlier for protons, electrons, and belt particles can also be applied to charged particles, such as ions in galactic cosmic

rays. The same nomenclature and materials can be used for this type of shielding. Notably, water, which contains hydrogen, is an effective shielding element. For manned missions, water can be stored outside the spacecraft to serve as a protective shield, allowing the crew to stay inside the spacecraft module. However, the drawback is that water is heavy, and its storage needs to be carefully considered, especially as it may require more water than what is needed for the Environmental Control and Life Support System (ECLS) on the spacecraft. The concept revolves around having a main module within a water bubble inside the spacecraft. If a solar storm or cosmic ray surge is expected, the astronauts can retreat into this water-protected module for safety during the storm's duration. This approach offers a viable solution for shielding astronauts from harmful radiation during their stay on the Moon. Jackets with ample water content are being studied for this purpose, serving as both protective gear and a hydration source for astronauts during extravehicular activities on the Moon, where radiation exposure is a concern.
It's worth mentioning that, just like before, graphene-based solutions still prove to be the most effective in terms of shielding with the same material quantity concerning radiation dose.

- **Single Event Effect**: In addition to evaluating radiations over time as a flux, we also need to consider the impact of single particles. These effects, known as Single Event Effects (SEE), can be troublesome, particularly in the vicinity of the South Atlantic Anomaly. SEE refers to the effect of a single ionizing particle (such as ions, electrons, photons, etc.) striking a sensitive node in a micro-electronic device, such as a microprocessor, semiconductor memory, or power transistors, causing a change in its state. SEE can be classified into two main classes:

  - Disruptive Effects: These events lead to permanent and irreparable damage to the electronic component, rendering it non-functional.
  - Non-Disruptive Effects: These events cause temporary malfunctions in the electronic component, but it can recover and resume normal operation afterward.

  While we are not required to be experts in assessing the hardness of these events, it's essential to be aware of their existence and possible impact. Data sheets for electronic components usually provide information about their tolerance to SEE events, in addition to their capacity to withstand Total Ionizing Dose (TID). Common types of SEE include:

  - Single Event Upset (SEU): A single ionizing particle causes a temporary change in the state of a device, leading to a bit-flip in memory or logic circuits.
  - Single Event Latch-up (SEL): An ionizing particle triggers a latch-up condition, causing the device to draw excessive current and potentially damaging it.
  - Single Event Transient (SET): A temporary voltage or current spike occurs due to an ionizing particle striking a sensitive node.
  - Single Event Burnout (SEB): An ionizing particle causes excessive current to flow, leading to the destruction of the affected component.
  - Single Event Gate Rupture (SEGR): An ionizing particle causes damage to a transistor's gate structure, impacting its functionality.

When selecting electronic components, it is crucial to check for their tolerance to these SEE events, along with their resistance to Total Ionizing Dose. This information will help in analyzing the environment and determining if special mitigation measures are necessary.

2.5.7 Focus on this last topic

SEU (Single Event Upset): The components susceptible to SEU are digital elements onboard such as processors, microcontrollers, and FPGA. SEU affects semiconductors by altering the status of the memory cell, causing a change in the stored data, which can lead to reading incorrect values. To safeguard against this issue, significant investments are made, including:

- Purchasing immune materials or implementing reset mechanisms in the components.
- Designing algorithms to include protective sequences within the coded data. By detecting any changes in these sequences, it becomes evident that the subsequent data is compromised and should not be relied upon.
- Incorporating correction algorithms that not only detect errors but also pinpoint their location and rectify the data string.

SET (Single Event Transient): Similar to SEU, SET affects analog components, such as converters and power controllers, causing transient disturbances in the output signal. To counteract SET, similar mitigation strategies are employed: using immune materials or implementing software filters to prevent the propagation of transients and isolating the system during potential solar storms or particle events.
SEL (Single Event Latch-up): SEL is the most hazardous effect, impacting digital components like SEU. It induces unexpected peaks of current (parasitic currents) flowing through the components, which can be detrimental. Mitigation approaches for SEL include:

- Using immune materials or incorporating dedicated circuits to limit parasitic currents.
- Implementing circuits to turn off the affected device during its passage through the South Atlantic Anomaly.

SEL can lead to a loss of device functionality due to high currents from SEE, and it may be catastrophic. Requirements regarding SEE are defined in terms of the allowable Linear Energy Transfer (LET). The component comes with a LET threshold, and its susceptibility to SEE is assessed accordingly:

- If LET(th) $<$ LET, a SEE rate analysis is required to identify the Bit Error Rate.
- If LET(th) $> 100MeVcm^2/mg$, the component is assumed to be SEE immune, i.e., less susceptible to SEE effects.

The reference unit for assessing the SEE event rate is events per bit per day. The software tool used for this analysis is CREME96, and radiation tests are often necessary to validate the component's resilience.

2.5.8 Analyses required

It is important to accurately specify and analyze the radiation environment to ensure the proper functioning and protection of space systems during missions. Phase A Required Analysis:

- The expected radiation environment specification for a space system includes various aspects, such as proton and electron energy spectra, fluence spectrum of solar protons, instantaneous energy spectra of trapped electrons and protons, and solar energetic protons.
- For manned missions, the environment is transformed into dose-equivalent.
- Damage-equivalent fluencies of 1 MeV electrons and 10 MeV protons are provided for solar cell damage estimates.
- NIEL (non-ionizing energy loss) 10 MeV equivalent fluencies are specified for CCD, optoelectronic, and optical components as a function of shielding depth.
- Orbital time-behavior of radiation-belt, cosmic ray, and solar energetic particle fluxes is considered if the mission is susceptible to radiation background in sensors.
- Additions to the above environments from onboard nuclear sources are taken into account.
- The specification considers the evolution of the mission orbit, which can have significant effects on radiation-belt exposure.

2.6 Microgravity

The concept of microgravity experienced by spacecraft orbiting Earth:

- Microgravity Environment: Spacecraft in Earth's orbit experience a state of free fall, theoretically not subjected to acceleration. However, factors like atmospheric drag, solar pressure, and spacecraft vibrations contribute to a microgravity environment, resulting in each spacecraft experiencing a few "g" of acceleration, a small fraction of Earth's gravity. Gravitational force weakens with the square of the distance from Earth.
- Scientific Implications: Microgravity leads to interesting phenomena, such as substances not floating in relation to each other, absence of convection currents during chemical reactions, no gravity vector reference for organism growth, and different fluid dynamics. These aspects benefit scientific experiments in various disciplines. However, microgravity poses challenges for humans, including shifts and loss of body fluids, bone tissue loss, loss of vertical reference, and space sickness.
- System Considerations: In microgravity, materials and elements do not naturally float, convection is absent, and organisms have no natural reference for orientation and growth. Design considerations include careful direction and control of liquids, pressurized systems or turbopumps for fuel tanks, and the use of solid components for friction reduction. Objects designed for microgravity may face challenges when exposed to gravity, necessitating thorough testing.

- Technical Considerations: Designing spacecraft for microgravity involves technical considerations, such as simulating free-floating conditions in ground tests, forced fuel feeding, understanding combustion behavior, adapting systems related to humans with artificial vertical references and restraints, and providing specific solutions for low-gravity missions, including reducing vibrations and shocks and compensating for active drag.

- Gravity Gradient and Testing: The gravity gradient allows spacecraft to align roughly with the local vertical using simple appendages, common for microsats. Testing components in microgravity involves four key opportunities:

  - Deployment of Extendable Parts: This method involves deploying extendable components, as seen in the top-left image (similar to Rosetta's S.A.). However, scaling the experiment is essential, considering not only geometry and time scale but also dynamic scale, including time, mass, and length. Scaling can sometimes pose challenges, and in certain cases, complete scaling might not be feasible, leading to issues where specific effects might not be visible within a reasonable time frame.
  - Dropping Tower: The facility shown below (in Bremen) offers the ability to conduct experiments in a microgravity environment. To use this setup, the experiment must be appropriately scaled as well. The primary advantage is that the experiment experiences brief moments of microgravity during the free-fall descent in the tower.
  - Parabolic Flight: This method allows for testing while flying in a plane that executes parabolic flight paths, creating short periods of microgravity during the plane's free-fall phase. This approach provides a unique opportunity to conduct tests directly with the test equipment.
  - Sounding Rockets: Among the options, sounding rockets offer the most extended duration of microgravity, typically providing around 14 minutes of microgravity environment. This extended time is beneficial for testing various components and systems. However, the launch process introduces challenges, necessitating the development of a telecommunication system to collect real-time data or designing a gondola that can be softly retrieved to retrieve data offline afterward.

  All of these testing facilities are crucial to assess whether the system under consideration can meet the requirements of the microgravity environment it will operate in. Conducting rigorous testing and verification ensures that components function as intended and are capable of performing effectively in the microgravity conditions.

2.7 Vacuum

Another significant environmental consideration is the vacuum present in space, which can lead to a process called outgassing. Outgassing refers to the sublimation of surface atoms when they experience pressures higher than their vapor pressure (ranging from 10^{-15} to 10^{-11} Pa). There are two main negative effects of outgassing:

- Loss of Material and Mechanical Properties: When certain materials experience outgassing, they lose material, leading to variations in their mechanical properties. For instance, structural protection materials may no longer function as intended.

- Deterioration and Condensation: The material that sublimates during outgassing can condense and deposit on other surfaces, which can be problematic and affect various components.

This phenomenon is particularly critical for thermo-optical properties of solar panel coverglasses and surfaces devoted to thermal control. To address these concerns, materials used in space systems are selected based on low outgassing criteria. The criteria are based on the micro-VCM (Volatility Condensable Material) test, considering the location of the outgassing source and its mass. The CVCM (Collected Volatile Condensed Material) is required to be less than 0.1% of the initial specimen mass, and the TML (Total Mass Loss) must be below 1%.
To determine compliance with outgassing specifications, dedicated tests are conducted. The process involves conditioning samples in a controlled chamber where humidity and temperature are stabilized. The material is then subjected to medium vacuum for 24 hours to allow water collected during the outgassing test to be released. After re-conditioning the material, TML and RML (Recovered Mass Loss) are evaluated to ensure compliance with the specified limits. Overall, outgassing is a critical consideration to ensure the proper functioning and longevity of space systems, and rigorous testing is necessary to assess the materials' performance in the space environment.

2.8 Debris

Debris has become an increasingly significant concern in the space environment over the last decade. This category includes not only metallic elements but also asteroids and natural small objects. Similar to other environments where humans have ventured, space near Earth is facing a pollution issue. The highest traffic of debris occurs at altitudes between 800 and 1000 km. When we examine the trackable objects, we find the following distribution: Only 6% are active and controllable satellites. 13% are "mission-related objects" such as lens covers, bolts, and jettisoned solar arrays. 16% are rocket bodies. 22% are inactive satellites. 43% are fragments resulting from more than 130 satellite and rocket explosions. Debris can originate from various sources, including small solar cell pieces, entire spacecraft, or upper stages. A collision with debris could trigger the "Kessler effect," where one impact leads to fragmentation and subsequent cascading impacts exponentially. The European Space Agency (ESA) provides a debris calculator called MASTER to estimate debris size, mass, and period of orbit.
Most of the debris is concentrated in low Earth orbit (LEO) or sun-synchronous orbit (SSO) used for Earth observation. To tackle the issue of space debris, two approaches are necessary:

- Mitigation: Designing spacecraft to avoid creating more debris by de-orbiting in a controlled manner, placing defunct satellites on "graveyard" orbits, and emptying propellants from spent upper stages, among other measures.

- Remediation (ADR - Active Debris Removal): Implementing techniques to protect new systems from collision risks. This may involve ground control and avoidance maneuvers for larger objects and developing special "shields" for long-duration spacecraft like the Space Station.

The CleanSpace ESA program is studying various ADR techniques to address space debris. By taking mitigation measures for the future and pursuing remediation efforts for the past, we can strive to mitigate the debris problem effectively.
Concerns related to collisions with space debris and meteoroids are essential to address. Some potential issues to consider are collisions with tanks or habitat zones, especially pressurized subsystems, can be problematic.To mitigate this, spacecraft designers can construct the front panel of the satellite with multiple layers to attenuate the impact and protect the inner surfaces. The analysis of impact risks is crucial, and it involves using tools to compute the probability of impacts. The tool considers the average density of particles as $2.8g/cm^3$ and assumes them to be spherical.
To assess the probability of a certain number of impacts occurring within a given time, the Poison distribution is used. Addressing collision risks with debris and meteoroids is crucial for the safety and functionality of space missions. By utilizing advanced tools and adopting appropriate design measures, space agencies can better protect their spacecraft and ensure successful operations in space.

2.9 Planetary protection

Planetary protection is distinct from planetary defense, as it involves regulations to prevent biological contamination of both the target celestial body and the Earth during sample-return missions. This responsibility is overseen by an international entity called COSPAR (Committee on Space Research), which focuses on space affairs and planetary exploration. COSPAR classifies missions into 5 categories:

- Category I: This is the lowest category where minimal attention is required. The planetary protection measures adopted are standard, ensuring a clean environment without any specific precautions. Missions falling under this category are not concerned with chemical processes or the origin of life. Examples include Mercury, lo (Jupiter's innermost moon), and S-type Asteroids, which are primarily silicon-based.

- Category II: In this category, missions involve celestial bodies where chemical evolution and the origin of life are not of direct interest. As such, no specific planetary protection measures are warranted, and no requirements are imposed. Category II missions comprise various missions to celestial bodies that are of significant interest in terms of chemical evolution and the origin of life. However, the chance of contamination from spacecraft is remote and unlikely to jeopardize future exploration. For these missions, simple documentation is sufficient, including a short planetary protection plan outlining intended impact targets and post-launch analyses. The applicability: Venus, Moon, Comets, C-type Asteroids, Jupiter, Jovian Moons, Saturn, Saturnian Moons, Uranus, Uranian moons, Neptune and Moons, Pluto & Charon, Kuiper Belts.

- Category III encompasses missions that are considered highly significant in terms of biological evolution, and thus, the interaction between the spacecraft and the target celestial body could have substantial implications. Consequently, specific procedures and documentation are required to ensure protection. In such cases, the clean room must adhere to a higher standard. Additionally, sterilization procedures, such as using electric devices or UV explosions during the flight or planetary entry, are employed to prevent terrestrial matter from contaminating the target environment. Category III missions mainly consist of fly-by and orbiter missions to celestial bodies of interest for chemical evolution and the origin of life. These missions have a notable chance of contamination that could potentially jeopardize future biological experiments. Consequently, detailed documentation is necessary, surpassing the requirements of Category I missions. Implementing procedures include trajectory biasing, cleanrooms during spacecraft assembly and testing, and bioburden reduction. While no impact is intended for Category III missions, an inventory of bulk constituent organics is necessary if the probability of impact is significant. The applicability: Mars, Europa, Enceladus.

- Category IV involves missions where contact with the surface, atmospheric entry (if present), or landing on a celestial body occurs. This category is particularly crucial for environments with a strong interest in astrobiology, necessitating the use of a bio shield to protect hardware directly in contact with the materials. Category IV missions include probe and lander missions to celestial bodies of interest for chemical evolution and the origin of life. Scientific opinion indicates a significant chance of contamination, which could potentially jeopardize future biological experiments. To comply, detailed documentation is required, exceeding the Category III requirements. Procedures involve a bioassay to enumerate the bioburden, a contamination analysis, inventory of bulk constituent organics, and an increased number of implementing procedures. These may include trajectory biasing, cleanrooms, bioload reduction, possible partial sterilization of direct contact hardware, and a bioshield. Generally, the requirements and compliance are similar to the Viking mission, except for complete lander/probe sterilization. The applicability: Mars, Europa, Enceladus.

- Category V applies to missions involving the return of components to Earth, presenting the most dangerous scenario. For instance, when returning Mars samples, the container holding them must be completely isolated from the rest of the spacecraft to prevent contamination. Verification procedures are conducted to ensure the container is fully sterilized, free from particles, or other non-terrestrial elements. Category V missions encompass all Earth-return missions. The primary concern is the protection of the terrestrial system, Earth, and the Moon. Measures must be taken to prevent back contamination of the Moon, maintaining the freedom from planetary protection requirements on Earth-Moon travel. For all other Category V missions, under a subcategory known as "restricted Earth return," strict containment measures are imposed. Destructive impacts upon return are prohibited, and all returned hardware that directly contacted the target body or unsterilized material from the body must remain contained. Any unsterilized sample collected and returned to Earth requires timely analysis under strict containment and sensitive techniques. The applicability: Restricted: Mars, Europa, Enceladus. Unrestricted: Venus, Moon, S-Type Asteroids, Io, Mercury. Unrestricted: No indigenous life form is expected.

2.10 Question

a) Discuss how the environment, in its different aspects, affects the space system design
b) Describe the effect of the high energy particles on the satellite systems
c) Ionospheric plasma
d) Criticalities of vacuum
e) Which are the effects of atomic oxygen on space vehicle parts
f) Criticalities of the microgravity imposes on space vehicle and components
g) What is the SEU? How do you counteract it
h) Effects of galactic cosmic rays
I) Define the van Allen belts, their properties, and their effect on space segments
l) What is the South Atlantic Anomaly and how does it affect space segments
m) What are the environmental aspects to take into account for a satellite in an LEO orbit
n) Briefly summarize the main points of attention drawn by the environment for an interplanetary mission

Chapter 3

Propulsion Subsystem

3.1 Introduction

Onboard spacecraft, there are two types of propulsion units: the primary propulsion and the secondary propulsion, each serving different sets of maneuvers. The primary propulsion is designed to handle critical maneuvers, such as changes to the center of mass, orbital adjustments for reaching the final orbit, plane changes, and modifications of Keplerian parameters of the current orbit. For these maneuvers, chemical bi-propellants are typically used for interplanetary trajectories, while electrical grid engines (electrostatic) or hall effect thrusters are employed for electrical propulsion.

- The **primary propulsion** unit also caters to tasks like landing on celestial bodies with significant gravity, such as planets. Depending on the landing site, it can use bi-propellants or, in some cases, monopropellants. On the other hand, station-keeping maneuvers involve controlling orbital perturbations to keep the spacecraft within designated regions or "boxes." For these maneuvers, the secondary propulsion system comes into play. It usually relies on monopropellants or electrical solutions, providing a lower level of thrust and impulse compared to the primary propulsion.
- The **secondary propulsion** units are also responsible for executing relative maneuvers, such as formation flights, rendezvous, and docking around celestial objects with negligible gravitational effects. Additionally, they serve as the Reaction Control System (RCS), supporting reaction wheels and enabling the dumping of momentum or maintaining attitude control.

It's quite common to find spacecraft with a dual propulsion architecture, and there is no obligation to have the same type of propulsion for both systems. For instance, a spacecraft may use a combination of chemical and electrical propulsion, with one serving as the primary propulsion and the other as the secondary, or vice versa. The specific configuration depends on the mission requirements and desired performance characteristics.

3.1.1 DeltaV of maneuvers

Depending on the type of maneuvers you need to perform, there is a specific Delta-V (change in velocity) requirement, which will determine the class of thrusters you should choose for your spacecraft. For example:

- Insertion into Low Earth Orbit (LEO) requires Delta-V in the range of 0.001 m/s to 0.004 m/s.
- Transferring to inclined LEO orbit may need around 0.043 m/s.
- Deep space maneuvers can range around 388 m/s.
- Trajectory control maneuvers typically require around 7.0 m/s to 10.0 m/s.
- Geostationary Orbit (GEO) station-keeping may demand approximately 0.053 m/s per 7 years.
- Earth Wheel Station-keeping needs about 30 m/s per 7 years.
- Orbit-to-Orbit transfers for missions like "to Mars" can be around 90 m/s.

You can select the appropriate class of thrusters based on these Delta-V values. When consulting with the team responsible for defining the mission phases and their duration, you can choose one specific solution or multiple solutions, tailored for each phase of the mission.

3.2 Types of propulsion

- Chemical propulsion systems involve several components, and the size of the subsystem increases with the inclusion of these components. Key components include propellant storage, propellant feed system thrusters, valves, piping, and electrical control unit.
- On the other hand, electrical propulsion systems have a crucial element known as the power processing unit. This unit operates at very high voltage and handles the electrical conversion, control, and distribution of power to the thrusters. This power processing unit is the heaviest component in the electrical propulsion system, not the thruster itself. Additionally, there is the EPS (Electrical Power System), which is responsible for managing the power generated.

When comparing chemical thrusters to electrical ones, there is a significant difference in specific impulse (200-300s for chemical and 2000s for electrical). This results in an advantage in fuel mass for electrical propulsion, regardless of the type of fuel used. However, the trade-off comes in terms of power requirements. Chemical thrusters may only need 10-30W, while electrical thrusters require 500-1000W. This additional power requirement translates into mass for components such as solar arrays, batteries, RTGs (Radioisotope Thermoelectric Generators), and more. For instance, if we solely consider the mass of the thruster, a gridded electrical thruster would be much lighter compared to a bi/mono-propellant chemical thruster (only some kg). However, it's important to note that the electrical thruster requires a power processing unit (PPU), which weighs around 6kg per power density. This PPU's mass is quite significant and can outweigh the thruster itself. In conclusion, it's crucial to consider the entire system as a whole, not just individual components like the thruster. Taking the complete system into account ensures a more accurate and informed evaluation.

3.3 The propulsion system anatomy

The propulsion technologies covered in the course are chemical and electrical systems, with the latter being a viable alternative to the former due to its high Technology Readiness Level (TRL). It is a misconception that electrical solutions are not available for small satellites; they do exist, albeit with some lag in development.
Chemical propulsion employs high-mass particles with low velocity, while electrical propulsion uses low-mass ions with very high velocity. The former is limited by the energy it can provide, while the latter is constrained by the availability of electric power required to accelerate ionized particles.
For chemical propulsion, freezing temperatures of propellants and temperature control for tanks are essential considerations. Tanks are typically covered with Multi-Layer Insulation (MLI) to decouple them from the rest of the system and maintain stable temperatures. The hypergolic nature of the propellants influences the energy required for use. Chemical propellants exhibit requirements, like energy ones (high reaction heat and low molecular weight) and physical ones (low freezing temperature, high density (resulting in low volume), high thermal conductivity, low viscosity, low luminosity of exhausted gases, low toxicity, cheapness, hypergolic and chemical storability. In the case of military spacecraft, monitoring exhaust gas temperature is crucial to avoid traceability, especially for solar panels that should not be highly reflective. Lower exhaust gas temperatures ensure reduced detectability during maneuvers. Depending on the application, the requirement might be to remain non-traceable or highly traceable in radar imaging.

3.3.1 Chemical propulsion

Solid propulsion

Solid propulsion is not widely used in space, except for launchers. It is sometimes utilized to provide an extra boost to close an orbit after an interplanetary transfer. Solid thrusters offer low impulse (around 200s) and high thrust, making them suitable for delivering impulse in a single burn. However, solid propulsion systems have limitations. They do not support multiple burns, restarting, or throttling. The thrust profile depends on the configuration of the propellant grain, which can result in significant changes in the planetary insertion thrust, making them unsuitable for precise in-orbit maneuvers. Solid thrusters are not well-suited for GEO apogee burns, especially for spacecraft applications where restart is not required. Additionally, they are not suitable for re-entry maneuvers, as maintaining tight control over the flight path angle and velocity vector is challenging. One significant drawback of solid propulsion is that it requires a separate subsystem on board. For example, if used for deorbiting, it needs an entire dedicated box, which can be impractical and inefficient in terms of space utilization.

Cold gas

Cold gas propulsion is a straightforward system that utilizes pressurized gas, which needs to be over-pressurized for proper fuel usage. It is the simplest and cheapest propulsion option with relatively low effectiveness. The propellants used are compressed inert gases like nitrogen or high-pressure hydrocarbons such as propane. The cold gas propulsion system has some limitations. It offers very low impulse (around 40-60s) and extremely low thrust (around 10mN). The thrust magnitude is not adjustable (no throttling) and the system allows for multiple starts and pulsing. It finds applications in attitude control, fine control, and is suitable for nano-platforms. From an engineering perspective, it is crucial to position the tanks in a way that they are easily reachable from the feeding lines. The tanks and feeding lines need to be quickly accessible, as they are filled up just before launch and may need to be emptied if the launch window is postponed. For a classical spacecraft, the tanks are sized and thrusters are selected, while for cubesats, the entire propulsion package is usually predetermined. Some companies offer quantized tanks, which come in different sizes to suit the mission requirements. The power needed for the thruster's functioning is evident from the slide. It is essential to consider the life cycle for which the thruster is qualified. If used beyond its intended life, reliability and feasibility may decrease. To extend the lifetime, multiple thrusters can be mounted and used in sequence, as seen in the case of BepiColombo, which employs electrical thrusters to match the mission's duration.

Monopropellant

Monopropellant thrusters offer an ideal combination of specific impulses, making them suitable for secondary propulsion and, in some cases, even for primary propulsion if plane changes are not required. They provide a large thrust range, typically ranging from 1N to 20N or more, making them versatile and easy to integrate into the spacecraft's architecture. One of the advantages of monopropellant thrusters is their restart capability and throttling ability, which allows for precise control during maneuvers. In contrast, bi-propellant thrusters, especially European ones, may face limitations in throttling down to lower thrust levels, potentially causing damage during continuous thrusting, particularly during landing maneuvers. To address this, nozzle control, inclining the thrusters, or using multiple thrusters differently can be employed. The feeding of monopropellant thrusters is achieved through pressurization, either using an exogenous gas or the same gas in hypercritical conditions within the same tank. However, the thrust profile is not constant, and as fuel is used, the thrust decreases due to decreasing pressurization pressure. This leads to a lower end-of-life thrust compared to the initial thrust. When sizing monopropellant thrusters and other subsystems, it is essential to consider the end-of-life conditions when the tank is nearly empty rather than focusing solely on the beginning of life. Additionally, these thrusters may require a relatively high power for warming up the catalytic bed, which enhances their efficiency. This trade-off needs to be carefully evaluated in conjunction with the electrical propulsion subsystem onboard. Hydrazine is a commonly used monopropellant due to its stability against impacts, making it suitable for launch conditions. However, it is toxic, which may be a drawback, and other considerations like green propellants may be explored for alternative choices. A regression curve can be employed to estimate the thrust required based on the fuel mass, tank mass, and other relevant parameters, providing rough numbers for propulsion system sizing.

−**Architecture** The propulsion system consists of a tank with pressurizing gas and fuel. To ensure redundancy and robustness, the tanks are sized first, and then suitable tanks are selected from the market. This approach may not be applicable for cubesats. The number of thrusters needed depends on the application, such as attitude control, etc. Redundancy is achieved by splitting the system into two parts, even if not required, to have multiple sources or storages with cross-feeding lines. This design ensures that there is no single point of failure, providing an added layer of reliability. When selecting a monopropellant thruster, it's essential to consider factors like required power, specific impulses, and life cycles (measured in the number of pulses). These thrusters come in various thrust levels, making them suitable for tasks such as station keeping, relative dynamics, and deorbiting. They find application in a wide range of missions (see slides for cited mission examples). The first monopropellant thrusters for small applications appeared on the market, featuring five nozzles for attitude control. Available in different sizes based on propellant needs, they are chosen to align with specific requirements and availability. While the level of thrust and total impulse is well-defined for the intended application, these thrusters may have limitations in terms of flexibility in time and acceleration authority. Nonetheless, they still offer valuable functionality, making them a better choice than having no propulsion capability at all.

Bipropellant

Bipropellants are the most complex type of propulsion, offering high impulse burns (oxidizer and fuel combined) with a low impulse (pulse) when each propellant is used separately. A dedicated pressurizing system is required to ensure uniform pressures for both oxidizer and fuel, providing a constant thrust until the end of the burning process (no for monopropellant). With this architecture, it becomes possible to feed different thrusters for various applications, such as primary propulsion and station keeping, without the need for separate pressurization lines. By using this approach, the advantages of both systems are retained, including the ability to restart and potentially achieve throttling for the monopropellant thruster. The main driver for selecting this bipropellant system is not solely the specific impulse but having tanks of the same size, which facilitates the overall configuration and because it is preferred to have the same pressure in the tanks. Therefore, the MR may not be chosen in order to obtain the maximum performance, but rather to ensure consistent tank sizes, easing mass distributions and inertia matrix considerations for the spacecraft. When sizing the bipropellant solution, MR values of 1.6 and 1.3 are typically used as classical well-known numbers for evaluation. The following graph shows how the mix ratio (MR) evolves with respect to the performances. It's important to note that from a system perspective, the design may not be pushed to its maximum as it would be for propulsion optimization. Other factors, such as mass changing during flight and attitude control, are also significant considerations in the overall design process. For satellite missions, the cryogenic solutions (keeping tanks at very low temperatures) are not preferable due to the impracticality of maintaining such low temperatures during the entire mission. Instead, the lower two boxes in the presented chart containing fuel and oxidizer options are commonly adopted.

3.3.2 Electric propulsion

In the realm of Electric Propulsion (EP), the acceleration of particles falls into three distinct categories:

- Electrothermal: In this category, ions are accelerated by utilizing the thermal energy present in the system.

- Electrostatic: This category encompasses various gridded EP solutions, where ionized particles are accelerated using electric fields generated from the propellant.

- Electromagnetic: Going a step further, this category utilizes both electric and magnetic fields for ionization and acceleration of particles.

The thrust-to-weight ratio (T/W) provides an estimate of the control authority a system possesses. Here are the typical T/W order of magnitude values for each category:

- Electrothermal ($T/W < 1E-3$)
  Types: Resistojets, Arcjets
  Principle: Thermal-mechanical energy exchange

- Electrostatic ($T/W < 1E-4$)
  Types: Gridded EP, Field Emission EP
  Principle: Electric-mechanical energy exchange

- Electromagnetic ($T/W < 1E-1$ to $1E-5$)
  Types: Hall Effect (magneto-static), Pulsed Plasma (PPT)
  Principle: Magnetic-mechanical energy exchange

Electric thrusters usually require high voltages on the order of kilovolts (kV). This necessitates careful consideration of the mass budget, power budget, and volume budget concerning the entire system, including the EPS (Electric Power Subsystem). Hall Effect thrusters, while less efficient than ion engines, require less power, making them a topic of significant interest in recent times. Regarding future developments, magnetic plasma dynamics, though partially tested, have not yet been implemented in actual missions. In EP, the power compartment is a crucial building block that involves sizing solar panels, batteries, and the electronics system managing power processing for the thrusters.

Energy conversion efficiency

Energy conversion efficiency in electric engines depends on various factors, and its calculation varies between chemical and electro-based thrusters. For chemical propulsion, you work with temperature (T) and molecular mass (M) alongside the specific heat (Cp) of the fuel. On the other hand, electro-based thrusters deal with voltage (V) and electric charge (Q) to achieve high exiting velocities, often tens of thousands of m/s.
The conversion efficiency (T$-\eta$) for a given electric engine performance can be expressed in different way depending on Electrothermal thruster and Electrostatic thruster. In GEO satellites, Electric Propulsion Systems (EPS) are commonly employed for attitude control or station keeping. However, their use as primary propulsion for trajectory control is a more recent development. A multidisciplinary approach is required for sizing these systems effectively. The graph shows the mass with respect to the exiting velocity, which is influenced by the charge and voltage. It is essential to find the optimum solution considering the multivariate nature of the problem, as there exists a breakeven point where different factors converge to determine the most efficient setup for the system.

Electrothermal

In this category, we find arcjets and resistojets, primarily used for station keeping and attitude control. These propulsion systems offer a higher impulse compared to chemical propulsion but have lower impulse compared to electric propulsion.
$-$**Arcjet**: Propellant is heated by creating an electric arc between the anode and cathode (efficiency, n, is typically 65-85%). It provides a low thrust level of 0.2-0.3 N and specific impulse of 100-400s. Suited for attitude control systems.
$-$**Resistojet**: Propellant is heated using a resistance (efficiency, n, is less than 50%).

Offers a low thrust level of 0.1-0.3 N and high specific impulse of 500-1500s. Suitable for station keeping. The propellants used as a conductor between two electrodes create intense electric arcs, exciting the propellant into superheated plasma. Superheated plasma can be aggressive against structures, and the temperature is limited by the thruster case. Both arcjets and resistojets have similar thrust levels and are well-suited for their applications. Arcjets utilize an electric arc to heat the propellant, while resistojets employ resistive heating. Common propellants include Hydrazine, Nitrogen, and Ammonia, with low molecular mass and high specific heat preferred. For these propulsion systems, it's crucial to consider power and voltage requirements, with resistojet typically requiring higher values than arcjet. Additionally, lifetime considerations must align with the mission requirements.

Electrostatic

Electrostatic propulsion harnesses electric fields to first ionize the propellant and then accelerates the ionized particles using an electrostatic field.

- Ionization Mechanisms
  - Electron discharge along a magnetic field in a cylindrical chamber.
  - Electron bombardment with noble gas flow.
- Ion Acceleration
  - High potential (kV) electric field between two perforated grids accelerates ions through an electrostatic field.
  - Exhaust velocities reach ¿104 km/s in the ion plume.
  - An outer grid is used to reduce ion exhaust velocities and prevent back-contamination.
- Ion Plume Neutralization
  - A hot hollow cathode outside the engine emits electrons and low neutral gas flow.
  - Electrons discharged into the ion plume combine with ions to form neutrals, neutralizing the beam.

The classical method involves using an emitter of electrons inside the chamber. Electrons move helically inside the chamber, and upon injecting propellant, collisions result in ionization of the propellant. The ions, with higher mass and momentum, are of interest for acceleration. To accelerate the ions, a grid induces a high-voltage electrostatic field between the two grids. However, exiting with a charged particle leads to thrust loss, requiring neutralization of the beam upon exit. Neutralizers, along with the grid, determine the system's lifetime.
The relative short firing time of these thrusters is due to two main factors: grid erosion, resulting in larger holes and reduced efficiency, and the survival of neutralizers. Ionization methods include Electron Beams (Kaufman) (GIEs) with direct electron beam injection, RIT using coils around the chamber to ionize the beam, and Microwaves (ECR) utilized by the USA and Japan.
Performance Characteristics:

- Very high specific impulse: 3,000-10,000 s
- Very high thrust efficiency: $> 60\%$
- Suitable propellants: Xe, Kr
- Precise throttling capability
- Low ion plume divergence: 15° (half-cone)
- High voltage (kV) leads to high Power Processing Unit mass (6kg/kW)

Energy Sinks:

- Propellant ionization (W/A)
- Propellant acceleration (W/mN)
- Ion beam neutralization

Ionization can be achieved through Electron Beams (Kaufman) (GIEs) used by the UK/Russia, RIT with increased efficiency by Germany, and Microwaves (ECR) employed by the USA and Japan.
The benefits of the electrostatic propulsion are high efficiency and low pollution and limited plume divergence, so there is a benefit also in efficiency. The drawback refers to the erosion. Erosion poses a significant challenge in electrostatic propulsion systems, leading to performance degradation. There are two primary erosion factors to consider:

- Grid Erosion: This occurs due to high-speed ion impingement on the acceleration grid, a phenomenon known as "sputtering." Grid erosion remains a critical limiting factor for the system's lifetime. To mitigate this issue, selecting a grid material with a low thermal expansion coefficient (CTE) is crucial. The most effective solution currently is using boron-coated molybdenum, which exhibits a low CTE, effectively balancing all aspects and limiting erosion while extending the thruster's lifetime.
- Secondary Erosion: Cathode and neutralizer erosion also contribute to the reduction in thrust efficiency and specific impulse due to hole enlargement and plume defocusing. When selecting materials for closely-spaced, curved grids, several criteria are considered, including:
  - Low thermal expansion coefficient (CTE)
  - High mechanical strength to withstand launch loads
  - Low sputter yield

  Different materials offer various advantages:

  - Molybdenum: It has a low CTE and high strength but exhibits a very high sputter yield, limiting its life to a few thousand hours only.
  - Carbon-carbon: It has zero CTE and a low sputter yield, extending thruster life to a few tens of thousands of hours. However, its low strength makes it unsuitable for large thrusters.
  - Boron-coated Molybdenum: This material provides the lowest CTE, low sputter yield, and high strength, making it suitable for large thrusters.

In traditional grid-less thrusters, the criticality of grid erosion is avoided by employing an alternative method to accelerate the ions.

Field emission electric propulsion thrusters

Within the electrostatic domain, there are several electric thrusters designed for specific maneuvers, particularly for secondary propulsion rather than primary propulsion. One such example is the Field Emission Electric Propulsion (FEEP) thruster, utilized in missions like LISA (Laser Interferometer Space Antenna) and LISA Pathfinder for gravitational wave detection through interferometry. Key Features of FEEP Thrusters:

- Propellant: Liquid metal (cesium, indium, rubidium, mercury) with low ionization potential and preferred low melting point.
- Thrust is produced by accelerating liquid metal ions in a strong electric field.
- T-level (Thrust level) is low, ranging from 1 to 100N, making it suitable for fine attitude control.
- Instantaneous switch on/off capability.
- Power-to-thrust ratio: 60W/mN.
- High impulses: 6000-10000s, with an efficiency of 98%.

These systems do not have any electromagnetic elements that could interfere with the dynamics of the overall system. A notable example of such a system is the LISA Pathfinder with its FEEPFT-150 thruster. It utilizes a liquid propellant, cesium, with a tight edge design to facilitate splattering and capillarity effects. The high voltage level at the edge leads to material splattering, immediate ionization, and subsequent acceleration by the electric fields. One notable advantage is the capability to provide thrust at a micro level rather than milli, and the power requirement is minimal, making them suitable for precision and low-power applications.

Hall effect thrusters

The propulsion system converts electrostatic energy into an acceleration of charged particles using an electromagnetic field. The use of high voltage in avionics can significantly impact the overall weight of the system. It is important to consider the effect of this propulsion system on the sizing of the Electrical Power System (EPS), taking into account the specific requirements of the propulsion system. The system relies on electrical fuel for ionization, and the acceleration occurs through the electrostatic field at the end of the thruster with electrocyclic motion. An auto-induced electric field helps prevent degradation. Aging is a concern in these systems, particularly erosion in the central element of the thruster over time. The presence of a large plume can reduce efficiency, but it also helps protect the nozzle of the thruster from degradation. The advantage of this system is that it requires less power, resulting in lighter avionics, especially the power conversion and regulation unit. Ionization Characteristics:

- Electrons are generated by an external cathode.
- The specific impulse of this system is lower than Gridded Ion Engines (GIEs), typically in the range of 1,800-2,600 seconds. (Lower specific power, and lower voltage)
- Electrons are injected into a dielectric annular chamber and drawn towards an axial anode.

- The thrust efficiency is lower than GIEs, around 50-60% (90% ionization, 70% discharge) due to the presence of a radial B-field between the inner and outer poles of magnets.

- Lorentz force acts on electrons as they cross the radial magnetic field lines, resulting in electron cyclotron motion in the chamber, forming a helical path towards the anode.

- Neutral gas is injected into the chamber and collides with electrons during the ion acceleration process.

- Ions are accelerated by the self-established electric field created by the electron current induced by the Lorentz force.

- To neutralize the ion plume, additional electrons are emitted by the external cathode and drawn into the ion plume, where they combine with ions.

The performance degradation is related to:

- Primary degradation involves erosion of the ceramic insulation of the dielectric annular chamber due to ion impingement during acceleration, limiting thruster lifetime.

- Secondary degradation includes cathode and neutralizer erosion, which is addressed using redundancy.

It has to be considered that chamber ceramic erosion presents several challenges:

- Ion impingement on the high-temperature ceramic material during the acceleration process causes erosion.

- The erosion rate is high, necessitating a thick ceramic layer, which results in wider beam divergence and reduced thrust.

- Current materials used, such as BN/SiO2/Al2O3, have a thruster lifetime of only a few thousand hours.

- The Fakel SPT-100 is space-qualified for up to 7,600 hours of operation before the risk of ceramic failure.

- There is a need for a deeper theoretical, predictive, and experimental understanding of erosion in different materials to improve thruster performance and longevity.

Hall thrusters can be employed in parallel within a single system for two main reasons:

1) Increased Thrust: One thruster's thrust may not be sufficient, especially when dealing with millinewton-level requirements. By using multiple thrusters in parallel, the overall thrust can be enhanced to meet the mission's needs effectively.

2) Extended Lifetime: Operating multiple thrusters in parallel can also contribute to extending the operational life of the propulsion system. By distributing the workload across multiple thrusters, each individual thruster experiences reduced stress, leading to prolonged mission duration.

Moreover, Hall thrusters offer the advantage of being controllable in terms of angular orientation. This capability allows for precise thrust vector control, making them highly useful for applications that require specific pointing directions. When assessing the performance of Hall thrusters in comparison to chemical propulsion systems, it is essential to consider the entire thruster unit as a whole. These steerable plasma thrusters, like those developed by Safran, provide unique advantages in terms of efficiency and precise control. However, one challenge faced by Hall thrusters is their relatively low operational life duration, which might limit their suitability for longer missions. Ensuring an adequate operational life becomes crucial in these scenarios where extended mission durations are required.

Pulsed plasma thrusters

Pulsed plasma thrusters offer impressive features such as high specific impulse, low power, and minimal fuel requirements, making them ideal for pulsing applications in station-keeping (SK) maneuvers. The underlying principle involves storing energy in a capacitor, and an ignitor shoots electrons between the anode and cathode to discharge the capacitor and create an arc. This arc then evaporates and ionizes the solid fuel, resulting in acceleration out of the thruster through Lorentz forces induced by the electromagnetic field. Following each pulse cycle, the capacitor is recharged from a power supply, allowing the process to be repeated. These thrusters find effective use in situations requiring precise control, station-keeping, attitude control, and maneuvers for relative dynamics. They are particularly suited for small satellites. Typically utilizing Teflon as the solid propellant, the thruster incorporates a spring that pushes the propellant forward during pulsing. This action takes place within a high-energy electrostatic field between the two plates. Consequently, the arcs vaporize the propellant, leading to the gain of ionized fuel particles. These particles then move within substantial magnetic fields, influenced by Lorentz forces, propelling them towards the thruster's exit. In essence, the particles pass through the arc, and the induced magnetic fields generate Lorentz forces that drive the ionized particles to produce thrust. Notable features of pulsed plasma thrusters include:

- Non-toxic propellant.
- Low power demand (50-70W).
- High specific impulse of 650-1350s.
- Thrust in the range of 90-860 uN-s.
- Ability to power multiple thrusters with a single capacitor.
- Mass of 5-6 kg, which is suitable for small satellite applications.

3.3.3 Feeding system

The purpose of sizing the feeding system is crucial for a propulsion engineer. However, for a system engineer, the focus is not on sizing the nozzles and chambers, but rather on the feeding system itself. As depicted in this tree diagram, there are numerous potential solutions, but it ultimately boils down to two main approaches: blowdown or regulated pressure solutions, as highlighted in red.
Sizing the feeding system encompasses several key aspects:

- Determining the mass and volume of the pressurant.
- Sizing the tanks for both the pressurant and the propellant or oxidizer.

These considerations apply to both liquid propellant systems and electric propulsion systems, though the latter usually employs a single propellant. A challenge with liquid propellant systems is the potential formation of bubbles and gaseous pockets in the tanks. This must be avoided, especially when the feeding line directly supplies the thrusters. Several approaches can address this issue:

- Spinning (spin-stabilized spacecraft): This method ensures that the fuel does not contain bubbles in the region feeding the thrusters. However, it requires relatively high spin rates (¿6 rpm) and may not always be preferred.
- Three-axis stabilization: In this approach, elements like sponges can be utilized to collect the liquid part in the region feeding the chamber. Capillary-based lines can also be used. This option involves directing the liquid directly towards the thrust. It can leverage capillary devices, diaphragms, bladders, or bellows to achieve the desired effect.

Selecting a tank involves considering one of these options and identifying an appropriate solution. Implementing such devices reduces the volume of propellant in the tank, so a margin is typically taken into account during sizing, around 5% in volume.
Specific solutions for the blowdown system include the use of membranes (physical separation between the pressurant and the fuel). Even with a membrane, volume is occupied and deducted from the propellant and pressurant. Clever solutions like diaphragms with different spherical sections can be employed to accommodate propellant deflation.
The pictures of the tanks represent classical options available in different shapes and sizes, with consideration for attachment mechanisms to other structures. The two possible mechanical interfaces are polar and equatorial, and this decision influences the overall configuration, making it a multidisciplinary consideration involving system and configuration engineers.

3.3.4 Pressurization system

The main goal of pressurization systems is to control the gas pressure in the tanks to ensure the correct pressure in the chamber during propulsion operations. There are two primary solutions for pressurization: blowdown systems and regulated systems. The pressurant gases used are usually Helium or Nitrogen, each with its own advantages and drawbacks. Blowdown systems are typically used in cases where:

- Only a single propellant (monopropellant) is employed, not a combination of oxidizer and propellant.
- Constant thrust is not required.

With blowdown systems, the pressurant's pressure decreases as the propellant is expelled, leading to a reduction in the ability to maintain a constant feeding pressure. Consequently, the thrust at the end of the mission will be significantly lower than at the beginning, making this impact more noticeable in long-duration missions. To mitigate this effect to some extent, multiple tanks can be used. If such flexibility is not possible, a regulated

pressure system would be a more suitable option, especially when precise thrust control is essential. However, it is crucial to avoid using the blowdown system with bipropellants. Here are the pros and cons of the blowdown system:
Benefits:

- Simple and reliable.
- Cost-effective

Cons:

- Pressure, thrust, and mass flow vary over time.
- Not applicable for bipropellants.

For sizing the system, the pressure range for the propellant tanks is typically set between 1.3 to 9 MPa (13 to 90 bar).

Sizing Blowdown system

To determine the proper sizing for the system, certain calculations need to be performed. The blowdown ratio is a fixed parameter within the range of 4 to 6. The usable volume occupied by the propellant at the initial status is known based on the deltaV. Normally, calculations assume slow flow, resulting in an isothermal evolution over time. However, if quick or impulsive maneuvers are anticipated, the variation may be isentropic (quick flow). Regarding tank sizing (both pressurant and fuel) the target is to determine the thickness of the tanks and the overall mass that needs to be included in the mass budget. Calculations are conducted to identify the ideal case, and then this is compared to available devices and equipment to reassess the final decision. The chosen solution will influence the fuel quantity and the maneuver considerations for the mission.
So the steps to follow are:

- Calculate the propellant mass using the equation
- Compute the propellant volume, taking into account a 3% margin for unusable propellant.
- Determine the volume of the bladder or diaphragm, which is typically 1% of the tank volume.
- Calculate the total tank volume
- Compute the tank mass
- Calculate the pressurant mass using the equation of state, assuming a typical initial pressure of 3-5 MPa.

Additional notes on the mentioned steps: the propellant mass can also be computed based on the deltaV, which represents the change in velocity required for the mission. The propellant margin is necessary to account for propellant that remains in the tank or gets trapped in capillary elements or sponges. Note that the pressure values are given in MPa, not bar, for consistency.
For the tanks:

In general, tank cost accounts for around 1% of the propellant mass. For liquid systems, the entire system is estimated to be 15% of the propellant mass. Next, determine the type of tank and its configuration based on mission requirements. Consider the maximum pressure inside the tank, which should be higher than the pressure in the combustion chamber. Select an appropriate material for the tank based on the mission requirements and available options. Calculate the tank thickness and its volume based on its shape (cylindrical or spherical).
Finally, compute the mass of the tank based on its volume and material density. Also, check if there are existing off-the-shelf tanks that meet the mission requirements.
Additional notes on the mentioned steps: first, you select the configuration of the tank. The spherical configuration is typically preferred initially due to its mass advantages. However, depending on the chosen mission requirements, you may opt for a cylindrical tank with a hemispheric cut to accommodate specific configurations. The maximum pressure inside the tank is usually set around 10% higher than the pressure in the combustion chamber. Additional equations are considered to account for any potential pressure losses. he tank material options are typically Titanium or Steel. However, if the mission takes place around Earth, Titanium may not be used as it cannot be disposed of easily (similar to batteries) due to its high melting temperature.

Sizing regulated pressure system

In the regulated pressure system architecture, the pressure level is significantly higher, typically one order of magnitude greater, reaching 200-400 bar (20-40 MPa) in a single tank. The steps for tank sizing in this type of system are outlined below:

- Compute the oxidizer and fuel volumes and masses based on the chosen mixture ratio (MR). The aim is to achieve an equivalent mass for both tanks for structural and inertial considerations.

- Account for propellant mass that may not be used, and increase the masses by 1-3% to accommodate ullage.

- Select the number of tanks and the materials for the tanks. Initially, one tank is usually chosen and can be adjusted later as needed.

- Fix the oxidizer and fuel pressures, considering losses.

- The pressurant pressure should be approximately one order of magnitude higher than the pressure in the main tanks, typically (10-100 bar).

- The pressurant gas options are typically helium or nitrogen. While helium is lightweight, it poses leakage problems, so alternative gases may be preferred.

- The pressurant volume is computed based on the conservation of energy, assuming adiabatic flow. Once the type of pressurant is determined, the volume can be calculated.

These steps help in sizing the pressurant and oxidizer/fuel tanks to meet the mission's propulsion requirements.

Losses

The tank pressure must meet the requirement of being equal to or higher than the pressure in the combustion chamber, considering all the line losses. Pressure losses at the tank exit can be calculated using the formula: Tank Pressure = Chamber Pressure + Pressure Drop due to feeding + Pressure Drop due to Inlet.
To ensure manageable feed system losses in the range of [35-50] KPa, flow velocities are typically maintained at around 10 m/s. The critical factor driving tank pressure is the combustion chamber pressure. While feed system and injector losses also play a role, their impact is relatively smaller, especially when dealing with bipropellants. The losses here is 0.2/0.3 the chamber pressure.
Using a flow magnitude of 10 m/s is a suitable approach for estimating feed system losses in the propulsion system.

3.3.5 Electric power

− **Electric Orbit Topping (EOT) Graph**: The X-axis represents the dry mass of the platform without propellant, representing the system as it is. The Y-axis represents the launchable mass. The graph demonstrates the effectiveness and advantages of using electric propulsion for topping the semi-major axis. It shows an almost linear trend, indicating the reliable performance of electric propulsion solutions for Earth applications, as well as other scenarios.
Electric Orbit Topping refers to the process of increasing the apoapsis (semi-major axis) using electric propulsion. The four lines on the graph highlight how we heavily rely on low-thrust energy exchange to navigate within the gravitational field, progressively increasing the energy levels. By keeping the dry mass constant at 3500, the usage of electric propulsion leads to reduced launch mass. Starting from a lower altitude allows for less fuel consumption compared to chemical propulsion. On the other hand, if the launcher mass (around 6000 kgs) is fixed, utilizing electric propulsion saves dry mass. The more we employ electric propulsion, the more we can allocate the saved dry mass (previously used for fuel) to other critical components like antennas, radars, and instruments. The gained mass can be substantial, often measured in tons, which is of great significance, especially considering that geostationary satellites typically weigh around 7-8 tons.

−**Power Supply Evolution**: The ongoing trend in space propulsion systems indicates a significant shift towards electrical and low-thrust solutions, which have now become the standard for center of mass control, trajectory adjustments, and orbit maneuvers. This marks a notable departure from the practices of 10-15 years ago. In terms of power supply, there has been a steady evolution in GEO commercial platforms. The comparison of various systems can be summarized as follows: in 1975, the power systems were limited in capability and efficiency. Today, the electrical and low-thrust solutions have become the baseline, offering better control and maneuverability with reduced power demands and lighter components. Hall effect thrusters are less efficient but require lower power, making them more lightweight than gridded systems. However, gridded thrusters are highly efficient with a closed plume, although they demand more power and are sensitive to aging.

−**Architecture**: When it comes to the architectural point of view, fuel storage and fuel lines are treated similarly to monopropellant systems. Key elements are positioned at the bottom of the configuration, ensuring effective fuel management.

−**Power Processing Unit (PPU) Configuration**: In the current architecture, one Power Processing Unit (PPU) can manage a maximum of two thrusters. This design approach optimizes the use of onboard mass. Therefore, there can be a maximum of two thrusters in this configuration.

−**Flux Control**: For flux control, each thruster requires a dedicated unit and cannot be shared with more than one thruster. This individualized setup ensures efficient and precise control of each thruster's flux.

3.4 Design process

In the design process, collaboration with mission analysis personnel is essential, and careful attention must be paid to the Attitude Determination and Control System (ADCS) part. Apart from the Delta-V budget, which is a crucial factor, understanding the authority of the propulsion system is vital. This includes assessing whether it will be used for maneuvers or trajectory control within a specific time frame. For instance, rapid and timely maneuvers where Delta-V is considered in an integral sense, rather than focusing on individual accelerations needed for the maneuvers. The following steps should be followed:

- Identify propulsion requirements based on:
  - Delta-V budget per phase and mode (orbit and attitude).
  - Time required to perform maneuvers and duty cycles.
  - Level of thrust needed.
  - Type of thrust, whether steady state or pulses.
- Select propulsion system components, which involves choosing:
  - Type of engines and propellant.
  - Feeding system.
  - Tanks.
- Determine the propulsion system sizing and architecture, considering factors such as:
  - Number of engines and control units.
  - Number and dimensions of tanks.
  - Redundancy in the system architecture.
- Size the tanks accordingly.
- Create the propulsion system schematic and establish the propulsion system budgets, including mass, power, and data.
- Prepare the equipment list.
- Finalize the propulsion system configuration.

When seeking a propulsion system with a large propellant inventory authority, the chemical option is the obvious choice. For situations where short burns or pulsing maneuvers are required and authority and timeliness are less critical, the electrical solution becomes a viable option. Typically, the mission analysis influences the selection or provides inputs for the propulsion architecture design, but it's not an absolute requirement. In some cases, limitations in size or mass may impact the propulsion subsystem, thus influencing the mission analysis. Various components play essential roles in the propulsion system, including feeding systems (tanks, valves, filters, lines, sensors, and heaters) and thrusters. Ensuring proper loading accuracy and managing uncertainties (e.g., trapped propellant, loading uncertainty) is crucial for maintaining system efficiency and performance. Ultimately, the choice between chemical and electrical propulsion depends on mission-specific requirements and constraints.
In the initial design phase (Phase A), the propulsion system selection process involves choosing the type of propulsion (chemical or electric), the specific thruster configuration (e.g., bi-propellant, mono-propellant, oxidizer + fuel), and determining the size of tanks and feeding lines. For spacecraft (not launchers), the propellant delivery to the chamber is achieved through pressurization (Delta-P) to feed the fuel and oxidizer. Similar sizing and configuration considerations apply to electric propulsion systems. Proper sizing of pressurizing tanks and strategizing the feeding process are essential components of this phase. The number and positioning of tanks play a significant role in managing the spacecraft's matrix of inertia, which can vary throughout the mission as the mass changes. Outputs of this sizing process include the number of tanks, identification of pressurization tanks, feeding lines (with redundancy to ensure system functionality in case of valve malfunctions), and inputs for mass and power breakdown. The propulsion subsystem's power budget is critical, not only for electrical systems but also for chemical systems, as it accounts for actuation of valves, sensors, and the warm-up of the catalytic bed if applicable. This information needs to be communicated to the on-board data handling, telemetry, and telecom experts to properly allocate memory space and data rates for these measurements. For sensor-related inputs, one must consider the telemetry requirements and data resolution needed to monitor temperature, valve states, mass variation, and pressure changes along the feeding lines. These measurements need to be quantized in bits, leading to memory and data handling sizing. As the design progresses from the initial concept to a detailed subsystem, the ancillary information must be passed to the On-Board Data Handling (OBDH) system. Accurate data collection and transmission are vital for effective spacecraft operation, making it essential to consider these aspects when sizing and designing subsystems.

3.4.1 Propulsion Subsystem Mass Margins and Estimates

When sizing the propulsion subsystem, it is crucial to account for margins to ensure mission success. The propellant mass should be calculated based on the total dry mass at launch and the total delta-v required for the mission.
Here are the margins and estimates to consider:

- CHEMICAL Propellant:
  - Volume of the propellant shall be sized for the total propellant mass + at least 10 %.
  - Pressurant as in the margin document, shall be sized with 20% of margin

  - Feeding System (Tanks, Valves, Filters, Lines, Sensors, Heaters) and Thrusters: provide adequate margins in the sizing of the feeding system components and thrusters to handle variations and uncertainties during the mission.

- ELECTRIC Propulsion:

  - Electric Propulsion Impulse: Apply a safety factor of 1.5 on the total impulse, lifetime, and the number of cycles throughout the mission when sizing the electric propulsion unit.
  - Duty Cycle: Assume a 90% duty cycle when evaluating the performance of electric propulsion.
  - total propellant mass with at least a 5% margin to account for uncertainties and variations.

Margins are included as specified in the margin document. For tank volume, size the propellant tanks to accommodate the total propellant mass plus an additional 2-3% margin and at least 10% for the ullage space (empty space left for vaporization of the liquid) and loading uncertainty. By incorporating these margins and estimates, the propulsion subsystem will be better equipped to handle real-world conditions and variations, ensuring a successful and reliable mission.
A safety factor must be applied to the impulses, lifetime, and duty cycles of electric propulsion systems. This is because electric propulsion technology is still relatively new in space, and its nominal functioning duration along a mission is also limited. An essential characteristic of the electric propulsion subsystem is the certified lifetime hours, which can be found in the datasheets. This value represents the number of hours the thrusters have been certified to operate, based on testing either on the ground or in space, typically ranging from 10,000 to 20,000 hours. However, for certain missions like interplanetary applications or spiraling topping missions for high-altitude planetary orbits, these certified hours may not be sufficient. If the thrusters need to operate continuously for more extended periods, it becomes the responsibility of the mission planner to assess the risk associated with using the thrusters beyond their qualified duration. The mission's reliability and robustness can be compromised if the thrusters are operated beyond their qualified lifetime. To ensure reliability and robustness, it is essential to adhere to the number of qualified hours during the mission. If thrusters qualified in the lab are used, margins must be applied appropriately. It is essential to recognize that exceeding the qualified hours does not necessarily mean that the thrusters will stop working altogether, but rather, it introduces a level of risk that needs to be carefully evaluated, as it is a common limitation for many missions.

3.5 Question

a) Enounce and discuss the main drivers for the propulsion subsystem design and highlight the mission-dependent aspects which affect the design process
b) List of chemical thrusters possibilities and explain selection criteria
c) Explain the benefits and drawbacks of a cryogenic liquid propellant and indicate its specific impulse level. Describe the maneuver scenarios in which their application is beneficial
d) Difference between blow-down and pressure regulated system. Benefits and limit
e) PMD
f) Types of electrical propulsion. When it can be used and limits
g) Types of electrothermal propulsion. When it can be used and limits
h) Explain the power system equation of the ion thrusters and briefly speak on the most important variables
i) Classify the electrostatic propulsion systems and explain the differences
l) Explain the principle of a GIT thruster, its main performance, and its areas of application
m) Explain the principle of a FEEP thruster, its main performance, and its areas of application
n) What is a Pulsed Plasma Thruster? Describe its working principle, performance, and field of application
o) Describe the electromagnetic thrusters and specify their performances
p) PS margin
q) Final Comparison

Chapter 4

Telecom Subsystem

The TTM&TC (Telemetry Tracking Telecommand Subsystem) task is to establish connections between the space system and other systems. This connection facilitates the transmission of data from the satellite to the ground station, between satellites, or communication of onboard stored data, including scientific data, system status (telemetry), commercial signals (broadcasting/phone/TV), and commands. The Telecom system also handles tracking, which involves reconstructing the position vector of the center of mass of the system for navigation purposes. This is achieved through doppler shift or ranging, with deep space missions often using this method, while near-earth missions can utilize GPS. Key aspects to consider for this subsystem are:

- Direct visibility is required between the two ends of the system for effective communication.
- There are limited visibility time windows during which communication can occur.
- Weak signal strength may be an issue due to various environmental stresses such as atmosphere interference, launch shocks, radiation, and temperature.
- Antenna thermally sensitive, and there are limitations in volume, power, pointing, and mass.

Collaboration with the MA (Mission Analysis) head is crucial to determine visibility windows and select the appropriate Ground Station for communication. In interplanetary missions, precise location and timing of the window are essential to size the signal transmission and account for light−speed delay. It's important to consider the time needed for the link to become operative after the two terminals establish visibility. There is a delay before the satellite is identified and the channel is opened, so margins must be implemented to account for this delay. Weak signal strength can be a bottleneck due to power demands and data transfer requirements. Atmospheric interference is also a factor to be taken into account, and it's preferable to choose high-altitude and dry regions for the ground station location to minimize noise.
Also, keep in mind that a significant portion of the tools/parts within the system are sensitive to temperature, radiation, and other environmental stresses. It is advisable to maintain the system at a cool temperature or operate it under nicely controlled conditions. For instance, it is essential to avoid transmissions when the Sun is in view, as it can cause significant noise and reception issues. A possible solution is to position the antenna in a shadow location to mitigate these effects.

The TTM&TC subsystem collaborates closely with other subsystems, as it relies on data taken and correlated from them. Therefore, it must effectively communicate and cooperate with all the other subsystems. Consequently, for the budget study of this subsystem, it is crucial to consider the system's configuration, both internal and external volume availability, since multiple antennas are often required. For instance, a high-gain directional antenna may be necessary to achieve big data rates, while during the initial flight stages, an omni−directional antenna with low data transfer capability may be used for tasks like detumbling operations. As always with spacecraft, mass limitation can be a critical requirement, especially if long distances and high power are involved. The image demonstrates that communication links can occur in any phase and under various conditions. Therefore, as mentioned before, it may be necessary to relay data between satellites if a direct connection to a ground station is not feasible.

4.1 Data transmission

In communication systems, data transmission can occur in two directions: the Space segment and the Ground segment. Data transfer might occur through direct/transposed links. Downlink and Uplink might occur with same or dedicated different ground station. Two Communication Directions:

- Space Segment to Ground Segment (Downlink)
  - Engineering data type: This includes measurements such as temperatures, pressures, voltages, currents, wheel speeds, rotational positions, and outputs from attitude control system (ACS) sensors in both analog and digital formats.
  - Status measurements: Positions of valves, switches, and relays, as well as safe/arm positions, are transmitted in digital format.
  - Science payload data type: Both analog and digital data from the science payload can also be downlinked.

  The downlink transmission, from the spacecraft to the ground station, is more common and involves various types of data. However, it's essential to note that the data can be broadly classified into two categories: Analog and Digital. Currently, the trend leans more towards digital data, and if there are any analog data stored, they need to be digitally converted before transmission.

- Ground Segment to Space Segment (Uplink)
  - Commands: This involves sending commands from the ground to the spacecraft for various purposes, including activity planning for short, medium, or long-term missions. It informs and updates the spacecraft on actions to be taken within a specific time window.
  - State updating: Uplink commands can also be used for updating the spacecraft's state. The uplink transmission, from the ground to the spacecraft, is less frequent than the downlink, but it is essential for planning and updating the spacecraft, especially in non-nominal situations or for receiving software patches and managing failures. Even if a spacecraft is designed to be autonomous, planning for uplink possibilities is crucial for such situations.

Regarding the sizing aspect, the chart below illustrates the signal elaboration process through a block diagram architecture:

- Data Source: Internal sensors in the spacecraft, including those for the module, payload, and scientific data.
- Analog to Digital Conversion: This step is fundamental for converting analog data into digital format, although it may not always be needed.
- Encryption (for classified missions): Data may be encrypted for military or classified purposes.
- Multiplexing: Multiple channels may be combined and processed together.
- Data Storage: Data storage is sometimes used to retain information until a visibility window opens.
- Encoding: Data is processed to protect against errors. Control bits are added to identify errors during reception.
- Transmit Block: Signal modulation and transmission occur during this stage. The data is modulated to higher frequencies, amplified, and then transmitted to ensure sufficient power for effective communication.

4.2 Sampling and digital coversion

The initial step in the analysis is the sampling and digitalization process. Often, Analog−to−Digital (A/D) conversion is required to send data to Earth.

4.2.1 Sampling

Sampling is the initial phase of the analog−to−digital conversion process. In this process, the continuous analog signal over time is discretized by taking samples at regular intervals. These samples represent snapshots of the analog signal at specific points in time. The goal of sampling is to capture enough information from the analog signal so that it can be represented in digital form without losing significant detail. Here's how sampling works:

- Sampling Interval: The decision is made about the time intervals at which samples will be taken from the analog signal. This interval is known as the sampling interval and is expressed in terms of the sampling frequency (the number of samples per second, measured in Hertz). Nyquist's sampling theorem states that, to avoid aliasing (undesirable frequency overlap), the sampling frequency must be at least twice (2.2) the maximum frequency present in the analog signal (Nyquist-Shannon theorem).
- Sample Acquisition: At regular time intervals, the values of the analog signal are measured. These values are the samples of the signal and represent its instantaneous value at those specific moments. For example, if we are sampling an audio signal, the samples would represent the amplitude of the sound signal at regular time intervals.

4.2.2 Quantization

After sampling, there is quantization. Quantization is the process by which the sampled values of an analog signal are mapped to discrete values, often represented with a certain precision in bits. For example, you could have an analog signal that can vary in amplitude from -10 V to $+10$ V, and you want to convert it into an 8-bit digital signal, which means you would have 256 possible discrete levels (2^8). Quantization error occurs when a sampled value of the analog signal falls between two possible discrete levels, and it is represented by the nearest discrete value. This process introduces an error between the actual value of the signal and the quantized value. Quantization error is the difference between the sampled value and the quantized value. This quantization error can lead to a loss of underlying information in the original analog signal, as differences between the sampled and quantized values can accumulate and impact the overall accuracy of the digital representation of the signal. The formula is:

$$Quantisation_{error} = \frac{M_{max} - M_{min}}{2N} \tag{4.1}$$

In general, quantization error can be reduced by increasing the resolution of quantization (i.e., using more bits to represent the values), but this will also increase the computational cost and the amount of data needed to store the converted signal. For instance, let's consider monitoring battery temperature with an interval of -10°C to 60°C and an accuracy of 1°C. This means you want to represent 70 distinct temperature values effectively. At the and there is digital representation: the quantized values are then represented in digital format using a certain number of bits. For example, a sampled and quantized audio signal could be represented using a specific number of bits for each sample. Determining the number of bits required to represent these values accurately is essential information to communicate to the TTM&TC (Telemetry Tracking Telecommand) team, as it influences the amount of data to be transferred for this telemetry.

4.3 Channel capacity and datarate

4.3.1 Channel capacity

Channel capacity refers to the maximum rate at which information can be reliably transmitted over a communication channel without errors. It represents the upper limit on the data rate that can be achieved over the channel while maintaining a specified level of error performance. The concept of channel capacity was introduced by Claude Shannon, a pioneer in information theory. Channel capacity is affected by various factors, including the bandwidth of the channel, the signal-to-noise ratio (SNR), and the presence of any interference or noise. The channel capacity theorem, known as Shannon's theorem, states that the maximum achievable data rate (in bits per second) over a noisy channel is determined by the channel's bandwidth and the SNR:

$$C = B \log_2(1 + SNR) \tag{4.2}$$

Where: C is the channel capacity (in bits per second), B is the bandwidth of the channel (in Hertz), SNR is the signal-to-noise ratio. The equation indicates that increasing the bandwidth or improving the signal-to-noise ratio will result in a higher channel capacity, enabling higher data rates.

4.3.2 Datarate

Datarate, also known as bit rate, is the rate at which data is transmitted over a communication channel. It represents the number of bits transmitted per unit of time (usually measured in bits per second, or bps). Data rate is a practical measure of how quickly information can be exchanged between sender and receiver. The data rate achievable in a communication system depends on several factors, including the modulation scheme used, the bandwidth of the channel, and the encoding technique. Higher data rates typically require more bandwidth and better signal quality. To be error free the datarate R has a superior limit imposed by the channel capacity (Shannon's Theorem), so $R \leq C$. That becomes a limit in terms of signal to noise to ensure the desired B: $SNR = \frac{S}{kTB}$. So given signal power S and constant noise energy kT, the maximum error free transmission rate can only be achieved with unlimited bandwidth B (SNR = 0). R has a limit as follows $limR(withSNR \xrightarrow{0}) = 1.433\frac{S}{kT}$. This R_{limit} in unreacheable but a signal-to-noise ratio of -10 dB allows getting 95 % of the maximum possible data transmission. For digital signal the $SNR = \frac{E_b R}{N_0 B}$ and $R \leq B\log_2(1 + E_b\frac{R}{N_0 B})$. So E_b/N_0 has an inferior limit $= ln(2) = -1.59dB$. Below no error free communication can be obtain.

4.4 Encoding

The encoding process can have two definitions:

- Increasing the size of the signal string to protect the data.
- Elaborating the sequence of bits differently, considering the increase in bits given to the input signal.

There are two possible strategies for encoding, aiming to reduce or at least detect the presence of errors: Block Code or Convolutional Codes. Both strategies aim to locate and correct corrupted bits, identifying errors. The graph presented here shows the signal-to-noise ratio on the x-axis. Minimizing the signal-to-noise ratio helps save power consumption, thus, you want to move towards the left part of the graph. On the y-axis, we have the probability of the number of bits that might have an error concerning the length of the received string. While the graph is qualitative, it illustrates that receiving an uncoded signal, with the same signal-to-noise value, results in a reduced probability of Bit Error Rate (BER), which is beneficial.

4.4.1 Block Code

- Blocks of k bits of data are created.
- Parity bits are added to the string, resulting in an n-bit symbol block.
- Block code definition (n,k).

Following block code, error can be detected, but not corrected due to the lack of position information. Instead using rectangular block code, error can be detected, located and subsequently corrected.
Let's review the two encoding techniques we discussed earlier. The first one is Block

Encoding, which involves adding 'parity bits' to the k-bit string, resulting in n-bit symbol blocks. The decision of adding parity bits depends on the number of 1s present in the string. If the number of 1s is odd, a parity bit 1 is added; if it is even (zero or two 1s), a parity bit 0 is added. This method can be applied by considering only the horizontal (string) parity or both horizontal and vertical (column) parity simultaneously. By applying vertical parity, we can not only detect but also locate and correct errors. However, using horizontal parity alone, we can only detect errors due to the lack of position information.

Two other commonly used techniques are the Hamming code block and the Reed-Solomon code, both explained in the slides:

Hamming Code Block

In this method, parity bits are placed at specific positions of the symbol block, which are powers of 2 (1, 2, 4, 8, 16, etc.), while the remaining positions (3, 5, 6, 7, 9, etc.) hold the data bits. The parity bit at position 2^k checks bits in positions that have bit k set in their binary representation. It can detect and correct single−bit errors, but it can only detect double errors. For example, if we have a bit at position 13 (1011), it is checked by bits 1000 (8) and 0001 (1), and the error can be detected and corrected.

Reed-Solomon

(BCH Block Code): This method works by oversampling a polynomial obtained from the data, making it over-determined. As long as "many" of the points are received correctly, the receiver can recover the original polynomial even in the presence of a "few" bad points. It provides a higher level of error correction than the Hamming code block.
In standard telemetry, t = (n-k)/2 series of bit errors can be detected and corrected using Reed-Solomon codes.

4.4.2 Convolutional Encoding

The second strategy is **Convolutional Encoding**, which offers certain advantages in hardware implementation. It is defined by the parameters (n, k, K), where K represents the constraint length. An example is a rate of 1/2 (k/n) with K = 7. Convolutional coding is known to be easier to implement in hardware compared to other coding techniques. For instance, given an input sequence of 101101, it results in 24 output symbols for every 6 inputs, achieving a rate of 1/4. The expected rate of 1/2 is due to the shortness of the input string. For a 10000-bit stream, it would generate a 20007 symbol stream.
To evaluate the performance of different encoding techniques, the graph compares the E_b/N_0 ratio (energy per bit to noise power spectral density ratio) on the x-axis with the Bit Error Rate (BER) on the y-axis. The goal is to stay in the lower left part of the graph. For science missions, a BER ranging from 10^{-5} to 10^{-6} is acceptable, while for telemetry, the value should not exceed 10^{-6}. Additionally, minimizing the Eb/NO is desired, which makes Convolutional coding a favorable choice as it reduces the required energy content relative to the noise for receiving.
The vertical line on the left side labeled "Shannon Limit" represents the ideal goal for design. It denotes an error-free transmission, but practically it may not be attainable due to factors like data rate (R), channel link capacity, and energy usage. The Shannon limit

corresponds to an E_b/N_0 value of -1.6 dB, below which error-free communication cannot be achieved.

4.5 Modulation

Typically, the data stored onboard a spacecraft consists of low−frequency, high−wavelength signals, which are challenging to transmit directly due to the size limitations of onboard oscillators. Therefore, to transmit these signals, modulation is required. Modulation allows for transmitting an analog signal through a communication channel in a way that it can be sent over greater distances and withstand interference without undergoing significant deterioration. In practice, modulation involves manipulating a carrier (a high-frequency waveform) based on the analog signal to be transmitted. Here's how the modulation process works for analog signals:

1. Analog Signal: We start with an analog signal that represents a continuous variation of voltage or amplitude over time. This signal could be a sound wave from a microphone or a video signal from a camera.

2. Carrier: A carrier is a high-frequency waveform used as a "vehicle" to carry the analog signal. The carrier is typically a sinusoidal waveform with a frequency much higher than the frequency of the analog signal. The modulation index is $\beta = \frac{Carrier frequency deviation}{modulation frequency}$. In general β is < 1.57 rad.

3. Modulation: The analog signal modulates the carrier so that variations in the signal are reflected in changes in the carrier's amplitude, phase, or frequency. There are different types of modulation, including (formulas on slides):

   - Amplitude Modulation (AM): In AM, the amplitude of the carrier varies based on the analog signal. The carrier's amplitude is increased or decreased in proportion to the signal.
   - Frequency Modulation (FM): In FM, the frequency of the carrier varies based on the analog signal. The carrier oscillates back and forth in frequency based on signal variations.
   - Phase Modulation (PM): In PM, the phase of the carrier varies based on the analog signal. The carrier's phase shifts back and forth in phase based on signal variations.

4. Transmission: The modulated carrier is then transmitted through the communication channel, such as a cable, radio wave, or optical cable. Modulation allows the analog signal to survive the journey through the channel and resist interference.

5. Demodulation: At the receiver's end, the modulated carrier is demodulated to extract the original analog signal. Depending on the type of modulation used, the demodulation process involves re-converting the carrier's amplitude, frequency, or phase to obtain the analog signal.

4.5.1 Channel bandwith

The channel bandwidth is a key concept in electronic communications, referring to the range of frequencies that a communication channel can carry without experiencing significant distortion or signal loss. Bandwidth is an important measure because it determines the amount of space available to transmit signals and data through the communication channel. Here are some fundamental aspects of channel bandwidth:

1. Frequency Range: Bandwidth represents the range of frequencies between the lowest and highest frequency that the channel is capable of transmitting. For example, a channel with a bandwidth of 10 kHz can transmit signals with frequencies ranging from 0 Hz to 10 kHz.

2. Channel Filter: A channel's ability to carry different frequencies is often determined by its passband filter, which allows only a certain range of frequencies to pass through the channel without being attenuated. Frequencies outside the passband range will experience significant attenuation or be completely blocked.

3. Transmission Capacity: Channel bandwidth influences its capacity to transmit information. A wider channel can carry higher-frequency signals, thereby enabling the transmission of more complex signals or signals with higher data rates.

4. Modulation: In modulation, channel bandwidth limits the type and amount of information that can be transmitted. For instance, frequency modulation (FM) requires a wider bandwidth compared to amplitude modulation (AM) to transmit the same signal.

5. Aliasing: Channel bandwidth is linked to the phenomenon of aliasing, where higher frequencies are mistakenly interpreted as lower frequencies due to limited bandwidth. To avoid aliasing, it's important to sample the signal at an appropriate sampling frequency according to the Nyquist-Shannon sampling theorem.

6. Communication Channels: Bandwidth is relevant in various communication technologies, such as computer networks, radio broadcasts, telephone systems, and optical communications. Different technologies and applications require varying bandwidths based on transmission needs.

Carson's rule

The Carson's formula, also known as the Carson's bandwidth formula, is used to calculate the approximate bandwidth required to transmit a frequency-modulated (FM) signal. The formula was developed by John Renshaw Carson and is often used to estimate the necessary bandwidth for an FM signal. The Carson's formula is expressed as:

$$B = 2(\Delta f + f_{mod} \tag{4.3}$$

Where: B is the approximate bandwidth required (in hertz), Δf is the maximum frequency deviation of the modulating signal (in hertz). It indicates how much the carrier frequency varies in response to the modulating signal variations. f_{mod} is the maximum frequency of the modulating signal (in hertz). It represents the maximum frequency of the modulating signal variations. The Carson's formula takes into account both the variations in the modulating signal (maximum frequency) and the frequency deviation (amplitude of variations) to estimate the approximate bandwidth required to transmit the FM signal. However, it's important to note that this formula is an approximation and may not account for all signal and modulation system characteristics. The Carson's formula is often used as a rough guideline in the design of FM communication systems and in calculating the necessary bandwidth to transmit frequency-modulated signals.

4.5.2 PCM encoding

Pulse Code Modulation (PCM) is an encoding technique used to convert analog signals into digital signals. It is widely employed in telecommunications, digital audio, and other applications where continuous signals need to be represented in digital form. Here's how the PCM encoding process works:

1. Sampling: The incoming analog signal is sampled at regular time intervals. Each sample represents the amplitude of the analog signal at that specific moment. The sampling interval is determined by the sampling frequency and adheres to the Nyquist-Shannon sampling theorem.

2. Quantization: Each sample is then approximated to the nearest value within a discrete set of possible values. This process is called quantization, similar to the concept explained earlier in encoding. Quantization involves dividing the range of possible amplitudes into discrete levels, and each sample is rounded or truncated to the corresponding quantized value.

3. Encoding: The quantized values are then encoded into digital form using a certain number of bits. Each quantized value is represented by a fixed-length binary sequence. The number of bits used to represent each sample determines the encoding precision. For example, an 8-bit PCM encoding can represent 256 discrete levels, while a 16-bit encoding can represent 65536 levels.

4. Transmission or Storage: The binary sequence obtained from PCM encoding can then be transmitted through a digital communication channel or stored in a digital storage device. The binary sequence now represents the original analog signal in digital form.

5. Decoding: On the receiver's end, the process is reversed. The binary sequence is decoded to obtain the original quantized values, and these values are then converted back into an analog signal through the inverse quantization process.

PCM encoding is a widely used technique in applications like digital audio, where sound signals are converted into digital forms for recording, transmission, and playback. This technique enables accurate representation of analog signals in digital form, allowing for efficient and faithful transmission and storage.

Typical PCM encoding

- NRZ-M (Non-Return-to-Zero Mark): In NRZ-M encoding, a "0" is represented by a constant level (often a negative value) of the signal, while a "1" is represented by a transition to the opposite level (often positive) of the signal. This means that each "1" causes a transition in the signal, while "0"s are represented by a constant level.

- UNI-RZ (Unipolar Return-to-Zero): In UNI-RZ encoding, a "0" is represented by a signal at a constant level (often positive), while a "1" is represented by a transition from the constant level to a zero voltage value. After the transition, the signal returns to the constant level.

Both encoding schemes, NRZ-M and UNI-RZ, are used for transmitting digital data through communication channels such as transmission lines or optical channels. The choice between them depends on the specific needs of the communication system, including transmission speed requirements, spectral efficiency, and robustness against noise and interference.

Alternatives to PCM encoding

Amplitude Shift Keying), FSK (Frequency Shift Keying), and PSK (Phase Shift Keying) are three modulation techniques used to transmit digital signals through communication channels. Each of these techniques exploits a different characteristic of the carrier signal (usually a sinusoidal waveform) to represent information bits. Let's see how they work:

1. ASK (Amplitude Shift Keying): In ASK, information is transmitted by varying the amplitude of the carrier signal based on the data to be transmitted. A "0" might be represented by a low amplitude (e.g., a low voltage signal), while a "1" could be represented by a high amplitude (e.g., a higher voltage signal).

2. FSK (Frequency Shift Keying): In FSK, information is transmitted by varying the frequency of the carrier signal. A "0" might be represented by one carrier frequency, while a "1" could be represented by another carrier frequency. PSK (Phase Shift Keying): In PSK, information is transmitted by varying the phase of the carrier signal. A "0" might be represented by a specific phase (e.g., 0 degrees), while a "1" could be represented by another phase (e.g., 180 degrees).

ASK, FSK, and PSK techniques allow for the transmission of digital information using different parameters of the carrier signal. The choice between these techniques depends on factors such as available bandwidth, robustness against noise and interference, and the complexity of the modulation and demodulation system. Each technique has its own advantages and disadvantages and is used in various applications, such as wireless networks, satellite communications, radio broadcasts, and more.

4.6 Encoding/modulation choice influence

4.6.1 Different choice

BPSK, QPSK, FSK, 8FSK, and MSK are all modulation schemes used to transmit digital signals through communication channels. Each of them leverages different characteristics of the carrier signal to represent digital data. Here's an explanation of each:

1. BPSK (Binary Phase Shift Keying): In BPSK modulation, information is transmitted by varying the phase of the carrier signal. A "0" might be represented by a phase (e.g., 0 degrees), while a "1" might be represented by another phase (e.g., 180 degrees).

2. QPSK (Quadrature Phase Shift Keying): QPSK is a variation of BPSK that allows the transmission of two bits at a time. In QPSK, data is split into pairs of bits, and each pair represents a point on the constellation diagram. Each point has a specific phase combination to represent four possible bit combinations.

3. FSK (Frequency Shift Keying): In FSK modulation, information is transmitted by varying the frequency of the carrier signal. A "0" might be represented by one carrier frequency, while a "1" might be represented by another carrier frequency.

4. 8FSK (8-Level Frequency Shift Keying): 8FSK is a variation of FSK that allows the transmission of eight frequency levels. Each level represents a combination of three bits.

5. MSK (Minimum Shift Keying): In MSK modulation, the carrier phase shifts by 180 degrees at half the bit period, enabling better separation between closely spaced signals in the frequency domain. MSK is often used in applications where robustness against noise is critical, as its minimum frequency shift structure makes it less susceptible to intersymbol interference (ISI) and enhances its noise resilience.

Each modulation scheme has its advantages and disadvantages and is used in various applications based on the specific needs of the communication system.

4.7 Amplification

Moving forward in our system block schemes, we now reach the amplification stage of the signal. It's important to note that this amplification process occurs both in the transmitter and the receiver. The amplifier is responsible for generating the required radio power for transmission. There are two classes of amplifiers commonly used: Solid state amplifiers and TWTA (Travelling Wave Tube Amplifier) amplifiers. TWTA amplifiers are heavier but provide higher output power, making them suitable for applications requiring higher power levels. On the other hand, solid state amplifiers are preferred for power outputs ranging from 5 to 10 W. In the graph, we can observe the characteristics of these two amplifier types in terms of power (above) and mass (below). During the design process, it's essential to find a trade-off between the two options. While TWTA amplifiers offer better output power for a given input, they come at the expense of increased mass, which can be a critical constraint for spacecraft. Thus, designers need to carefully consider the power requirements and weight constraints to make an informed decision.

4.8 Antenna design

The final block in the design process is the terminal antenna. There are various configurations of antennas to choose from, and it's essential to select the most suitable one based on specific requirements. The basic type is the Isotropic Antenna, where the oscillator emits signals uniformly in all directions. However, this configuration is not very powerful since the energy is distributed over a larger surface as the distance increases.

$$\Psi_{radiated} = P_t/(4\pi r^2) \tag{4.4}$$

To achieve more focused and powerful transmission, directional antennas are preferred. One such example is the Directional Antenna, which concentrates the signal in specific

directions.
The gain of a directional antenna is the ratio between the maximum flux and the overall isotropic radiated flux. The gain is influenced by the reflector's geometry. The beamwidth angle (θ_{eff}) represents the aperture within which the power is concentrated and determines the field of view. After sizing and choosing the beamwidth angle, it's crucial to consider the masking angle to account for practical limitations in real-world scenarios. The masking angle (ϵ) helps in dealing with the ideal beamwidth angle and ensures efficient antenna performance.

$$G_{ant} = \Psi_{max}/\Psi_{rad,iso} = (4\pi A_{eff})/\lambda^2 \tag{4.5}$$

$$A_{eff} = \eta(\pi D^2)/4 \tag{4.6}$$

$$\theta[deg] = (70\lambda)/D \tag{4.7}$$

$$\epsilon = \pi/2 - \phi - \alpha \tag{4.8}$$

The masking angle accounts for various imperfections on the celestial sphere, such as morphology (e.g., mountains, valleys) and atmospheric interferences. To ensure efficient antenna performance, it is common to reduce the field of view by a reasonable value, typically ranging from 5° to 10°, regardless of the antenna type being used. The chart below provides different rules and equations for sizing antennas. Although directional antennas offer superior performance, omnidirectional antennas remain essential in certain situations. Isotropic antennas, for example, are necessary during the initial phases of a mission or in safe mode when power needs to be conserved, and communication with Earth is essential regardless of the antenna's pointing direction. The most commonly used configurations include Horn Antennas, Helical Antennas, Reflectors, and Patch Antennas (which are less intrusive in configurations but still deliver good performance and are therefore often used in small satellites). As mentioned earlier, directional antennas require well-sized pointing mechanisms, which in turn demand power. This power demand can be a critical aspect, leading to the consideration of alternative solutions, such as electronically steered antennas using phased arrays. Phased arrays consist of omnidirectional antennas electronically combined to form a narrow beam, achieving significant gain (up to 30 dB with 1000 elements). While highly precise, phased arrays should only be considered if necessary, as the benefits gained in terms of mechanical parts and power are offset by increased electronics, mass, and thermal control requirements.

4.9 Losses

During the system sizing process, it is crucial to consider the potential signal losses that may occur from signal generation to signal reception. These losses can be attributed to various factors and can be categorized as follows:

- Transmission Losses: (free space, misalignment, atmospheric)
- Hardware losses
- Noise (environment noise, antenna noise, passive or active device noise)

4.9.1 Transmission Losses

Free space losses

These losses occur due to the spreading of the signal in space and are strongly dependent on the frequency and distance. The formula below illustrates this relationship:

$$L_{space} = (\frac{4\pi r}{\lambda})^2 = (\frac{4\pi fr}{c})^2 \tag{4.9}$$

Misalignment losses

Result from offset pointing between the transmitter (tx) and receiver (rx). Achieving perfect symmetry in axis pointing is rare, and larger beams can be used as a solution. Although larger beams reduce power, they increase the certainty of receiving the signal.

$$L_{mis} = -12(\frac{e}{\theta})^2[dB] \tag{4.10}$$

Atmospheric Losses

These losses arise from the interaction of the signal with gases in the atmosphere and are present in environments with an atmosphere, not limited to Earth. Several factors affect atmospheric losses:

- Elevation angle: Lower elevation angles lead to more signal attenuation. Balancing the angle at around 45° is optimal but may impact the visibility window and require more power for quicker data dumping.
- Choice of frequency: Certain frequencies, like 60 GHz, are prone to severe losses in environments with H2O, making it advisable to avoid them.
- Ground station location: Dry and high-altitude locations are preferred to minimize wet (H2O) conditions and mitigate atmospheric losses. In some cases, placing the terminal in space on a GEO and outsourcing the data transmission to Earth could be a more straightforward solution, avoiding the need to tackle this issue during the design process.

4.9.2 Noise

The final element to consider is noise, which primarily arises from thermal sources and can be represented by the equation:

$$P_{noise} = KT_eB \tag{4.11}$$

where K is the Boltzman constant, T_e is the equivalent noise temperature and B is the equivalent noise bandwidth.

$$K = 1.38e^{-23}[WsK^{-1}] \tag{4.12}$$

$$B = 1.12B_{-3dB}[Hz] \tag{4.13}$$

The equivalent noise temperature should be considered for different sources: purely thermal sources, passive devices (like transmission lines), and active devices (receivers). For thermal sources, the equivalent temperature is the thermodynamic temperature $T_e = T_t$.

For passive devices, figures of merit are provided in the datasheets of components, and the equivalent temperature is calculated as follows: $T_{ei} = T_t(L-1)$, $T_{eo} = T_t(L-1)/L$. For active devices, like receivers, the equivalent temperature is given by $T_e = T_o(F-1)$. F is the noise figure.
To include these measurements in the noise power, you create a chain that accounts for the antenna temperature, cable losses, temperature noises, and active devices. The noise characteristics depend on the frequency and the direction the antenna is pointing. For example, when looking at the graph below, curve D represents a sun-looking antenna, while curves E and F are for Earth-looking devices. The lower the curve, the quieter the environment being observed. Ground stations are often located in remote areas to minimize environmental noise and atmospheric disturbances. Regarding noise components, the figure of noise required in the formulations is the real versus ideal output power of the component. Although it may not always be readily available, it is usually suggested to adopt values in the order of 0.1 or 0.2 when necessary. Another crucial aspect to consider is that ground stations are cooled down to approximately 70 K, ensuring that the receiver remains noise-free. Consequently, the noise to be taken into account primarily comes from the onboard devices.

4.10 Link budget

The link budget is a crucial calculation that correlates the energy per bit to the noise per bit, taking into account various factors. These factors include the transmitted power (amplifier output), amplifier characteristics, line losses in the on-board system, antenna gain (related to Attitude Determination and Control System), space and atmospheric losses, and receiving antenna gain. The link budget equation is expressed as follows:

$$E_b/N_o = \frac{P_{tx}G_{tx}G_{rx}L_1L_sL_a}{KT_sR} \tag{4.14}$$

Where: Eb / No is the energy per bit to noise per bit ratio, Pe is the emitted power per area, G is the gain of the transmitting/receiving antenna, K is the Boltzmann constant, T is the noise temperature, R is the data rate, L represents the overall losses in the transmission. By using this link budget equation, you can evaluate the accuracy required for your system (error versus coding/modulation). This will help you fine-tune the system parameters or select the appropriate modulation scheme. Keep in mind that you should have a margin of at least 3dB above the calculated value to ensure a robust and reliable system. If the margin is significantly higher, it may indicate that your system is too heavy or over-engineered, and you may need to reevaluate your mission profile.
Remember to consider all the requirements for each subsystem, the antenna location, and the connection with the ground station during the link budget computation. For example, let's consider the steps for computing the link budget focusing on science data for a mapping mission:

- Start by selecting the appropriate frequency band (e.g., X-band) based on statistical power data and identify the maximum acceptable error. Determine the worst-case scenario regarding orbit position and visibility of the ground station. This will give you an estimation of the downlink data rate, considering time window duration and data volume.

- Choose the modulation scheme based on the previous chart (refer to the slide on PCM coding and modulation). Take into account any constraints related to the antenna beam and other factors. Remember that a constraint on one antenna or diameter will impact the entire vehicle, including the launcher fairing dimensions.
- Involves implementing all the losses and constructing the link budget elements to calculate the receiver power and the energy per bit in relation to the noise. To ensure accuracy, cross-check the results and verify if there is a margin of 3dB. Additionally, you have the option to split the power into the main carrier frequency and the side lobes by fixing the modulation index. This means allocating more power to the central carrier signal and distributing the rest across the side lobes. Once the modulation index is determined, calculate the percentage of total power in the carrier and the side lobes accordingly. The following steps outline the process. For each phase that includes a Telemetry, Tracking, and Command (TMTC) mode:
  - Design the link: Select the Tx-Rx frequency, determine the transmitter power, estimate internal losses, determine the beamwidth, and calculate antenna gain and diameter.
  - Size the communication subsystem: Choose the antenna configuration, compute antenna size, estimate antenna mass, and determine transmitter mass and power requirements.
  - Define output: Establish the mass budget and power budget for each phase, specify requirements and constraints for other subsystems (ADCS, outputs, OBDH, etc.), determine the overall configuration, plan for operation, and select the appropriate ground segment and Control/Science centers.

4.11 Design steps

Now that all the individual building blocks have been defined, it's time to integrate them to create the complete system. The design process involves determining the type of activities, data collection, timing of data acquisition, data size, and understanding the orbital dynamics of the mission. This information helps in selecting the best configuration for the on-ground network. You have the flexibility to decide when to transmit the data to the ground station. Except for telemetry, which should be regularly sent, you can choose when to make your data available. Significant events like gravity assists or detachment of elements must be communicated with the ground station, as they require monitoring, even if the spacecraft is autonomous. When planning data transmission, it's essential to coordinate with the On-Board Data Handling (OBDH) engineer to ensure that the acquired data can be efficiently and reliably stored. It involves a trade-off between data transmission requirements and the available resources, as you are not bound by strict obligations other than user constraints. The uplink and download considerations depend on whether your mission involves artificial control or ballistic maneuvers. Data rate collection is related to the volume of data your system generates, primarily determined by scientific acquisition and utilization needs. Understanding the average data volume helps estimate the time required for data transmission. Compliance with international regulations is crucial while sizing the system, especially when selecting the frequency. Certain frequencies may be prohibited for specific missions. For example, X-band, suitable for large data volumes in deep space missions, may not be allowed for use on Earth, limiting you to S-band. Such restrictions can significantly impact data download times. While you may not be responsible for detailed system sizing, understanding the logic and sequence of the process is important. For the ground segment, you are required to select the appropriate ground station, taking into account the location and frequency compatibility with the spacecraft's transmission. This information is then used to compute the link budget and design all the necessary onboard equipment.

4.11.1 Focus on this part

A few points should be highlighted regarding the data rate in the telecom subsystem:

- Data rate for housekeeping is typically in the range of kbps, which is sufficient for regular monitoring and control. However, for attitude data, which is sampled at a very high frequency (1 to 4 times per second), a dedicated channel may be required, especially during critical mission phases involving payload pointing, station alignment, or specific maneuvers. For worst-case scenarios, the data rate for attitude data might jump to megabits per second. Payload data requires a dedicated channel with a higher data rate, typically in the range of Mbps or Gbps, depending on the mission's requirements.

- It's important to distinguish between the acquisition data rate and the downlink data rate. The acquisition data rate refers to the rate at which data is acquired or generated on board, such as capturing images during mapping. The downlink data rate is the rate at which the acquired data is transmitted to the ground station. These rates may not be the same and data can be split into packages as needed.

- The frequency bands used for communication must consider factors like atmospheric attenuation. For example, Q, V, and W bands have challenges due to atmospheric effects, but in-space communications may not face these issues. Frequency allocation and regulations depend on the location of the ground station and are typically overseen by the ITU (International Telecommunication Union).

- The data architecture may vary not only for the space segment but also for the ground segment. Data from the space segment can be categorized into housekeeping, attitude, and payload data, each with specific requirements. Selecting the ground network involves specifying which kind of data is downloaded at each specific ground station. The data rate is related to the transmission frequency, and different frequency bands can be used for different purposes, like S-Band for telemetry and X-Band for science data. Ground stations need to be correlated with the Mission Operations Center (MOC) and the Science Operation Center (SOC) to ensure smooth coordination and data management.

4.11.2 Ground segment architecture

The figure illustrates the connections between the three main actors of the ground segment (GSN, MOC, SOC), the space segment, and the final user. One crucial point to note is that personnel is always present on the ground segment side to perform infrastructure maintenance and carry out procedures commanded by operators on the ground. Some procedures are automated and executed by the MOC's software or the spacecraft's onboard software, such as pointing the antennae towards the station.

The Ground Station Network (GSN)

The ground station network represented by the light blue square, consists of radio frequencies that interface with the spacecraft. It comprises antennas that directly communicate with the onboard antenna. The GSN interfaces with the MOC, providing tracking for both uplink and downlink, telemetry (downlink) consisting of housekeeping and payload data, and telecommand (uplink) received from the MOC.

- Tracking: Involves strategies employing radio frequencies to gather information about the spacecraft for orbit determination and navigation. This data is essential for the flight dynamics and orbit determination routines performed at the MOC.
- Telemetry: Comprises both housekeeping telemetry (HKTM) and payload data (PL).
  - HKTM: Contains engineering data regarding the spacecraft's health status, including temperatures on certain surfaces, voltage and current measurements at various points, and readings from sensors like angular velocities from the IMU. For technological demonstrations, this data can include payload information, such as GPS tracking data.
  - PL: Contains scientific data collected from the payload.
- Telecommand: Involves sending commands to instruct the space segment to execute specific procedures, such as decommissioning the payload or turning components on/off.

The selection of a specific ground station network over others can be guided by three main categories:

- The number of antennas available, which can lead to the exploitation of different frequencies onboard and provide redundancies in case of ground antenna failures.
- Number of Stations: This refers to the different sites where the ground stations are strategically positioned. By increasing the number of stations, the mission can enhance its coverage for both uplink and downlink data. A larger number of stations allows for better connectivity and improved communication with the spacecraft throughout its orbit.
- Uniformity of Antennae: This criterion emphasizes the use of a single provider operating the entire network. Opting for uniformity enables the exploitation of central control mechanisms, such as the core network of ESA with its centralized control center in Darmstadt. Having a single provider and a centralized control center offers several advantages, including streamlined operations, standardized procedures, and efficient coordination among the ground stations.

−**ESA Ground Station Network**: In Figure, the ESA Ground Station Network (GSN) is depicted, comprising three distinct categories of ground stations:

- The light blue dots represent the core stations directly owned and operated by ESA, under the control of the Network Operation Centre in Darmstadt.
- The orange dots signify the augmented stations, which are owned and operated by industry partners.
- The green dots denote stations belonging to other space agencies, such as NASA or ASI, which collaborate with ESA on various missions. (Note that Malindi is the ASI ground station).

The distribution of these stations is geographically diverse to maximize coverage. Polar stations are strategically placed for sun-synchronous and polar orbits. Cebreros, Malargue, and New Norcia stations, forming ESA's deep space network, are equipped with 35m dish diameter antennae, positioned at 120° intervals to ensure continuous coverage for deep space missions. Additionally, Kourou station plays a crucial role in supporting launch operations.
−**NASA Ground Station Network**: NASA's network is divided into two main segments:

- Near Earth Network: Utilized for Low Earth Orbit (LEO) operations up to Geostationary Orbit (GEO).
- Deep Space Network (DSN): Engaged in missions beyond GEO. NASA's DSN adopts a similar philosophy to ESA's, with stations positioned 120° apart. Key locations for DSN are Goldstone (USA), Madrid (Spain), and Canberra (Australia). The DSN ground stations are equipped with four 34m dish antennas and one powerful 70m dish antenna, making it the most robust network in existence.

Commercial services

In recent years, the emergence of Ground Segment as a Service (GSaaS) companies has provided increased support for space missions. GSaaS offers mission designers greater flexibility by eliminating the need to build and operate their own ground stations. These services encompass Ground Station Network (GSN) support and Mission Operation Centre (MOC) services like decoding, passage scheduling, data storage, and protection. Prominent companies in this field include LeafSpace, Amazon Web Services (AWS), Atlasor, and KSat. Inter-satellite connectivity is another commercial service gaining popularity. Companies owning satellites in Geostationary Orbit (GEO) or part of Low Earth Orbit (LEO) constellations provide data link services for third-party satellites. These companies receive data from third-party satellites and transmit it to ground stations for further processing and distribution to end-users. Such services benefit satellites facing constraints in direct ground station interfacing due to factors like latency requirements or extended coverage needs. By utilizing these commercial services, satellite operators avoid the challenges of setting up their own ground station networks. Both private companies and government entities offer inter-satellite connectivity services. Notably, ESA collaborates with Airbus to provide the DRS service in GEO, while NASA's TDS operates a fully functional GEO constellation. Established commercial services like Globalstar, Inmarsat, and Iridium, originally designed for Earth communication, are now supporting space missions, especially cubesats, to reduce costs further.

Chapter 5

Mission Analysis

5.1 Deep space maneuvering

Algorithm for Constructing a Multiple Gravity Assist (MGA) Trajectory:

- Divide the trajectory into multiple phases, each connecting two celestial bodies.
- Compute the outgoing velocity at each celestial body using the linked conic model.
- Determine the length of each subphase, the departure time, and the initial departure velocity to minimize the total required ΔV (velocity change).
- Minimize the cost function, f [v, T, ...], by optimizing the trajectory parameters, including velocities and times, for each phase.
- The sequence of celestial bodies, times of flight (tof), distances (r), and gravity assists (GA) for each phase should be represented as N subphases: [tof, r, GA, ..., tof, r, GA, ..., r, GA, tof].
- Apply appropriate transformations (transf) to refine the trajectory optimization.

By following this algorithm, a multiple gravity assist trajectory can be efficiently constructed, reducing the total propulsion requirements while navigating between celestial bodies to achieve the mission's objectives.
The existing model can be enhanced by incorporating deep space maneuvers (DSMs). The introduction of a DSM involves adding a Lambert arc to an unknown position (ΔV_0) in deep space. The trajectory with DSMs follows the pattern as shown below:

- Split the trajectory into multiple phases, each connecting two celestial bodies.
- Each phase is further divided into two sub-phases, separated by a deep-space maneuver (DSM).
- Compute the outgoing velocity at each celestial body using the linked conic model.
- Determine the length of each subphase, the departure time, and the initial departure velocity to minimize the total required ΔV (velocity change).
- The set of optimization variables should include three variables for the DSM ri(t) position and one time variable for the DSM occurrence. The set of cost functions should include one ΔV for each DSM.

By introducing DSMs, the trajectory model becomes more versatile and effective in optimizing deep space missions, allowing for efficient navigation between celestial bodies and reducing propulsion requirements.

5.1.1 Degrees of uncertainty

The model that is linked establishes the state vector [rs/c−sun, $v_{i,sun}$] of the spacecraft at the planet as a point along its trajectory. From the planetocentric reference frame, only a portion of the spacecraft's state vector is defined; specifically, only the $v_{i,pl}$ vector is fully characterized. The v_{inf} vector lacks a precise point of application P within the Sphere of Influence (SOI). As a result, a family of hyperbolas is generated for each identified v_{inf}. We could have potential inbound hyperbolic trajectories are determined based on the given $v_{i,pl}$, with the impact parameter $|\Delta|$ established, though its unit vector is not defined, or the same but the vector of the impact parameter is define but not its modulus. To uniquely determine the characteristics of the flybys, an additional three scalar quantities must be specified: $|\Delta|$, v_{inf}.

5.2 B plane

The plane is positioned at the focus of the hyperbola and is orthogonal to the incoming asymptote. Characteristics of the B-Plane include:

- The B-plane is positioned perpendicular to the plane of the trajectory.
- The B-plane establishes a flat coordinate system that is valuable for precision targeting during fly-bys.
- The B vector resides at the intersection of the B-plane and the trajectory plane, which governs the design of the hyperbolic plane.
- The B-plane's local coordinate system is defined by two vectors, T and R.

The incoming asymptote unit vector $\hat{S} = \frac{\vec{v_{inf}}}{|\vec{v_{inf}}|}$, and in the perifocal reference frame $\hat{S} = cos(_{inf})\hat{e} + sin(_{inf})\hat{p}$.
The **T vector** is referred to the planet's north pole $(\vec{N})$: $\vec{T} = \frac{\hat{s} \vee \vec{N}}{|\hat{s} \vee \vec{N}|}$. The **R vector** is define as the cross product between $\vec{S}$ and $\vec{T}$. So the **B vector** on the B−plane is define as $\vec{B} = \vec{\Delta} \vee \vec{T} + \vec{\Delta} \vee \vec{R}$. When a point on the B-plane is given by Δ and θ, the post-gravity assist conditions become fully determined, allowing for the representation of changes in the incoming vector onto the B-plane.

5.2.1 TCM

TCMs are computed in flight to target the incoming asymptote for the GA on its B plane and reduce the dispersion ellipse size at the planetary SOI entrance.

5.2.2 Titan example

Aim points in the B-plane are defined by: • the magnitude of the impact parameter vector B • its associated azimuthal angle with respect to the T-axis . Equivalently, the atmospheric entry angle at a given altitude could be used instead of the impact parameter B. This alternative was adopted for Huygens mission planning purposes. Huygens delivery accuracy (3) in the B-plane is given by the ellipse encircling the tip of the target vector (=-60°, = 64°). Contours of constant are drawn as concentric dotted circles. Contours drawn in solid vertical lines report the constant cosine of the angle between the E-W direction and the line-of-sight probe-orbiter (ZWP regions).

Chapter 6

ADCS Subsystem

AOCS, which stands for Attitude and Orbit Control Subsystem, refers to the functional chain of a satellite responsible for managing its attitude and orbit when the orbit guidance is not carried out on board. This subsystem comprises several essential components:

- Attitude and orbit sensors: These sensors help gather information about the satellite's current attitude and position in space.
- Attitude estimation and guidance: This part involves estimating the satellite's attitude (orientation) and providing guidance to maintain the desired orientation.
- Attitude and orbit control algorithms: These algorithms process data from sensors and compute the necessary adjustments to control the satellite's attitude and orbit.
- Attitude and orbit control actuators: Actuators are mechanisms responsible for physically changing the satellite's attitude and adjusting its orbit as needed.

In addition to these components, there might be an orbit estimation function, often referred to as Navigation, which aids in determining the satellite's position accurately. On the other hand, GNC, which stands for Guidance, Navigation, and Control, is typically employed for the on-board segment when the satellite's position is controlled in a closed-loop manner. The functions of GNC include:

- Computation of targeted orbit and attitude: GNC calculates the desired orbit and attitude for the satellite based on mission requirements.
- Attitude and orbit determination: This process involves determining the current attitude and position of the satellite in real-time.
- Attitude and orbit control: GNC is responsible for actively controlling the satellite's attitude and orbit, making adjustments as needed to maintain the desired trajectory.

The Attitude Determination and Control Subsystem (ADCS) is a subset of the broader AOCS system, focusing specifically on the determination and control of the satellite's attitude.

6.1 Requiremets

6.1.1 Functional requirements

The AOCS must possess both hardware and software capabilities to fulfill the following tasks essential for the mission:

- Attitude Measurement, Estimation, Guidance, and Control: The AOCS should be able to measure, estimate, guide, and control the spacecraft's attitude as required for the mission.
- Orbit Control Maneuvers: The AOCS should be capable of performing orbit control maneuvers based on the specific mission requirements.
- Safety Assurance: The AOCS must ensure the spacecraft's safety at all times, including emergency and anomaly situations, in accordance with the specified failure management requirements.
- Mission Availability: The AOCS must contribute to the overall mission availability as defined at the satellite level.

Acquisition & **Keeping Requirements**: Throughout all phases of the mission, the AOCS must possess the capability to acquire and maintain all the necessary attitudes required to execute the mission successfully.
Attitude Determination Requirements: The AOCS must provide the necessary hardware and software tools to autonomously determine the spacecraft's attitude onboard, adhering to the mission requirements.
Orbit Determination: If the mission necessitates a navigation function, the AOCS must have the hardware and software means to autonomously determine the spacecraft's orbital state, which includes position, velocity, and time.
Mission Pointing Requirements: During the mission operational phase, the AOCS must ensure that the attitude guidance and pointing specified by the mission requirements are consistently achieved.
Orbital Acquisition and Maintenance Requirements: The AOCS must possess the necessary capability to execute orbit control maneuvers as determined by the mission analysis.
Safe Mode: In the event of a major anomaly, the AOCS must autonomously achieve and maintain a safe pointing attitude and angular rates to safeguard the spacecraft's vital functions, including power, thermal, and communications integrity.

- The activation of the safe mode can be initiated by ground telecommand (TC).
- The deactivation or return from safe mode can be initiated by ground telecommand (TC).

6.2 Performance requirements

Assuming APE (Absolute Performance Error) is less than 1 degree with a 90% probability, the interpretation of this requirement can be approached in two different ways: ensemble interpretation and temporal interpretation. Additionally, a mixed interpretation can be considered for cases falling between the extremes. If 90% of the time for all members of the ensemble it's temporal interpretation. If 100% of the time for 90% of the ensemble it's an ensemble interpretation. For cases falling between the extremes, a mixed interpretation can be considered based on the specific mission needs and statistical analysis. The requirement can be statistically interpreted as follows:

- In the general case or applicable to a Gaussian distribution, the APE is within 3σ (standard deviations) with a probability of 99.7%.
- In the general case or applicable to a Gaussian distribution, the APE is within 2σ (standard deviations) with a probability of 95.5%.
- In the general case or applicable to a Gaussian distribution, the APE is within 1σ (standard deviation) with a probability of 68.3%.
-

It's essential to consider the statistical interpretation of the requirement based on the specific needs of the mission and the type of distribution that applies to the APE data.
Absolute Pointing Requirements - Accuracy in Control: During the operational mission phase, the AOCS must ensure an absolute pointing performance with an accuracy of TBS microradians, achieved at TBS % confidence level. The statistical interpretation used for this requirement can be either temporal, ensemble, or a combination of both (mixed).
Attitude Absolute Knowledge Requirements - Accuracy in Determination: Throughout the operational phase of the mission, the AOCS is responsible for maintaining on-board absolute attitude knowledge with a performance of TBS microradians, attained at TBS % confidence level. The statistical interpretation applied to this requirement can be temporal, ensemble, or mixed.
Relative Pointing Requirements: During the operational phase of the mission, the AOCS must deliver on-board relative pointing performance of TBS microradians over a duration of TBS seconds, achieved at TBS % confidence level. The statistical interpretation considered for this requirement is TBS.
Orbit Knowledge Requirements - Accuracy in Determination: The navigation function should provide on-board orbit estimation with the following accuracy: TBS metres (for position), TBS metres per second (for velocity), and TBS seconds (for time). This estimation must be in a TBS unambiguous space and time reference frame, attained at TBS % confidence level. The statistical interpretation used can be temporal, ensemble, or mixed.
Orbit Control Requirements - Accuracy in Control: For orbit control, the AOCS must execute Delta-V maneuvers commanded by the ground with superior accuracy:

- TBS % of the Delta-V magnitude along the commanded direction.
- TBS % of the Delta-V magnitude on the perpendicular directions.

AOCS Performance Requirements: TBS values are to be specified based on the mission's unique needs and constraints.
Agility Requirements: The AOCS is required to have the capability to execute attitude maneuvers under the following conditions:

- Attitude changes of TBS degrees on the roll axis must be completed in less than TBS seconds, which includes the tranquilization phase.

- Attitude changes of TBS degrees on the pitch axis must be completed in less than TBS seconds, including the tranquilization phase.

- Attitude changes of TBS degrees on the yaw axis must be completed in less than TBS seconds, including the tranquilization phase.

AOCS Performance Requirements & **Budgets:** In addition to the specified agility requirements, other performance aspects that might not be immediately apparent can be quantified and demanded:

- Stability: This measures the ability of the system to withstand bounded external disturbances and remain within a bounded domain around an equilibrium position or trajectory. Stability margins, such as gain margin and phase margin, can be used to quantify the maximum parameter excursions that maintain stability properties.

- Transient Response: This evaluates the system's ability to reach a steady-state with specific maximum parameters, such as settling time and overshoot.

- Robustness: This assesses the capability of the controlled system to maintain performance or stability characteristics despite uncertainties in the plant, sensors, actuators, and environmental factors.

- Jitter: This specifies the bound on high-frequency angular motion, which can be defined in terms of RPE (Rate Pointing Error) or RKE (Rate Knowledge Error).

- Drift: This sets a limit on slow, low-frequency angular motion. Drift can be defined in terms of PDE (Position Drift Error) or KDE (Knowledge Drift Error).

The AOCS performance requirements and budgets should encompass these various aspects to ensure the system's effectiveness and reliability.
The AOCS must provide detailed allocations, known as budgets, for several key performance parameters:

- Absolute Performance (Attitude Pointing) Budgets (APE): These budgets define the acceptable levels of deviation in absolute attitude pointing during the mission, ensuring that the spacecraft maintains the required accuracy in its orientation.

- On-board Absolute Attitude Knowledge Budgets (AKE): These budgets outline the permissible tolerances for the on-board absolute attitude knowledge, ensuring that the spacecraft's knowledge of its attitude remains within specified limits.

- Relative Performance (Attitude Pointing) Budgets (RPE): These budgets specify the allowable deviations in relative attitude pointing during the mission, guaranteeing the desired level of precision in relation to reference points.

- Duration Budgets: Duration budgets encompass various aspects, including the time allocated for mode transitions, agility maneuvers, convergence to desired states, AOCS availability periods, and scheduled outages.

- Contribution to Propulsion Related Budgets: This category outlines how the AOCS affects propulsion-related aspects, ensuring that it operates within predefined limits to maintain propulsion efficiency and reliability.
- Orbit Correction Performance Budgets: These budgets define the acceptable performance levels for orbit correction maneuvers, ensuring that the spacecraft can achieve and maintain its intended orbital trajectory with the required accuracy.

In summary, the AOCS performance budgets play a crucial role in defining and monitoring key performance parameters to ensure the mission's success and efficient spacecraft operations.

6.3 Design Process

1. Definition of Control Modes: Control modes refer to different modes or configurations of the Attitude and Orbit Control Subsystem (AOCS) that are used to achieve specific mission pointing requirements. These modes are determined based on the payload (p/l) needs, mission phases and operations, as well as the capabilities of the onboard sensors and systems.
2. Summary of Mission Pointing Direction Requirements by Control Mode: The mission pointing direction requirements are tailored based on several factors:
    - Payload (p/l) type, such as Nadir pointing, scanning, or inertial target tracking.
    - Specific mission phases and operations.
    - The capabilities of the onboard sensors and systems that affect the pointing accuracy.
3. Summary of Payload and Mission Pointing Knowledge/Control Accuracy Requirements: The pointing accuracy requirements for different payloads and mission tasks vary based on the vehicle's tasks and phases. Here are typical pointing accuracy values for different payload types:
    - Synthetic Aperture Radar (SA): 4° to 10°
    - High-Gain Antenna (HGA): 0.1° to 0.5°
    - Optics and Cameras: 0.001° to 0.1°
4. Definition of Required Maneuvers for the Mission: The required maneuvers for the mission include specific details such as angles, duration, rates, and frequency of the maneuvers. These maneuvers are designed to achieve the desired spacecraft orientations, repositioning, or trajectory adjustments necessary for the successful execution of the mission.
5. Quantification of Disturbance Torques: The internal and external disturbance torques, including perturbations, sloshing, and effects of flexible appendages, need to be quantified and assessed.
6. Selection of ADCS Architecture: Choosing the appropriate Attitude Determination and Control System (ADCS) architecture requires considering various factors, such as:

- Payload pointing, thermal, and power requirements.
- Thrust vector control needs.
- Control authority available for maneuvering.
- Admissible attitude errors during translational DeltaV maneuvers.
- Requirements or constraints on maneuver slew rates (significant if exceeding 0.5°/s).
- Approximate solar panel area, which might influence the preference for 3-axis stabilization if wings exceed spacecraft area.

7. Selection and Sizing of Major Hardware Components: Key hardware components, including sensors and actuators, must be carefully selected and sized based on the mission's specific needs.

8. Definition of Attitude Determination and Control Algorithms: Precise algorithms for attitude determination and control must be defined to achieve the desired performance and stability of the spacecraft.

9. Trade-Off Studies to Improve Requirements and Baseline Selections: Conducting trade-off studies is essential to optimize and enhance the requirements and baseline selections, making informed decisions on design and performance.

10. Incremental Testing of Hardware and Software: Testing the ADCS hardware and software at increasing levels of complexity and fidelity, such as Virtual Model (VM), Engineering Model (EM), and Flight Model (FM), allows for validation and verification of the system's functionality and performance.

As soon as you receive directions or requirements for pointing and precision, the selection of sensors and actuators is influenced by how accurately you need to know or maintain the directions. For instance, if you are using a high gain antenna with a specified beamwidth (the antenna's field of view where it concentrates its power most effectively), you must ensure that the antenna remains aligned with at least an occurrence that is higher or half of the beamwidth. Otherwise, your communication will be directed in the wrong direction. On the other hand, if you are using a telescope searching for random cosmic events, like star mergers, the precision in reconstructing the pointing direction when these events occur becomes crucial. In such cases, the accuracy of knowledge (i.e., addressing the sensor's capabilities, such as bolometers and gyros) is essential. When control and knowledge accuracy are well-coupled, you have a clear understanding of your position and can apply the necessary control effectively. Tight control requires tighter accuracy in knowledge.
In this chain of accuracies, the sensors you select must have accuracy levels approximately an order of magnitude higher than the control you wish to achieve. This principle holds true not only for attitude control but also for guidance navigation, center of mass control, and other aspects of the mission. For specific tasks like solar array pointing (SA) or using a high gain antenna (HGA), you need to perform computations based on the gain, frequency, and diameter to determine the required accuracy and suitable sensors and actuators. Similarly, for cameras, the field of view is an important consideration. The entire sizing process, including the pointing budget, is designed to ensure achievable performance, and it helps in verifying compliance with pointing requirements.

6.3.1 Control Modes

AOCS (Attitude and Orbit Control System) modes are designed to perform specific tasks with varying characteristics:

- Safe Hold Mode: Tasks: Detumble the satellite and determine its attitude, point the solar panels towards the Sun. Characteristics: Utilize the most reliable hardware, minimize power consumption, rough pointing is acceptable, fully autonomous operation.

- Standby Mode: Tasks: Point the solar panels towards the Sun, align antennae towards Ground Station, reduce drag during non-operative phases, use a radiator for thermal dissipation, and other tasks depending on the mission. Characteristics: Involvement of more complex hardware than Safe Hold Mode, better pointing accuracy than Safe Mode, but potentially not as precise as in Operational Mode. Typically autonomous with different guidance, navigation, and control (GNC) functions from Safe Mode.

- Operational Mode: Tasks: Fulfill the primary mission objectives such as image acquisition, antenna pointing, etc. Characteristics: Utilize the most precise hardware available, achieve highly accurate pointing, may involve complex filtering or control techniques. Typically ground-assisted for optimal performance.

- Orbit Control Mode: Tasks: Perform orbital maneuvers to control the satellite's orbit. Characteristics: Propulsion systems involved, need to control thrust-induced parasitic torques. Typically ground-assisted to ensure precise orbital adjustments.

- Transfer Mode: Tasks: Control the attitude and orbit during long transfers, for example, to reach GEO or interplanetary orbits. Characteristics: Involves propulsion systems to perform transfers, requires control of thrust-induced parasitic torques. Can operate autonomously, but may be ground-assisted for critical maneuvers.

- Mission Mode: Tasks: Fulfill specific mission objectives beyond routine operations. Characteristics: Diverse tasks based on the mission's unique goals, may involve various operational modes with specific hardware and control requirements.

Each mode is tailored to meet specific mission requirements, balancing autonomy, hardware complexity, and precision of pointing and control techniques. Ground assistance is often used to ensure optimal performance and safety during critical operations. Control modes are established based on several important factors:

- Flexibility: This refers to the system's ability to adapt and address various circumstances effectively, without ambiguities. It involves providing simple and efficient configuration options to handle different situations.

- Autonomy: The control mode should be capable of operating autonomously, without the need for constant ground intervention. This ensures that the system can function independently, minimizing the need for constant human oversight.

- Redundancy: It is essential for the control mode to maintain operability even in the event of hardware component failures. Having redundancy ensures that the system can continue to function, even if some hardware components become non-functional.

- Performance: The control mode's performance is measured by its capacity to deliver control with diverse control and computing capabilities. It should be able to handle various tasks efficiently and effectively.

Overall, the selection of control modes takes into account these key factors to ensure a well-balanced and robust control system that can meet the mission's requirements. Control modes are defined based on the mission profile and objectives of the ADC (Attitude Determination and Control) system. Each mode corresponds to a specific action or condition of the spacecraft's ADC system. There are standard modes that are commonly used, as well as special modes that depend on the mission's unique requirements and payload operations.

Standard Modes:

- Orbit Insertion: a. Occurs during and after the boost phase when the spacecraft is brought to its final orbit. b. Options include: spacecraft control, simple spin stabilization, or full spacecraft control.
- Acquisition: a. Used when the spacecraft is released from the mothership. b. Involves the initial determination of attitude and stabilization of the vehicle. c. Can be used to recover from emergencies.
- Normal On-station: a. Applied for most of the mission and drives the system design requirements. b. Allows for reorienting the spacecraft as needed.
- Contingency or Safe Mode: a. Applied in emergencies or unexpected situations.

Special Modes:
Special modes are tailored to specific mission profiles, operations, and payload requirements. A detailed list of these modes must be defined, outlining the unique requirements for each operational mode.
It's important to consider that increasing the number of required modes can lead to higher complexity and costs. Additionally, verifying and tracing the requirements during the entire life cycle progress becomes more challenging with an increased number of modes. Careful analysis and consideration of the mission's needs are essential to strike a balance between the number of modes and system complexity while ensuring optimal ADC performance throughout the mission.

6.3.2 Transition between modes

Control mode transitions must be integrated with system mode transitions. These transitions can occur autonomously, automatically, or be commanded from the ground.

- Autonomous Transitions: Event-driven and commanded by internal logic (e.g., FDIR - Fault Detection, Isolation, and Recovery). Ensured to never put the spacecraft in danger.
- Automatic Transitions: Time-driven and executed after a specific task is completed or when a predefined schedule is ongoing.

To achieve a well-designed control mode architecture and transitions, Finite State Machines (FSM) can be utilized. FSMs are mathematical models that represent abstract system behaviors:

- FSM can exist in one of a finite number of states at any given time.
- Transitions between states are triggered by specific inputs or events.
- FSMs are dynamic, discrete, and finite in nature.

By leveraging FSMs, spacecraft control systems can be efficiently organized, ensuring smooth and safe transitions between different operational modes based on internal logic, external events, or predefined schedules.

6.3.3 Pointing budget

The pointing budget is a compilation of essential information, including:

- Mission-specific pointing directions and rates requirements for each control mode.
- Performance requirements for achieving the desired pointing directions and rates.

This budget comprehensively documents all the possible pointing directions and rates to be attained, along with the corresponding performance expectations. Consequently, the spacecraft's Attitude Determination and Control System (ADCS) design aims to identify and implement the most suitable solution that ensures compliance with the defined pointing budget requirements. The chosen ADCS design should effectively meet the performance criteria outlined in the pointing budget to achieve accurate and precise pointing for the mission's success. The types of errors that can occur in a system are categorized as follows:

- Systematic Errors: These errors have a consistent and predictable pattern, leading to consistent deviations from the true values.
- Random Errors: These errors are unpredictable and vary randomly in magnitude and direction, causing fluctuations around the true values.
- Parametrization Errors: These errors arise due to inaccuracies in the parameters used for calculations and modeling. They include:
  - Bias: Constant offset from the correct value.
  - Scale Factor Error: Incorrect scaling or sensitivity of measurements.
  - Nonlinearity: Deviation from a linear relationship between inputs and outputs.
  - Asymmetry: Unequal response around a reference point.
  - Noise: Random variations introduced during measurement or processing.
  - Quantization: Errors due to discrete representation of continuous data.
- Time Response Errors: These errors are related to the time it takes for a system to respond to changes. They include:
  - Delay Time: Time lag between a change in input and the corresponding output response.
  - Dead Time: The period where the system does not respond to changes in input.

Identifying and understanding these various types of errors is crucial for system analysis, calibration, and improvement.

6.3.4 Quantify disturbance torques

- External Disturbance: External disturbances affecting the spacecraft can be of two types: cyclic or constant. The following are examples of external disturbances:
  - Gravity Gradient
  - Atmospheric Drag
  - Solar Radiation and Pressure
  - Magnetic Moment
- Internal Disturbance: Internal disturbances, on the other hand, can be constant or dynamic and arise from various sources within the spacecraft. Some examples include:
  - Actuator Misalignment (e.g., thrusters, reaction wheels)
  - Sensor Misalignment (e.g., gyroscopes, magnetometers)
  - Thruster Misalignment (resulting in deviations of approximately 0.1-0.5 degrees)
  - Mismatch of Thruster Outputs (leading to unbalanced torques when firing sets of thrusters, typically within a range of -5
  - Uncertainty in Center of Mass (usually within 1-3 cm)
  - Structural Dynamics (associated with arrays, booms, appendages, etc.)
  - Thermo-Structural Shocks (encountered during eclipse entry/exit)
  - Fluid Slosh (movement of fluids within the spacecraft)

These disturbances can significantly impact the spacecraft's attitude and trajectory and must be accounted for in the design and operation of the spacecraft's attitude control and determination system.

Gravity gradient

During phase A, the torque generated is modeled to have a static contribution, considering the worst-case conditions. This static contribution remains constant for scenarios involving planet orientation. However, for inertial pointing, the torque may exhibit a cyclical pattern. The magnitude of the torque depends on several factors, including the height of the system, the inertia matrix (which describes the distribution of mass and moments of inertia), and the misalignment between the system's orientation and the nadir direction (the direction pointing directly down to the Earth's center).

$$F_g = -\frac{\mu m}{R^2} \tag{6.1}$$

$$T_{gX} = \frac{3\mu m}{2R^3}|I_Z - I_Y| sin(2\theta_Y) \tag{6.2}$$

$$T_{gY} = \frac{3\mu m}{2R^3}|I_Z - I_X| sin(2\theta_X) \tag{6.3}$$

with X, Y,Z roll, pitch, yaw.

Aerodynamic drag

For phase A, the torque generated is modeled with a static contribution in worst case condition. It is constant for planet orientation or cyclical for inertial pointing. It depends on the misalignment with the center of mass wrt the center of pressure, air density, area to mass ratio, drag coefficient and velocity. Different parts of the satellite have different drag coefficient.

$$T_d = F_d(c_p - c_g) \tag{6.4}$$

$$F_d = \frac{1}{2}\rho C_d A V^2 \tag{6.5}$$

SRP

During phase A, the torque generated is modeled with a static contribution, particularly in the worst-case conditions. The behavior of the torque varies depending on the type of orbit being considered. For sun-synchronous orbits, the torque remains constant over time. However, for planet-oriented orbits, the torque exhibits a cyclical pattern, which means it varies periodically. The magnitude of the torque is influenced by several factors, including:
1) Solar activity: The intensity of solar radiation and the resulting pressure on the spacecraft surfaces affect the torque generation.
2) Area-to-mass ratio: This ratio describes the relationship between the spacecraft's exposed surface area and its mass. A higher area-to-mass ratio leads to a higher torque.
3) Misalignment between pressure and mass centers: If the pressure center and mass center of the spacecraft do not coincide, it leads to torque generation.
4) Spacecraft material: The material properties of the spacecraft surfaces can impact the torque produced under solar radiation pressure.
Considering these factors is crucial during phase A to accurately model the torque and its effects on the spacecraft's attitude and trajectory. Proper analysis and understanding of these parameters help in designing effective Attitude Control and Determination Systems (ACDS) for the mission's success.

$$M_s = F(c_{ps} - c_g)(1 + q) \tag{6.6}$$

6.3.5 Magnetic field

During phase A, the torque generated is modeled to include a static contribution, specifically in the worst-case conditions. This modeling is particularly relevant for missions around Jupiter, Saturn, and Earth. Additionally, it can also be applicable to Attitude Control and Determination Systems (ACDS) in polar orbits, where torque can be generated in various directions. The magnitude of the torque, denoted as Mm, is influenced by factors related to the spacecraft's residual dipole (D) and the interaction with the planet's magnetic field (B). The spacecraft's residual dipole is a result of internal perturbations and is obtained from tests. It is initially assumed as 1 Am^2 but is later accurately computed in phase B. The equation for the magnetic field (B) is modeled statically and takes into account parameters such as the magnetic field at sea level, the planet's radius, the height of the orbit, and the latitude. These factors play a significant role in determining the magnetic field's characteristics during the mission. As the mission progresses from phase A to phase B, more precise values and models for the spacecraft's residual dipole and the magnetic field are calculated, ensuring accurate torque calculations and successful mission execution.

$$M_m = DxB \tag{6.7}$$

$$D = NIAu_{n_{coil}} \tag{6.8}$$

6.3.6 ADCS architecture

The proposed method consists of different approaches, each offering specific advantages based on the application requirements.

- Zero Momentum: Utilizes 3 wheels, providing unrestricted pointing and excellent maneuverability. Thrusters can be employed for precise pointing and high-rate maneuvers, without constraints. Well-suited for orbits below 1000 km (for Earth missions).

- Gravity Gradient + Momentum Bias: Effective for orbits below 1000 km (Earth missions). The roll and pitch axes can be actively controlled. The yaw axis remains stabilized using a momentum wheel. Stable configuration is ensured when $I_pitch > I_roll > I_yaw$, and sometimes when $I_roll > I_yaw > I_pitch$.

- Momentum Bias: Implemented using a momentum wheel spinning at a nearly constant high speed. Enables local vertical pointing or inertial targeting. Provides inertial stiffness in two axes. Control of the wheel speed allows for control in the third axis.

- Spin Stabilized: The spacecraft (s/c) is stabilized along an axis by maintaining a constant angular velocity around it. In the absence of disturbances, the angular momentum remains constant. Perpendicular disturbances cause the rotational axis to process. Can be oriented inertially in any direction. Repointing is slower using torquers and faster using thrusters. Parallel disturbances change the angular momentum modulus, limiting translational maneuvers to occur only along the spinning axis.

- Dual Spin Stabilized: This approach involves spinning the main mass while a platform with payloads or antennas is de-spun. The method is limited only by the articulation on the de-spun platform, enabling controlled motion and stability of the payload.

By incorporating these methods, the spacecraft can achieve stable and precise attitude control, making it suitable for a wide range of applications depending on the specific mission requirements.

Pro and Cons

- Zero momentum:

  - Pros: Unlimited pointing capability in any direction. Excellent pointing accuracy, ranging from 0.0001 to 1 degree (from wheels to thrusters). Suitable for missions requiring substantial power. Provides high maneuverability. Enables precise active thrust vector control. Suitable for applications needing microgravity control.

- Cons/Limitations: Hardware complexity due to the need for intricate systems. Heavy and power-consuming, leading to increased power requirements. Prone to failures, requiring careful redundancy and backup systems. Expensive to implement and maintain.

- Gravity gradiant+Momentum bias:
  - Pros: Suitable for achieving modest pointing accuracy. Economical and cost-effective. Robust and reliable design. Requires limited power consumption. Effective for missions operating below a given altitude (e.g., 1000 km on Earth).
  - Cons/Limitations: Lifetime is limited by the wheel bearings. Pointing accuracy limited to approximately ± 5 degrees. May not be suitable for missions requiring highly precise pointing.

- Momentum bias:
  - Pros: Ideal for long-duration missions. Cost-effective and budget-friendly. Provides good pointing accuracy in one axis, particularly in pitch (0.01 degrees). Well-suited for missions where high accuracy in non-wheel axes (roll/yaw) is not critical.
  - Cons/Limitations: Limited pointing accuracy in non-wheel axes, approximately 1 degree for roll and yaw. May not be suitable for missions requiring precise control in all axes. Each method offers distinct advantages and trade-offs, and the choice depends on the specific mission requirements, budget constraints, and desired performance characteristics.

- Spin stabilize:
  - Pros: Well-suited for scanning instruments, making it useful for certain types of missions. Simple and cost-effective design. Particularly useful during thruster burns and for propellant control. Requires low pointing accuracy, typically ranging from 0.1 to 1 degree on two axes. Enables tight control of moments of inertia, ensuring stable performance. Suitable for solar cells mounted on the spacecraft body, although with lower efficiency. Effective slew rates of less than 0.5 degrees per second.
  - Cons/Limitations: Provides low pointing accuracy on two axes, limiting its suitability for certain precision missions. Requires careful attention to ensure the greatest moment of inertia is around the spinning axis to avoid reversing the vehicle's rotation. Forbidden to perform body pointing of payloads or antennas. Offers poor maneuverability due to its inherent design.

- Dual spin stabilized
  - Pros: Suitable for both scanning instruments and pointing capabilities, making it versatile for different mission objectives. Useful for propellant control and during thruster burns. Provides stable and controlled performance during operations.
  - Cons/Limitations: Implementation is expensive and complex due to the need for two spacecraft sections. Prone to potential failures, requiring careful redundancy and backup systems. Requires meticulous attention to distributing

inertial properties between the two spacecraft sections to ensure proper performance and stability.

6.3.7 Preliminary sizing

First we start from the inertia properties. Then we define:

Sensors

− **Sun sensors** (0.005 − 3 deg): Sun sensors offer the Sun's direction vector but not the complete attitude state of the spacecraft. They are not operational during eclipses. Initialization is not necessary. Absolute attitude reconstruction requires the Sun's ephemeris data.

−**Star Sensors** (0.0003 − 0.01 degrees): Star sensors provide two or three-axes attitude state information. They require initialization, typically taking a few seconds. Not suitable for accurately tracking large rates of rotation. Star scanners are used on spinning spacecraft, where light from various stars passes through multiple slits in the scanner's field of view. Star trackers are used on three-axis stabilized spacecraft and utilize CCDs to capture images of the star field, which are then compared with a star catalog to determine attitude information.

−**Horizon (Earth) Sensors** (0.05−1 degree): Horizon sensors provide Earth-relative information, specifically for Earth-pointing spacecraft. Accurate orbital data is required for absolute attitude reconstruction.

− **Magnetometers** (0.5−3 degrees): Magnetometers offer the magnetic field direction vector and magnitude, but not the complete attitude state, unless complex reconstruction algorithms are employed (recently used for small spacecraft). Accurate magnetic models are necessary for absolute attitude reconstruction. Crucial for the usage of magnetic torquers, which rely on the knowledge of the magnetic field for attitude control.

− **Gyroscopes**: Gyroscopes provide measurements of angular rates, enabling the determination of attitude evolution. To obtain absolute attitude states, an external attitude reference is required. Gyroscopes can be utilized for attitude propagation when the external attitude reference is lost or unavailable. On-board calibration of gyroscopes, such as bias estimation, is essential for accurate attitude propagation. The most relevant performance factors for gyro selection are bias drift stability and bias drift, as these impact the accuracy and reliability of attitude measurements. Careful consideration of gyro performance is crucial for mission success.

− **Global Navigation Satellite Systems** (GNSS): GNSS provides orbit position information, which may be necessary to feed internal models used for attitude reconstruction, such as magnetic field models and Earth absolute position. Differential GNSS measurements can be utilized for attitude reconstruction, achieving accuracy in the range of 0.25 to 0.5 degrees with a 1-meter differential antenna baseline.

− **Accelerometers**: Accelerometers provide acceleration measurements that are valuable for orbit propagation and orbit determination filtering when used in conjunction with GNSS, for example, GPS + Accelerometers in Inertial Navigation System (INS) applications. However, accelerometers do not directly sense the external gravitational acceleration, necessitating the use of a gravitational model (e.g., Kep, J2, Spherical harmonics) for orbit propagation. Accelerometers are particularly useful for precise orbit maneuvers actuation and accurate on-board orbit determination.

– **Inertial Measurement Unit** (IMU): An IMU comprises three gyroscopes and three accelerometers, providing a comprehensive solution for complete attitude and position sensing. This combination of sensors allows for accurate determination of the spacecraft's orientation and position in space.

Any combination of two vector measurements from the aforementioned sensors allows the determination of four independent measurements, enabling the determination of the spacecraft's orientation in space. The choice of the sensor depends on the mission's requirements, spacecraft design, and accuracy needed for attitude control and determination.

Actuators

These actuator systems play a critical role in spacecraft attitude control, providing the necessary torque and control authority to maintain precise orientation, counteract disturbances, and execute required maneuvers during the mission. Actuators must possess enough torque authority to effectively counteract disturbances. Control Authority is calculated as the difference between the Control Torque and the disturbance torque. For instance, if CT = 2DT, a 100 % Control Authority margin is achieved. This margin ensures that the actuators have ample capability to overcome disturbances and maintain precise control over the spacecraft's orientation and trajectory.

– **Reaction Wheels**: Equipped with torque motors and high-inertia rotors, reaction wheels maintain a nominal zero momentum. Each reaction wheel can spin in either direction and provides control over one axis. Torque is applied by adjusting the rotor's speed, allowing for control to counteract disturbances and perform slew maneuvers.

– **Momentum Wheels**: Similar to reaction wheels but with a nominal non-zero spin, momentum wheels provide nearly constant momentum. Stiffness is achieved on two axes, while the motor torque controls the pointing direction along the third axis.

– **Control Moment Gyros** (CMG): CMGs generate torque by utilizing torque motors to alter the spin axis's direction. They rotate at high speeds with small inertia. CMGs are movable angular momentum vectors and can be controlled through motors in the gimbals. They are suitable for large torques or fast maneuvers. Although more massive, CMGs are more efficient than reaction wheels.

– **Thrusters** For the initial sizing, we assume constant acceleration, and the required torque for the maneuver is determined as follows: By utilizing thrusters, we can calculate the mass of propellant needed to provide the necessary torque. It's important to note that during thruster firing, a $3-$axis control system should be implemented to address disturbances caused by misalignment of the applied force and the exhaust plume. This control system ensures stability and accuracy during the maneuver. They can use hydrazine (0.5

N to 9000 N) or cold gas (< 5 N).

− **Magnegtic torques** Magnetic torques, or magnetic couples, are used as actuators in space applications to control the orientation and rotation of a satellite or other spacecraft. These torques are based on the interaction between the Earth's magnetic field (or another source) and the intrinsic or artificial magnetization present onboard the satellite. Here's how magnetic torques function as actuators:

- Magnetorquers: Magnetorquers are electromagnetic devices onboard the satellite that generate a controlled magnetic field. This field interacts with the Earth's magnetic field, generating a magnetic force that acts as a torque on the satellite. This torque can be used to change the satellite's orientation.
- Orientation Change: By applying a current to the magnetorquers, it's possible to generate a torque that affects the satellite's rotational axis. Modulating the intensity and direction of the current allows for control of the satellite's orientation. For example, to orient the satellite along the roll axis, a current can be applied to a magnetorquer positioned along the pitch or yaw axis.
- Rotation Control: Using combinations of magnetorquers positioned along different rotational axes, it's possible to control and adjust the satellite's rotation speed. Furthermore, by changing the direction of the current in the magnetorquers, it's possible to reverse the torque and influence the satellite's rotation.

With them there is a propellant savings because they allows for orientation and rotation control operations without using propellant. This can extend the operational life of a satellite by reducing the propellant consumption required for control maneuvers. But they have limitations, such as their dependence on the Earth's magnetic field or external sources, and their effectiveness can vary in different orbits and spatial regions.

6.3.8 ADCS Maneuvering sizing

The first two assumptions in order to the ADCS maneuvering sizing is that are considered only constant angular velocity and single axis maneuvering. Doing maneuvering with thrusters the amount of propellant required is inversely proportional to the specific impulse (Isp) or the arm of the force (L). The torque applied is represented by nFtb, where n is the number of thrusters, F is the force generated, and tb is the burning time. When it's a maneuvering limit cycle consider that the s/c is swung between present angular limits. The correction pulse should be kept as smaller as possible as it must be reversed. This in the case there aren't external torques. In the case that they are present, the cycle is one side limited and the correction pulse should be kept as smaller as possible as before.

slew maneuvering

In order to get a desired angle there are two possible options: constant acceleration, coasting and constant braking, or constant acceleration and constant braking. (For formulas using slides). With RW it's the same but with coasting = 0. Consider that wheels may need to be de−saturated for unbalanced angular momentum.

Chapter 7

Electrical and Power subsystem

The Power System Purpose is:

- Ensuring an uninterrupted supply of electrical power to spacecraft loads throughout all mission phases and modes.
- Managing and allocating electrical power to various spacecraft components.
- Meeting the demands of both average and peak electrical power needs.
- Supplying converters for alternating current (AC) and regulated direct current (DC) power distribution.
- Enabling command, telemetry, and control of Electrical Power System (EPS) health and status through ground stations or autonomous systems.
- Safeguarding the spacecraft payload from potential EPS failures.
- Mitigating transient bus voltage fluctuations and preventing bus faults.

so the power System Components are:

1. Energy Sources: involves converting diverse energy sources into electrical energy. This depends on the trajectory and orbit.
2. Energy Storage:captures surplus energy from the primary source to ensure a continuous power supply when the primary source is unavailable or during periods of high demand. It's impossible not to have a storage, because this allows safety and a surplus of power for short periods of time.
3. Power Distribution: encompasses a collection of components dedicated to interfacing with loads and sources.
4. Power Regulation and Control: regulation and control are achieved by adjusting voltage and current to cater to load specifications. Regulation is essential due to factors such as load requirements, mission profile fluctuations, power source deterioration, and the management of battery charge and discharge cycles.

It's important to define the phases and the modes of the mission before computing the power budget. In order to do that it's essential to know all the components. So in order to define and address Power, some requirements are necessary. The information needed are:

- Mission profile and operational lifespan.
- Solar irradiance and eclipse durations (if applicable).
- Spacecraft design and layout.
- Mission phases and modes.
- Power peak and average demands per subsystem.
- Power peak and average demands per payload.
- Alignment with launcher requirements (if applicable).
- Thermal environment considerations.
- Compliance with international regulations.

From these point is possible to identify, select and size the system components:

1. Identify subsystem (s/s) requirements, so fill the power budget per phase/mode and generate the Sun irradiance profile.
2. Select and size power source, so define EOL power needs, the type of power source and the size, mass and configuration.
3. Select and size energy storage, so define battery life cycle and eclipse profile, the battery type and the battery mass, volume and configuration.
4. Select power regulation and control, so the regulation architecture, the control architecture and the bus voltage.

At this point we are able to obtain outputs such as:

- EPS system schematic that outlines the fundamental elements of the spacecraft's Electrical Power System.
- Power Budget Allocation, so the allocation of power budget for each operational phase, ensuring optimal energy distribution.
- System Budgets, so comprehensive breakdown of mass and power budgets, enabling efficient resource management.
- equipment List, so enumeration of all components and devices within the EPS, facilitating clear identification and tracking.
- EPS System Configuration so visual representation of the interconnected arrangement of components within the Electrical Power System.

The power budget allocation serves as a foundational step in designing the spacecraft's Electrical Power System.

7.1 Electric power sources

Various primary sources of electrical energy can be used in various space applications, such as satellites and space probes.

1. Primary Batteries: These are batteries that store chemical energy and release it as electrical energy. They are particularly useful for short-term missions or providing power in emergency situations. Once depleted, these batteries cannot be recharged and must be replaced.

2. Solar Cells – Photovoltaic Generator: Solar panels, also known as photovoltaic cells, directly convert sunlight into electrical energy. They are commonly used in long−term space missions as they can provide continuous power during exposure to sunlight.

3. GPHS – RTGs (General Purpose Heat Source − Radioisotope Generators): These generators use the heat produced by the radioactive decay of isotopes like plutonium−238 to generate thermal energy, which is then converted into electrical energy through devices called thermopiles.

4. Fuel Cells: Fuel cells directly convert the chemical energy of a fuel (such as hydrogen) into electrical energy, along with a byproduct of water or other harmless gases. They are often used in space missions to extend autonomy and can be fueled from external sources.

5. Solar Dynamics: This term could refer to systems or technologies that harness the dynamic behavior of the sun to generate energy, but further context is needed for a more precise explanation.

6. Nuclear Reactors: These devices utilize nuclear fission to generate heat, which is then converted into electrical energy. They are very powerful but complex and require rigorous management.

7.1.1 Selection drivers

Factors that drive the selection of power sources for space missions:
Power Level and Power Density: The amount of electrical power needed by the spacecraft and how densely it must be generated. Some missions require higher power outputs, while others can operate with lower power levels. So this means the kind of SP or the kind of batteries
Mission Lifetime: The expected duration of the mission plays a significant role. Longer missions may require power sources with greater longevity and sustainability. This factor acts on the aging of SP or semi−life of radioactive materials.
Source Availability: The accessibility and availability of the chosen power source, considering whether it can be reliably obtained for the mission's entire duration.
Sensitivity to the Sun Distance: How the power source's performance is affected by the varying distance between the spacecraft and the sun. Solar power, for example, becomes less effective as a spacecraft moves farther from the sun.
Secondary Source Needs and Requirements: Whether additional power sources are needed to supplement the primary source, and the specific criteria these secondary sources

must meet.
Rad-Hardness (Tolerance): The power source's ability to withstand and continue functioning under the harsh radiation environment of space.
Degradation: How the power source's efficiency and performance degrade over time due to factors such as radiation exposure, temperature changes, and operational stress.
Reliability and Security Requirements: The power source's ability to provide consistent and secure power, especially for critical systems, without compromising the mission's success.
Cost: The financial implications of selecting and implementing a specific power source, including factors like development, deployment, and operational costs.
Each of these factors influences the decision-making process when choosing the most suitable power source for a space mission. Balancing these drivers is crucial to ensuring the mission's success, longevity, and optimal performance.

7.1.2 Primary batteries

Two types of batteries used in space applications, along with the advantages and disadvantages of each type, and some of the applications in which they are used:

Silver-Zinc Batteries

Benefits: Widely adopted and mature technology (TRL9), high specific energy (90-230 Wh/kg), acceptable for launch vehicles but need replacement in case of extended launch delays.
Drawbacks: Short lifespan (months) if stored with electrolyte; can be stored dry for a limited period, costly, narrow temperature range: from 0°C to 40°C, reduced lifecycle (200 cycles).
Applications: Ariane V

Lithium Batteries (Sulphur Dioxide, Thionyl Chloride)

Benefits: Used in scientific applications requiring long storage life, excellent specific energies (up to 400 Wh/kg), wide temperature range: from -60°C to 70°C, low discharge rate, long lifespan (years).
Drawbacks: Hazardous batteries, reactive with water and nitrogen, with potential gas release. Applications: MER Lander, missions like Genesis, Stardust, Galileo probe, Philae, Huygens, Mascot.

In essence, the text explains the Silver-Zinc and Lithium battery types used in space. Both types have their advantages and disadvantages, along with specific applications where they are employed. For instance, Silver-Zinc batteries are technologically mature and offer high specific energy, although at the cost of a short lifespan and a limited temperature range. Lithium batteries have a longer lifespan and a broader temperature range, but precautions are necessary due to their chemical reactivity.
Specific energy refers to the energy stored in a battery relative to its weight. It is a measure of how efficiently a battery can convert and store chemical energy into electrical energy. In the context of primary space batteries, specific energy is a key factor in selecting the appropriate energy source for a mission. The higher the specific energy, the more energy a battery can store relative to its weight, allowing missions to last longer

and/or perform more complex tasks. For example, lithium batteries are known to have high specific energy, reaching up to 400 Wh/kg. Silver-Zinc batteries are another option with a good level of specific energy, ranging from 90 to 230 Wh/kg, depending on configurations and technologies used. In summary, all four batteries have high energy densities and are suitable for various space applications. Lithium batteries (sulfur dioxide, carbon monofluoride, thionyl chloride) generally offer higher energy densities compared to Silver-Zinc batteries. Lithium batteries have broader operating temperature ranges than Silver-Zinc batteries, but Thionyl Chloride batteries have the widest range. Lithium batteries offer higher discharge voltages than Silver-Zinc batteries, with Thionyl Chloride batteries having the highest discharge voltage. The storage life of lithium batteries is typically longer than that of Silver-Zinc batteries.

7.1.3 Solar arrays

A solar panel reserves 10% of its surface area for essential connections. It also requires structural and thermal controls, along with protective glass, which adds to its overall weight. Interestingly, the efficiency of a solar panel increases as its temperature decreases. There are three primary types of solar panels:

- Mono-crystalline silicon cells: These cells have a somewhat lower efficiency (around 13%), but they offer high resistance to radiation and age relatively slowly over time.
- Multi-junction cells: These cells provide better efficiency (around 30%), but they are more vulnerable to degradation due to radiation.
- Thin-film cells: These cells are flexible and thin, allowing for the creation of rollable solar panels like those utilized in the DART Mission.

Multi Junction solar panels employ a range of semiconductors to capture a wider spectrum of frequencies. The process involves placing two semiconductors, one of type n and the other of type p, in close proximity and applying a voltage. When sunlight strikes the panel, electrons move between these layers. By adjusting the applied voltage, the generated current can be modified, a feature evident in the solar cell's characteristic curve. This curve can change due to factors like aging, orientation, and temperature. Cells can be connected either in series or parallel, leading to increased voltage or current, respectively. In series-connected cells, different voltages can be obtained, typically 5V and 28V. Blocking diodes (colored red) and bypass diodes (colored green) play pivotal roles. Blocking diodes prevent failures or low power generation (umbra) in parallel arrays, ensuring the solar panel doesn't become a power drain. Conversely, bypass diodes protect against failures in series arrays, preventing interruptions that might lead to power losses or reverse currents. These diodes are crucial since they prevent the entire solar panel from losing power when a single cell is shaded or fails.

Performance of photovoltaic source

In aerospace applications, the performance of photovoltaic sources (PV) is influenced by various factors, including the angle of the sun, solar radiation, temperature, and distance from the sun. These factors have a significant impact on the current-voltage relationship in the current-voltage characteristic curves (IV curves) of solar panels. Let's explore how each factor affects the performance of photovoltaic sources in space:

- Effect of Solar Angle: The angle between solar rays and the surface of solar panels affects the amount of energy that strikes them. When solar rays hit the panel perpendicularly (an angle of 90 degrees), maximum efficiency is achieved. Deviating from the perpendicular angle reduces incident energy, leading to a decrease in generated current. This is particularly crucial in space missions, as the orientation of the spacecraft can vary, affecting the angle at which sunlight hits the solar panels.

- Effect of Solar Radiation: The amount of solar radiation reaching solar panels directly impacts the amount of energy converted into electricity. Near the Sun, solar radiation is intense and can result in increased current generation. However, as the spacecraft moves away from the Sun, solar radiation decreases, adversely affecting solar panel performance.

- Effect of Temperature: The temperature of the space environment can vary significantly. At higher temperatures, current generated by solar panels tends to increase, while voltage decreases, influencing the IV curve. Conversely, at lower temperatures, current decreases while voltage increases. This thermal variation can have a notable impact on the overall performance of photovoltaic sources.

- Effect of Solar Distance: The distance between the spacecraft and the Sun affects the amount of solar radiation that reaches the solar panels. Greater distances from the Sun lead to reduced solar radiation, resulting in a decrease in energy generated by the solar panels. This is especially relevant for interplanetary missions that move away from the solar system.

In aerospace applications, optimizing the performance of photovoltaic sources takes into account these factors, designing systems that can adapt to variations in solar angle, solar radiation, temperature, and solar distance. Understanding how these factors influence the current-voltage relationship is essential to ensure that solar panels provide the necessary energy throughout the duration of the space mission.

Degradation of power cell

The degradation of solar cells in photovoltaic panels is an inevitable phenomenon over time due to exposure to solar radiation, space radiation, and thermal variations. Degradation is particularly relevant in geostationary (GEO) and low Earth orbits (LEO). Let's explore how solar cell degradation varies over time and altitude in these two types of orbits:

- Geostationary Orbit (GEO): In geostationary orbits, where satellites are positioned at a fixed altitude above a specific location on Earth, solar panels are exposed to intense and relatively constant solar radiation. Over time, continuous exposure to solar radiation can lead to the degradation of solar cell materials, gradually reducing their efficiency. Degradation is generally slower compared to lower orbits due to less dynamic conditions.

- Low Earth Orbits (LEO): In low Earth orbits, where satellites orbit closer to Earth, environmental conditions are more dynamic. Satellites in these orbits pass through Earth's shadow more frequently, experience greater thermal variations, and are exposed to terrestrial and solar-origin radiation. These conditions can accelerate solar cell degradation. Intense solar radiation, along with charged particles and cosmic radiation present in space, can damage solar cell materials over time. The frequent variation between periods of solar illumination and terrestrial darkness can subject solar cells to cyclic thermal stresses, contributing to degradation.

In both orbits, solar cell degradation is a factor to consider in the design and planning of space missions. Designers must balance the initial efficiency of solar cells with the expectation of degradation over time, ensuring that the system still has enough energy to perform required functions at the end of the mission's lifespan. The use of energy management algorithms and mitigation strategies can help maximize the operational efficiency of solar panels despite degradation.

From cell to array

Coverglass serves several functions including thermal control, reduction of reflection, protection from radiation, debris, and atomic oxygen. The back surface reflector acts as an electrical insulator. Additionally, the adhesive used serves the purpose of absorbing thermal stress. The primary objectives are as follows:

1. Maximizing absorption of solar energy.
2. Maintaining lower temperatures whenever feasible.

Consequently:

1. Enhancing the emissivity (ϵ) of the coverglass to increase its energy absorption capacity while reducing reflectance (ρ).
2. Increasing thermal conductance (K) from the top to the bottom.
3. Elevating the emissivity (ϵ) of the rear panel.

The most prevalent packaging style for rectangular arrays is the flat packaging, which is less efficient but offers enhanced safety. The integration process is not undertaken by the same company responsible for cell production. A prospective solution aimed at improving solar panel efficiency, currently in development but not yet launched, involves incorporating inclined mirrors to amplify energy capture. This approach results in increased complexity and mass, accompanied by a reduction in the angle of visibility. Alternatively, magnifying lenses (such as Fresnel lenses) can also be utilized, as demonstrated by the launched SCARLET project. When considering materials:

- Aluminium exhibits high thermal expansion and has the potential for eventual failures, but is cost-effective.
- Carbon Fiber boasts low thermal expansion and greater rigidity, albeit at a higher cost.

Regarding shingling packaging, the pros are offering a high packaging factor, and the cons are maintenance proves to be challenging.

Solar concentration

Solar concentrators are devices designed to augment the solar flux directed onto photovoltaic cells. However, they come with certain limitations:

- Temperature Rise: They can lead to an increase in temperature.
- Sensitivity to Aspect Angle: Their efficiency can be influenced by the angle of incidence.

To address these issues, potential solutions include:

- Lateral Mirrors: Mirrors placed strategically to redirect sunlight.
- Fresnel's Lenses: An example is the SCARLET project utilized in Deep Space 1 by NASA.

Focusing on Fresnel's lenses for solar concentration, they offer several advantages:

- Mass Reduction: They lead to a significant reduction in mass, up to 50
- Array Area Reduction: They can slash array area requirements by 85
- Cost Savings: They contribute to cost containment efforts.
- Enhanced Output: The adoption of Fresnel's lenses can result in a notable increase in output power, ranging from 15% to 20%.

Sizing process

Initial stage: Identifying the Worst-Case Scenario. Begin by pinpointing the worst-case scenario, which encompasses the following parameters:

1. Te: Maximum duration of eclipse.
2. Td: Corresponding period of daylight.
3. Pe: Peak power requirements during eclipse.
4. Pd: Peak power requirements during daylight.
5. Proceed to calculate the total power demand, Psa, while factoring in efficiency coefficients for both eclipse (Xe) and daylight (Xd). This calculation involves the completion of a Power Budget table, which should encompass various operational modes.

Additionally, consider the requisite voltage from the loads. This step aids in sizing the bus effectively, determining the appropriate number of strings or cells for the solar array and battery configurations.
Gaining an understanding of the size of your solar arrays becomes crucial, particularly when dealing with smaller structures. This understanding is pivotal in deciphering the power requirements and battery needs for your system. Two key parameters, Xe and Xd, are influenced by the power control architecture—specifically, whether it employs Direct Energy Transfer or Peak Power Tracking. In determining your power needs, you aggregate the energy demanded by both the battery and the daylight-utilizing sources. The

parameter Pd is contingent on the designer's choices, considering operations such as the simultaneous activation of ADCS sensors and actuators. While a sensor might function independently, actuators typically require sensor input for activation. Sequencing matters too, like having one payload activate before the other, allowing for manipulation of cumulative power across mission phases and modes. Normalization enters the picture, hinging on the efficiency of internal elements, electronics, or components that regulate current and generated power. This normalization aligns with the active time of the source. In the context of solar panels, daylight serves as the normalization basis since energy generation is limited to these hours. Neglecting this aspect leads to full battery discharge, rendering batteries unavailable during eclipse or battery mode. Regarding charging rates, higher rates induce elevated temperatures and subsequently impact charging efficiency. Thoughtful design is necessary for determining phases where power consumption, production, and availability are critical.

Stage two: Characterizing the Power Source.

- Proceed to delineate the characteristics of the power source, which are represented by efficiency (η) and degradation (d).
- For solar arrays, initiate calculations for power generation at the Beginning Of Life (BOL), denoted as PBOL.
- Incorporate the inherent degradation factor (Id) in the range of 0.49 to 0.88, along with P0 in W/m^2, which represents the specific power at 1 Astronomical Unit (AU) for the chosen solar cells.
- Estimate the degradation factor for the solar array, labeled as Ld.
- Calculate the power production at the End Of Life (EOL), referred to as PEOL.
- Proceed to determine the total area requisitioned for the solar array.
- Finally, establish the corresponding mass associated with the solar array configuration.

PBOL signifies the initial power output at the start of a mission, involving the integration of new cells on board. Po stands for power density (W/m^2), a parameter influenced by the distance from the sun—particularly relevant in the context of solar panels. Id, an index reflecting system integration, acknowledges that the available area might be limited. The effective cross-sectional area is represented by alpha: when the surface is at a 90° angle relative to the sun's direction, no power is supplied. During the sizing process, alpha is typically set around 20-25°. Factoring in degradation is crucial. We possess average annual degradation rates for distinct technologies (expressed as d, a percentage), accounting for the radiation effect. Triple junction cells degrade at a rate of 3% annually, while silicon-based cells degrade at 1%. For meticulous calculations, the SPENVIS tool can be employed. By inputting environmental particle flux and cell technology, it offers degradation estimates based on mission duration. With the end-of-life power (PEOL) known, the power requirement at mission completion, you can make an informed choice between solar cells and Radioisotope Thermoelectric Generators (RTGs). It's important to stress that sizing endeavors always focus on the mission's worst-case real scenarios, considering aspects like positioning, solar distance, achievable cell area, and mass. For larger missions, mechanisms for orienting solar panels are used, but these remain in the developmental phase and come with significant weight implications.

7.1.4 Nuclear power sources: RTG

A Radioisotope Thermoelectric Generator (RTG) is a device that converts the heat produced from the radioactive decay of isotopes into electrical energy using the thermoelectric effect. Here's how it works:

- Radioactive Decay: RTGs utilize radioactive isotopes like plutonium-238 as a source of heat. These isotopes undergo decay, emitting radioactive particles and gamma rays. This decay activity continues over time, producing a constant amount of heat.
- Thermoelectric Conversion: RTGs harness the thermoelectric effect, known as the Seebeck effect. This effect relies on the temperature difference between two junctions of different materials. One junction is exposed to the heat source (radioactive isotope), while the other junction is kept at a lower temperature. This temperature difference induces an electric current flow through the thermoelectric material.
- Electricity Generation: The electric current generated from the temperature difference between the thermoelectric junctions is collected and used as electrical energy. This is the energy that powers devices onboard the instrument or spacecraft.
- Cooling: Since thermoelectric efficiency is tied to the temperature difference between the junctions, it's important to maintain the cold junction at a lower temperature. Cooling systems or radiators are used for this purpose to dissipate excess heat generated by the junction exposed to the heat source.
- Efficiency and Longevity: RTGs can provide a continuous source of electrical energy over long durations due to the constant radioactive decay of isotopes. However, the efficiency of RTGs is relatively low compared to other energy sources like solar panels, as a significant portion of the heat produced by decay is not converted into electrical energy.

RTGs are often utilized in long-duration space missions, such as interplanetary probes or rovers on planets like Mars, where access to solar energy is limited or inefficient.

General purpose

Generating static electricity through the Seebeck effect involves harnessing thermal energy from the natural decay of radioactive isotopes. General Electric pioneered its production in the 1990s. This process revolves around utilizing Pu-238's thermal energy derived from its decay, boasting a half-life of 86.8 years. This emission is easily shielded. With a specific energy output of 7.7 W/kg (compared to solar arrays at 100 W/kg), a Beginning Of Life (BOL) output of 300 W is achieved from an initial 4400 Wt. The entire system weighs 55 kg, with the Pu content at 7.8 kg. SiGe thermoelectric couples are employed for this purpose. The cost is approximately 15,000 dollars/W. Ensuring safety mechanisms for launch accidents is crucial due to RTGs' substantial weight. However, RTGs have low efficiency and are considered a last resort when alternatives are unavailable. Solar cells maintain an efficiency of 30%, while RTGs exhibit 7% efficiency (20% for newer generations). Radiators are essential to dissipate the 93% of energy not converted into electricity. The goal is to transform thermal energy from radioactive decay into electrical power using thermionic effects. The two steps in this conversion process contribute to the system's notably low energy efficiency. In instances like the Perseverance rover, the Multi-Mission Radioisotope Thermoelectric Generators (MMRTGs) are employed. Positioned at the rear, they serve to dissipate surplus thermal energy and shield the onboard avionics. Shielding and thermal dissipation underscore the necessity of RTGs in certain configurations. Traditionally, RTGs were purchased as is in terms of voltage and mass (e.g., 300 W, 56 kg) without much flexibility. This is in contrast to solar cells, where voltage adjustments are feasible through connection manipulation.

Thermoelectric generators

This include two phases:

- The initial phase involves the generation of thermal energy via decay: In this phase, radioactive isotopes, such as plutonium-238, undergo radioactive decay, emitting particles and gamma rays as a byproduct. This continuous decay process releases a constant amount of heat energy. This thermal energy is the primary source of heat used in the thermoelectric generator.

- The subsequent phase involves the conversion of this thermal energy into electricity, achieved either statically or dynamically: After the thermal energy is generated through the radioactive decay, it is harnessed to create electrical power using the thermoelectric effect. This phase is where the thermoelectric generator comes into play. The thermoelectric effect, known as the Seebeck effect, involves two different materials or junctions with a temperature difference between them. The heat source, which is the radioactive decay heat, is applied to one junction, while the other junction is kept cooler. This temperature difference leads to the movement of electrons within the materials, creating an electric potential difference (voltage) across the junctions. This generated voltage creates an electric current when a circuit is connected between the two junctions. This electric current represents the conversion of the initial thermal energy into usable electrical energy. The subsequent use of this electricity powers the instruments or devices onboard the spacecraft or instrument that employs the RTG.

Power degradation

In RTGs, the amount of power they generate tends to decrease exponentially over time. This decrease is mainly attributed to the decay of the radioactive material used as fuel, such as plutonium. As the radioactive material decays, it emits particles and radiation, which leads to a reduction in the heat produced and, consequently, in the generated power. P0 and :

- P0: P0 represents the initial power output of the RTG at the beginning of its operational life, when the radioactive material is fresh and its decay has just started.

- (Tau): signifies the half-life period of the radioactive material. The half-life is the time it takes for half of the radioactive material to decay. This value is significant because it determines the rate at which the power output of the RTG will decrease over time.

As an example is Plutonium 238. It has a power density of 0.41 watts per gram (W/g) and a half-life of 86.8 years. This means that after 86.8 years, half of the initial amount of Plutonium 238 will have decayed.

Thermoelectric conversion

- Thermoelectric Conversion: Thermoelectric conversion is a process used to convert heat directly into electricity using a phenomenon called the Seebeck effect. This effect involves the generation of an electric voltage when there's a temperature difference between two junctions made of different materials. This process is used in thermoelectric generators (TEGs) to generate electrical power from a heat source, such as the heat produced by the decay of radioactive isotopes in RTGs.
- Hot and Cold Parts with Different Junctions: In a thermoelectric generator, there are two main parts: the hot part and the cold part. The hot part is exposed to the heat source, which could be the radioactive decay in the case of RTGs. The cold part is maintained at a lower temperature. Each part contains junctions made of different materials that exhibit the Seebeck effect. These materials have different electron behavior at different temperatures.
- Energy Conversion and Power Production: The temperature difference between the hot and cold parts creates a thermal gradient. This thermal gradient causes electrons to flow from the hot side to the cold side through the thermoelectric materials. As the electrons move, they generate an electric potential difference (voltage) between the two sides, resulting in an electric current when a circuit is connected.

This electric current is essentially the conversion of the thermal energy from the heat source into electrical energy. The greater the temperature difference between the two sides, the higher the voltage and the greater the potential power production.

Radiations

Radioactive materials emit different types of radiation:

1. α Particles: These particles are relatively easy to shield against. They are also efficient at converting their energy into heat. However, the emission of particles, which are helium nuclei, requires a container that can withstand the pressure they generate.
2. β Particles: These particles are less effective at converting energy into heat compared to particles.
3. γ Radiation: radiation involves high-frequency electromagnetic energy and is considered hazardous. It is more difficult to shield against compared to and particles.

These characteristics provide an overview of the distinct types of radiation emitted by radioactive materials and their effects on the operation and efficiency of RTGs.

Multu mission RTG

The emerging direction in RTG technology is the adoption of smaller units, exemplified by the Multi Mission RTG approach. While the fundamental concept remains consistent, these RTGs are scaled down to enhance the compatibility with various missions. This size reduction is a prevailing trend within this technology.

SRG

The term "SRG" refers to "Stirling Radioisotope Generators," which are a type of thermoelectric generator used to convert heat produced by radioactive decay into electrical energy. Let's see how they work:

- Stirling Working Principle: SRGs are based on the Stirling thermodynamic cycle, a thermal cycle involving a compressible gas (usually helium) confined within a closed system. This cycle allows the utilization of heat to perform mechanical work, which can then be converted into electrical energy.

- Radioactive Decay: Similar to RTGs, the heat source for SRGs is the radioactive decay of isotopes like plutonium-238. The radioactive isotope provides the necessary heat to initiate the Stirling cycle.

- Stirling Cycle: The Stirling cycle consists of a series of phases in which the compressible gas is heated and then cooled. When the gas is heated, it expands and performs mechanical work on a piston or another mechanical device. When the gas is cooled, it contracts and recovers energy. These phases of mechanical expansion and contraction generate the mechanical work used to generate electrical energy.

- Conversion into Electrical Energy: The mechanical work generated by the Stirling cycle is converted into electrical energy through an electrical generator connected to the piston or mechanical device. This electrical energy is then available to power the instruments or devices onboard the spacecraft or instrument.

- Higher Efficiency: SRGs are known to have higher efficiency compared to RTGs because they utilize a thermodynamic cycle to convert heat into mechanical energy before converting it into electrical energy. This can lead to a more efficient utilization of the heat produced by radioactive decay.

This type of generator is used to provide electrical energy in long-duration space missions, similar to RTGs, but leveraging a more complex and efficient conversion cycle. So they use a dynamic approach. It incorporates a radioisotope and engages in 110 thermodynamic cycles involving the expansion of gas to facilitate mechanical-to-electric conversion. This progression involves transitions from decay to thermal energy, then from thermal energy to mechanical energy, and finally from mechanical energy to electrical energy. Despite these multiple energy conversion steps, the efficiency achieved is higher, rendering it suitable for rover applications. The key distinction lies in the potential for multiple energy conversions, which can more than double the efficiency of thermal energy collection from decay, leading to significant mass savings.

New tecnology

The Next-Generation Radioisotope Thermoelectric Generator (RTG) features a static architecture akin to the GPHS RTGs, with significant advancements in thermal conversion and dissipation management. This design can be tailored to meet specific system requirements, and its efficiency is notably enhanced, potentially reaching up to 15%. The flight-ready unit is projected to be completed by 2028.

Comparisons

Comparing the modules used in both General-Purpose Heat Source (GPHS) and the new technology, it's evident that the mass is reduced in the latter. RTGs and SRGs can be classified into static and dynamic power systems. While RTGs lack moving parts, SRGs incorporate a piston mechanism. The SRG's schematic highlights the pistons that interact with heated gases, utilizing residual thermal energy not fully utilized in thermionic conversion. Despite being more efficient in conversion, it's important to note that SRGs do involve moving parts in their operation. A summary of the differences:

1. Radioisotope Thermoelectric Generators (RTGs): RTGs operate by directly converting the heat produced from the radioactive decay of isotopes like plutonium 238 into electricity using thermoelectric materials. While they have been used successfully in various space missions, their efficiency is relatively lower compared to other power generation methods. RTGs are known for their simplicity and reliability, making them suitable for missions in environments where solar energy is limited, such as deep space exploration. However, their design can be limited by size and weight considerations.

2. Stirling Radioisotope Generators (SRGs): SRGs are a more advanced type of generator that utilize the Stirling thermodynamic cycle to convert heat into mechanical work and then into electricity. This cycle allows for higher efficiency compared to traditional RTGs. SRGs have the potential to provide improved power output and efficiency, making them suitable for missions with greater power demands and longer operational lifetimes. However, SRGs are more complex due to their moving parts and require careful engineering to ensure reliability.

3. Multi Mission RTGs (MM RTGs): MM RTGs are a variant of traditional RTGs designed to be more compact and adaptable. They retain the basic principle of converting radioactive decay heat into electricity using thermoelectric materials. MM-RTGs are developed with versatility in mind, allowing them to be used in various mission types with different size and power constraints. While they may have slightly reduced power output compared to larger RTGs, their adaptability makes them suitable for a wider range of applications.

4. Next Generation RTGs: Next Generation RTGs are an evolution of traditional RTGs, incorporating advancements to improve efficiency and overall performance. These RTGs aim to address some of the limitations of earlier designs by incorporating new materials, technologies, and design considerations. The goal is to achieve higher efficiency and potentially higher power output, making them more suitable for future space missions. As technology continues to advance, Next-Generation RTGs seek to provide reliable power for longer durations while adapting to changing mission requirements.

7.2 Power distribution

Two Distinct Source Control Strategies: The first approach involves discharging excess current in parallel when it's not needed. For instance, if a solar panel or an RTG generates more power than required by the loads, you can close the circuit on a variable resistance. This redirects a portion of the generated power that is surplus or unused. Alternatively, a series-connected strategy can be employed known as peak power tracking. Instead of simply dissipating excess current, this strategy focuses on optimizing power production based on specific mission needs. This is achieved by adjusting the voltage to find the optimal point for power production, effectively matching the source output to the load requirements. A voltage converter is utilized to ensure the source operates efficiently concerning the load.
Battery Management: In illuminated conditions or if the primary power source is intended to be solar panels, batteries function as a load. Consequently, batteries impose constraints on the line's voltage due to charging. However, it's possible to decouple this relationship, maintaining a fixed line voltage that isn't influenced by battery charging voltage. An alternative approach involves utilizing a discharge regulator to maintain voltage control from the perspective of the battery when it transitions to acting as a power source. Voltage regulators cater to different loads while adhering to a consistent bus voltage.
The Power Distribution System Encompasses: Wiring arrangements, safety measures such as fault protection and switch gear that control the activation and deactivation of spacecraft loads, command decoders, enabling specific loads to be commanded.
The Design of the Power Distribution Architecture is Determined by: the spacecraft's scale and complexity and the power demands/requirements.
There are two possible architectures to consider:

- Distributed Architecture: Each load is equipped with its own dedicated feeding and voltage control system.
- Centralized Architecture: Everything is managed from a central bus.

The choice of Bus Voltage depends on the required power: a 28V bus is suitable for small to medium spacecraft with a total power of under 2kW. For larger spacecraft with total power exceeding 2kW, a 100V-150V bus is employed to minimize losses. It's important to note that cabling and harness mass can account for a significant portion (15-25%) of the total Electrical Power System (EPS) mass, which can, in turn, be 5-10% of the entire satellite mass.

7.2.1 The regulation and control system

The regulation and control system is responsible for providing the correct current-voltage (I-V) combination to loads, including batteries. The [I-V] demand can change over the mission's lifetime due to various factors:

- Different onboard subsystem demands.
- Varying mission profiles.
- Degradation of power sources.

Power regulation can be achieved with the primary source and the bus. Three main power topologies available are:

1. Unregulated Bus Voltage Power System: The system works based on load requirements, which can be risky for the electrical components. Loads experience a fluctuating voltage determined by the battery's voltage.

2. Sunlight/Quasi Regulated Bus Voltage Power System: Power is regulated based on solar panel output. This is controlled during battery charging (Sun regulated) and follows the battery's voltage status during eclipse.

3. Regulated Bus Voltage Power System: This approach is the most robust but demands a considerable amount of power. It's controlled through dc/dc converters in both charge and discharge phases, maintaining a voltage within a few percent of the nominal voltage throughout the orbit period (typically within 2-5%).

and energy source can be regulated with 2 different solutions:

1. The energy source can be regulated with Direct Energy Transfer (DET) (The energy source directly feeds the load) and the Peak Power Tracking (PPT) (The power source operates at the point where maximum power can be extracted).

2. the bus:

7.2.2 Unregulated bus

The energy sources are directly linked to the distribution bus. Concerning the system architecture, the primary array provides power directly to the platform and the secondary solar array functions as a current source for battery charging. The distributed voltage (V) is directly influenced by the characteristics of the energy sources.

Unregulated with DET

The logic behind this setup involves determining the operational point based on load requirements, which includes the battery acting as a load itself. This approach maintains a constant voltage level and exercises control over the generated current. Any surplus current is managed by diverting it through a shunt, thereby ensuring a safety margin. In the case of DET (Direct Energy Transfer) regulation, dissipation is involved. A shunt is introduced in parallel to the energy source to handle excess power by dissipating it. Furthermore, certain considerations come into play:

- Thermal control measures are necessary.

- The design prioritizes simplicity and lightweight construction.

- This approach is particularly well-suited for missions of medium to long durations, exceeding 5 years.

- The Xd factor is set at 0.85.

- The role of battery voltage is prominent, and unregulated bus voltages inherently depend on the type and configuration of battery cells connected in series.

Common voltage ranges are as follows:

- For bus power up to 2 kW, the range is 28V +/− 6V.
- For bus power up to 3.5 kW, the range is 35V +/− 7V.
- For bus power up to 5 kW, the range is 42V +/− 8V.

Characteristics of DET

The Direct Energy Transfer (DET) approach has specific characteristics: It is vulnerable to lock-up situations. If the power output of the Solar Array (SA) falls below the required load power, the battery steps in to supply the deficit. This results in a decrease in voltage (V) that can lead to a trapped state. To recover from this scenario, the voltage (V) needs to be adjusted away from the battery voltage (Vb). Recovery can be achieved by setting the battery power (Pbattery) to zero, thereby aligning SA characteristics with the required power demand. As a countermeasure, oversizing solar panels is considered. This not only reduces mass and cost but also minimizes significant voltage fluctuations. However, this solution requires active power conditioning at the load level. It also results in a heavier harness, as larger cables are needed to minimize power dissipation. Devices connected to the system must tolerate bus voltage fluctuations of up to 20%. It's important to note that the DET approach is not suitable for architectures involving more than one battery pack. The architecture is: during periods of direct sunlight, the operating point of the SA is maintained at a fixed voltage (V) through the use of a shunt regulator. During eclipse periods, the bus voltage is directly connected to the battery, eliminating the need for regulation during battery charging.

The energy source of DET

The Energy Source in DET Configuration:

- Power requirement: 280 W during non-eclipse periods, with a voltage (V) of 27V.
- Solar Array (SA) configuration: 18 cells across 20 strings. However, the available SA power falls short of the required 280W, causing the battery to compensate for the deficit and potentially leading to latch−up issues.
- Increasing the number of SA strings in parallel effectively boosts the current level, enabling the provision of 320W during non eclipse periods.

In scenarios where the characteristic curve of the energy source lies above the power demand curve (for instance, the green voltage times the current), there exists a difference referred to as Delta. This Delta signifies the amount of current that is routed through a parallel resistor to deplete power. This configuration can be employed when cells or arrays possess the appropriate power demand, allowing power depletion. However, if the curve corresponds to the blue one, the positive Delta relative to loads or negative Delta relative to solar panels renders this approach unfeasible. Increasing parallel configurations within the cell's topology serves to enhance current availability, which is a matter of design, not operational utilization. It's essential to account for worst-case scenarios, assessing characteristic curves concerning topology to ascertain the viability of such solutions. It's worth noting that the DET approach avoids power consumption but introduces a hotspot. This technique is suitable for scenarios characterized by relatively stable power demand, consistent eclipse frequency, or the need for a simplified system design.

Unregulated with PPT

Active Control in PPT Regulation:

- PPT (Peak Power Tracking) regulation involves an active control mechanism.
- A direct current to direct current (dc/dc) converter is incorporated in series with the Solar Array (SA) to synchronize with the power demand of the bus.
- This configuration absorbs approximately 4-7% of the Photovoltaic Solar Array's (PSA) power output.
- Although it offers advantages, such as adaptability and increased efficiency, it is somewhat heavier and less efficient when compared to the DET approach.
- PPT regulation is particularly suitable for missions of shorter duration and those characterized by a high number of cycles.
- Efficiency degradation factor (Xd) is typically around 0.8.

Operational Logic: the regulation action primarily targets the voltage aspect. Unlike DET, the bus voltage remains independent of the SA voltage.

The energy source of PPT

In the context of PPT regulation, the 18-series-20-parallel configuration effectively meets the power demand upon exiting an eclipse phase. An alternative solution to the direct energy control discharge is the implementation of PPT. Unlike the direct control method, PPT operates in series, affecting the voltage while leaving the current supplied to the loads unchanged. It adheres to the voltage-current points on the characteristic curve, aligning them precisely with the load's requirements at that specific operational moment. PPT proves to be highly effective, as it dynamically adjusts both voltage and current, but it comes with a power consumption requirement. Evaluating Xd, which represents energy demand normalization, it becomes evident that PPT is comparatively less efficient or more power-demanding, translating to a trade-off in efficiency.

7.2.3 SA regulation comparison

In a scenario where you are dealing with a specific cell or current generator, and you have the power requirement curve as depicted, two distinct approaches come into play. If you opt for Direct Energy Transfer (DET), the system operates at a fixed voltage level. This voltage remains constant whether the battery is connected (taking on the battery voltage) or decoupled (matching the load voltage, which is also constant). In this setup, the current is discharged in parallel to meet the power demand. However, with Peak Power Tracking (PPT), a different methodology is employed. The PPT method involves determining the operating point at the intersection of the curve with a cursor that moves along the voltage axis, allowing for voltage adjustments. This approach accommodates the changing voltage in order to optimize the power production, ensuring alignment with the load's specific power requirements.

7.2.4 Fully regulated bus

During the sunlit phase: In case of a bus voltage increase, a signal prompts the shunt to adjust the operating point to maintain the voltage. If the bus voltage decreases, a signal triggers the boost to draw more current from the battery, stabilizing the voltage. It Offers great flexibility and it is free from lock-up issues. It ensures no transient effects on the bus. Battery design is unaffected by power bus voltage variations. It is suitable for managing multiple battery packs. (BCR = Battery Charge Regulator; BDR = Battery Discharge Regulator) There is 3–13% power loss in BDR and a 3–10% power loss in BCR It's slightly heavier and relevant for:

- Larger or medium-sized satellites
- High voltages (50, 100, 120V – not exceeding 125V)
- Situations with rapidly changing power demands and numerous eclipse cycles

7.2.5 Semi regulated bus

Quasi regulation involves controlling the bus voltage during battery charging (Sun-regulated), and it tracks the battery voltage during eclipse periods. In terms of power conditioning strategies, there's the semi-regulated bus approach.
Advantages:

- Establishment of a main bus with BR-type stabilization, connected to a solar generator operating optimally.
- The operating point and usable power of the solar array are no longer determined solely by the battery voltage, eliminating the risk of power lockup issues.
- Reduction in the weight and cost of the EPS due to the absence of additional BDR devices.

Disadvantages: Users requiring a stable supply voltage from the bus, but unable to operate during certain periods, face limitations.
Applications: This approach is well-suited for geostationary spacecraft with low power needs during eclipse operations.

Chapter 8

Thermal Subsystem

There are three thermal mechanisms: conduction, convection, and radiation, which are essential for thermal assessment. These mechanisms involve material parameters respectively like conductivity, convective heat transfer coefficient, and thermo-optical fluid properties that influence heat transfer (reflectance, absorptance, transmittance, emittance). The chart emphasizes parameters like cross-section, surface area, view factor, and distance, which play a crucial role in determining temperature dependencies and coefficients in the equations. In space applications, convection is less relevant as thermal control primarily relies on radiation, which considers optical properties such as reflection or absorption, area, and view factor. On the other hand, conduction requires attention to conductivity and the surface and length of the conductive material. Regarding the link between conduction and radiation, you can connect a highly conductive material, like a battery, to the external surface and direct it toward space. This arrangement enhances the dissipation of thermal power from the payloads to the radiator. The connection can be either mechanical or electrical.

8.1 Radiation

The behavior of the system you are dealing with, whether it's a subsystem or the entire satellite (initially represented as a sphere), is mainly characterized by two crucial aspects: absorptivity (α) and emissivity (ϵ).

- When dealing with sources in the high-frequency domain, high absorptivity is essential. For instance, if you need to raise the system's temperature, selecting materials with high absorptivity is the most effective approach.

- Conversely, for low-frequency radiation (infrared domain), high emissivity of thermal energy is significant. If you intend to lower the temperature, choosing materials with high emissivity will be beneficial.

- In some cases, you may choose materials with both high emissivity and high absorptivity, but this usually involves using multilayer structures or materials with multiple optical properties. It's not possible to find opposite properties in a single physical material.

It's important to consider these coefficients in relation to the incoming radiation. Remember that the sum of all coefficients must be 1 since each represents a percentage of the total spectrum. Regarding the source while being in flight or in orbit: the maximum energy emitted or received from the Sun remains at high frequencies, and therefore, it is managed through absorptivity. On the other hand, the reflected thermal energy emitted by any planet will be controlled based on emissivity related to the flux.

8.1.1 Optical properties

Absorptivity and emissivity are the primary control variables to manipulate, along with surface properties. The diagram above illustrates the optical properties' evolution concerning the incidence angle with respect to the surface, specifically focusing on metals and non-metals. For metals, the behavior is favorable at higher angles of incidence, while the opposite holds true for non-metals. The graph on the right displays wavelength on the horizontal axis and the nominal emittance on the vertical axis. Higher values on the vertical axis indicate a greater capability to reject internally produced power. This helps identify surfaces suitable for your project. For instance, polished aluminum is an ideal emitter with good emissivity (0.7 or higher), making it excellent for power rejection. At very low wavelengths (high frequency), materials like ideal polished aluminum serve as great emitters, similar to white bands. Conversely, black paints are insensitive to wavelength, with a constant absorptivity of one, whereas ideal reflectors have a constant absorptivity of zero.
The graph on the right displays emittance concerning absorbance. The rectangles on the graph represent domains for materials. When aiming for high absorptivity (to keep the satellite warm), emissivity tends to be poor, meaning energy is captured at high frequencies but not released in the thermal domain. On the other hand, emitting in the low frequency makes the system less efficient in obtaining energy from outside. For cooling and non-absorbing high frequencies, the classical white painting is effective. Black paints are unique in having identical absorptivity and emissivity, capturing high frequencies and allowing low frequencies to exit.
To model the system, consider a mono-node approach, treating it as a single element (e.g., a sphere representing the spacecraft) without differentiating materials.

View factor

The View Factor, represented by a geometric fraction, signifies the portion of diffuse energy leaving surface I and directly intercepted by surface J. It purely accounts for the geometry of the system. The subscripts "i" and "j" refer to the two elements involved in the exchange. For radiative exchange factors, two essential aspects come into play: geometry, which includes the fraction of rays captured, and surface properties, incorporating the emissivity of the surfaces. For simplified surfaces, such as a sphere in a sphere, a plane in front of another plane, or Monte-Carlo analysis, reference analytic numbers can be assumed. The widely accepted code by ESA is ESATAN, which calculates all view factors for discretized surfaces based on the given geometry and orbit. For instance, the view factor between Earth and the satellite is crucial for evaluating radiative heat exchange, while the view factor between the illuminated fraction of Earth and the satellite is essential for assessing Albedo heat. The computation of view factors can be challenging, especially analytically, which can be long and sometimes even impossible. Hence, a statistical approach, such as Monte Carlo (MC) simulation, is often employed. In this method, a large number of rays are shot in random directions, and statistical analysis is performed to determine the percentage of rays striking other surfaces.

The Plate to Earth View Factor is determined by the offset angle, lambda, which represents the angle between the center of the planet and the normal to the flat plate. Lambda is parameterized with respect to the altitude. This angle plays a crucial role in calculating the view factor between the flat plate and the Earth's surface, as it accounts for the positioning and orientation of the plate relative to the planet's center and the direction of the normal vector of the plate.

8.2 Heat sources

In the environment, when conducting missions, we need to consider the different energy sources outlined in this slide. The relevance of these sources depends on the phase of the mission and whether the vehicle is on a surface or in deep space.

1. One of the primary sources is the sun radiation, which falls under the high-frequency band. To protect the vehicle from this radiation, we can manipulate the absorptivity coefficient.

2. Next, there's the Albedo, which refers to the energy in the high-frequency band that is reflected by the Earth. The absorption of this energy depends on the spacecraft's surface properties. The coefficient we need to consider for Albedo still falls in the high-frequency band since the solar flux remitted by the Earth is less than 1, and the radiation is absorbed by the spacecraft using a coefficient within that band.

3. On the other hand, there are other sources that emit low-frequency (high wavelength) radiation, typically corresponding to the thermal frequencies of hotspots, such as planetary sources. This radiation impinges on the spacecraft's surface and is received through the emissivity coefficient, which is determined by the surface's optical properties.

4. Additionally, there is always internal dissipation occurring as long as the vehicle is not dormant. The contribution from each of these sources may vary and depends on what systems are activated, thus being related to the power budget breakdown specific to that phase of the mission.

8.2.1 Solar flux

The solar flux source follows an inverse square law with distance from the Sun. This means that the solar flux intensity depends on the location in space, and the reference temperature is defined at a specific distance where the planet is assumed to behave as a black body across the entire frequency spectrum, with emissivity ϵ equal to 1. The solar flux is typically represented as 13675 W (watts) for the reference sphere with a=c=1, which serves as a rough indicator of the local thermal environment.

8.2.2 Albedo

Albedo refers to the fraction of radiation coming from the Sun that is reflected by the planet's surface, and it operates in the high-frequency band. It is related to the absorptivity coefficient. Albedo is not constant and depends on various factors, including the seasons, regions traversed (illuminated side or shadow side), and altitude. Additionally,

Albedo is connected to the view factor, which considers the orientation of the surface. For practical purposes, it is often assumed that the planet radiates uniformly from its entire cross-sectional area. So the incoming Sun radiation at orbit varies with R^2_{orbit}.
The formula to calculate the Albedo flux at a specific orbital height (Rorbit) is given by:

$$q_{albedo} = q_{sun} acos(\theta)(\frac{R_{pi}}{R_{orbit}}^2) \tag{8.1}$$

Alternatively, you can explicitly include the view factors (F_p) with respect to the planet's surface orientation, including the terminator region, and the angle of incidence (theta) of solar radiation. In this case, the formula becomes:

$$q_{albedo} = q_{sun} acos(\theta) F_p \tag{8.2}$$

To compute the Albedo with the formulation explained earlier, you may also incorporate the inclination of the plate concerning the nadir point and the region with respect to the impinging solar radiation angle (theta). The coefficient indicating the percentage of reflected flux shall be multiplied accordingly.

8.2.3 IR

The IR emission is associated with the normal to the Albedo flat plate in the outer planetary region. The solar radiation intensity at different wavelengths is represented in the table. The planet's temperature can be computed considering its angular velocity or predetermined based on its characteristics, depending on whether it is a fast or slow rotating planet. When the planet is flying in shadow, the emitting power is solely due to the contribution of the IR radiation. The planet's temperature (T_{planet}) can be calculated using the first equation for fast rotating planets, and the second for slow rotating ones:

$$T_{planet} = (\frac{(1-\alpha)Q_{sol}}{4\sigma\epsilon})^{\frac{1}{4}} \tag{8.3}$$

$$T_{planet} = (\frac{(1-\alpha)Q_{sol}}{\sigma\epsilon})^{\frac{1}{4}} \tag{8.4}$$

For practical purposes, it is often assumed that the planet radiates uniformly from its entire cross-sectional area. The incoming IR emission at orbit varies with $1/R^2$, where R is the distance from the planet to the Sun. This can be evaluated using the formula:

$$q_{IR} = \sigma\epsilon T^4_{planet} F_{planet} \tag{8.5}$$

8.2.4 BETA angle

The beta angle serves as a crucial parameter to evaluate various aspects before delving deep into the analysis.

$$\beta = \arcsin cos(\delta_s) sin(RI) sin(\Omega - \Omega_s + sin(\delta_s) cos(RI)) \tag{8.6}$$

The beta angle represents the angle between your orbit and the solar vector. It has to be comprised between -90 and 90 degrees and it provides a comprehensive understanding of the entire orbit around the Sun. By evaluating the beta angle with respect to inclination (ϵ), declination (δ), and angular position ($\Omega - \Omega_s$), you gain valuable insights into your specific conditions and potential variations.

8.2.5 Eclipses

Eclipses play a significant role in altering the thermal situation of the system. The duration of an eclipse can be computed based on your preferred method. If you already have the appropriate code to consider the correct evolution of your ellipse, you can employ it to intercept the shadow cylinder of the planet. This helps you determine when to consider the IR contribution solely from the planet and when to account for the albedo and IR effects.

8.3 Heat power dissipation

All power produced in the system eventually becomes thermal energy. Therefore, during the initial modeling phase, the dissipation power should account for the entire produced power. There is a rare situation where you can exclude the emitted power from the power reported in the link budget. This occurs when the emitted power is dissipated in space, making it irrelevant for your considerations. Each component in the system has an efficiency factor, such as solar arrays, where the obtained power is multiplied by the efficiency to determine the power used in the system. The crucial task is to identify the WORST CASE scenario, considering the most incoming flux and the least flux that must be summed. However, all power dissipated on board must be radiated out to space, except for specific cases like:

1. Transmitted power (radiofrequency)
2. Electric thrust (ion thrusters)
3. Transient events for rotating motors (in steady state, torque is dissipated by friction)
4. Solar arrays (wings)

Regarding the temperature constraints for on-board equipment, the top right table provides the allowable temperature ranges for each component. These temperature ranges must be adhered to for proper operation. The battery and camera equipment, particularly the electronic units and CCD camera, often present the most restrictive thermal conditions that need careful consideration.

8.4 Thermal analysis and design

The thermal analysis and design process involves several steps:

1. Select Worst Conditions: Determine the minimum (Tmin) and maximum (Tmax) temperatures to consider.
2. Computation: Perform thermal analysis for both cold and hot conditions, considering external and internal heat sources. Identify the cold case and hot case based on mission profile, orbit, operational modes, payload, and subsystem efficiency.
3. Thermal Control Strategy: Choose between active and passive thermal control strategies.

4. Select Thermal Control Components: Decide on the materials, coatings, paints, and radiators for the thermal control system.

5. Thermal Analysis: Perform thermal analysis using preliminary single-node or more detailed multiple-node numerical solutions (e.g., finite elements or finite difference methods). Adjust emissivity and absorptivity accordingly.

6. Size and Location of Radiators: Determine the size and placement of radiators for efficient heat dissipation.

7. Size and Location of Heat Pipes or Phase Change Devices: Decide on the sizing and architecture of heat pipes or phase change devices for effective thermal control.

8. Location and Sizing of Active Control Components: Identify the location and size of pumps and pipes for active thermal control.

9. Testing: Conduct tests to validate the thermal analysis and design.

Steady state

The analysis can begin with a preliminary single−node approach, considering steady−state conditions. The spacecraft can be modeled as a point mass. The total input heat plus the total internal heat plus the total output heat must balance to zero. In general, you have two scalar equations to solve, considering optical properties, surface, and temperature (radiative balance). The thermal control strategy can be determined by keeping the passive approach, neglecting conduction modes at this level. You need to identify the active power required to fill the gap between absorbed and emitted power. This analysis starts with increasing the area for heat emission or absorption and then addressing other factors. For steady−state conditions, the simplified equation to consider is:

$$Q_{internal} - Q_{output} + Q_{input} = 0 \tag{8.7}$$

In general we can consider:

- Hot Case:
$$Q_i + Q_{IR} + Q_a + Q_s - Q_d = 0 \tag{8.8}$$

- Cold case:
$$Q_i + Q_{IR} - Q_d = 0 \tag{8.9}$$

These equations statically balance the total input and output heat. In the cold case, there are no external heat sources to consider. Remember that these are representative equations; you should adjust them to fit your specific case, considering any necessary simplifications or neglecting certain factors.

Single node flat sphere

1. Radiation (emitted)
$$Q_e = A_{surf}\sigma\epsilon_{s/c}(T^4_{s/c} - T^4_{space}) \tag{8.10}$$

2. Solar absorbed (directed)
$$Q_s = q_{sun}A_{cross}\alpha_{s/c} \tag{8.11}$$

3. Infrared absorbed
$$Q_{IR} = q_{IR} A \epsilon_{s/c} F_{pl} \tag{8.12}$$
4. Albedo absorbed
$$Q_A = q_{sun} a A F_{pl} \alpha_{s/c} K_a \tag{8.13}$$
5. Internally generated: More in general, the Qi heat power generated internally must be computed for each internal component taking into account its efficiency which gives the size of the not used power which is thermally dissipated

Single node plate

1. Radiation (emitted)
$$Q_e = A F_{s-sp} \sigma \epsilon_{s/c} (T^4_{s/c} - T^4_{space}) \tag{8.14}$$
2. Solar absorbed (directed)
$$Q_s = q_{sun} A \alpha_{s/c} cos(\theta) \tag{8.15}$$
3. Infrared absorbed
$$Q_{IR} = q_{IR} A \epsilon_{s/c} F_{pl-surf} \tag{8.16}$$
4. Albedo absorbed
$$Q_A = q_{sun} a A F_{pl-surf} \alpha_{s/c} K_a \tag{8.17}$$
5. Internally generated: A single flat plate, representative of: a radiator $Q_i = 0$ and a solar panel so part of the Sun impinging power is converted into electrical power: $P_{elect} = \eta Q_{sun}$ to be removed from Sun power entry More in general, the Qi heat power generated internally must be computed for each internal component taking into account its efficiency which gives the size of the not used power which is thermally dissipated

In the emitted contribution, the output temperature is typically assumed as a variable, allowing you to manipulate the emissivity and the area dedicated to emitting low-frequency band energy. For the solar and infrared contributions, you have the flexibility to adjust the absorptivity (optical properties) in the high-frequency band. Keep in mind that the surfaces considered must have consistent visibility (view factor). These simple equations express the energy emitted, absorbed from the Sun, absorbed from other objects with a temperature above 0K, and absorbed when looking at an illuminated surface. By manipulating these variables, you can achieve passive control and fine-tune the results to ensure your static temperature falls within the identified range. If, after computation, the temperature values remain between 0°C and 25°C, the design is suitable for both hot and cold cases, and the goal is achieved. Suppose your satellite is white-painted and flying above the Earth, resulting in a temperature of 15°C. In that case, you can fix the values of alpha, epsilon, and the area and incorporate them into the computation. If the temperature falls within the required range (above 0°C), it is acceptable for the cold case, and you can consider the problem solved. However, if the temperature deviates from the desired range, say reaching -10°C, you need to loop in two different ways:

- Start from the cold case and adjust emissivity, absorptivity, and area, then cross-check for the hot situation.

- Start with the required area and decide to limit the exposed area to reduce thermal energy emission. Alternatively, increase the power contribution from internal components in the computation (e.g., switching on previously turned-off components). This helps maintain passive control and keeps the temperature within the desired range.

If these adjustments still do not achieve the desired temperature range, you can relax the expectations for the hot and cold cases, considering, for example, 30°C or 40°C for the hot and −10°C for the cold. Regarding the "internally generated" power contribution, it pertains specifically to the solar arrays. If you have a wing, you need to account for the mechanical and thermal connections with the main body. By decoupling the wing from the body, you create an independent node, allowing the wing to control its temperature separately. In this case, the energy to be dissipated is solely attributed to the wing. Alternatively, if you couple the wing with the body, there is an internal flux to consider, generated by the satellite itself. When modeling in a single node, this contribution is already considered within the sphere used to model the satellite.

8.4.1 Steps for sizing

1. Definition of Temperature Limits
2. Evaluation of Power Dissipation
3. Represent Satellite Area as an Equivalent Sphere.
4. Selection of Worst Case Scenarios:
   - Hot Case: Consider albedo, solar, IR, and maximum internal heat.
   - Cold Case: Focus on IR and minimum internal heat.
5. Computation of Hot Case Temperature for Optical Properties and Area Selection:
   a) Determine the maximum allowable temperature (Tmax) (1) and analyze the hot case scenario (4). Compute the radiator area, considering the limitations imposed by the equivalent sphere in a single-node analysis.
   b) Check if the area complies with mechanical constraints:
   - If yes, proceed to assess power in the cold case.
   - If no, consider: Optimize radiators' coating, evaluate the required $delta_{load}$ for the next refined phase and relax thermal boundary constraints, if possible.
6. Verification of the Cold Case: Ensure that the temperature requirements are met with the selected optical properties and involved area. If yes, proceed to more detailed modeling with multiple nodes. If not, proceed to step 7.
7. Heaters Sizing:
   A) Determine the minimum temperature based on the area (consider heaters or heat switches if it is too cold).
   B) Communicate the heaters' power to the EPS (Electrical Power System) engineers.
8. Identify Acceptable Thermal Limits for Both Cases: Begin addressing components that need separate treatment in the multi-node modeling.

8.4.2 Margin

Concerning the margin, there is a margin philosophy document available on BeeP. At your current level of analysis, you must account for a significant margin, which involves the constraints of each component onboard. Since you are in the conceptual phase and modeling with a single node, you are not leveraging the opportunity to articulate the main body and its components fully. The typical uncertainties for temperature measurements in different phases of ESA ECSS−E−10−03A (Space Engineering: testing) are as follows:

- Phase A: Typical uncertainty of 15 K.
- Phase B: Typical uncertainty of 10 K.
- Phase C/D:
  1. Typical uncertainty of 8 K before thermal balance tests.
  2. Typical uncertainty of less than or equal to 5 K after thermal balance test and Thermal Mathematical Model (TMM) correlation.

8.5 Thermal control components

When it comes to thermal control in space, passive control is often preferred due to its simplicity, reliability, and cost-effectiveness. On the other hand, active thermal control (ATC) is employed in situations where passive methods are inadequate. ATC allows for precise temperature regulation and is suitable for environments with variable conditions, cryogenic applications, and configurations requiring adjustments in attitude, materials, etc. In the context of radiation, manipulating optical properties becomes crucial. The parameters of interest are alpha (absorptivity) and gamma (emissivity), as they significantly impact the overall heat power during single-node analysis. The table illustrates the classification of thermal control solutions into passive and active categories, considering two primary heat mechanisms: radiation and conduction. Radiation control involves working with materials, such as coatings (paints, metal finishes), to influence thermal properties. Multilayer insulation (MLI) blankets are employed as an intermediary step before resorting to active control. MLI blankets facilitate thermal decoupling from the environment, creating an adiabatic scenario where internal conditions are solely managed by the system engineer. These blankets can be placed around specific components like batteries or payloads, or even the entire spacecraft. To achieve active thermal control, various techniques are utilized, including phase change materials, heat pipes, thermoelectric control, fixed or variable conductance devices, electronic control, and fluid covers. The decision to employ active control is made after careful analysis and, in many cases, involves multi-node analysis rather than relying solely on single-node simulations. Overall, radiation control plays a pivotal role in managing thermal issues in space, and the selection of suitable materials and techniques greatly influences the spacecraft's performance and longevity.

Radiators serve a specific function in thermal control, differentiating them from coatings. For instance, the external surface of a spacecraft primarily functions to provide structural support against launcher loads and protect from the external environment. While thermal control is not its primary purpose, we utilize radiators to manage heat. For instance, a wing deployed as a solar panel, positioned 90° away from the solar panel facing deep space, can act as a radiator. In hot situations, this surface can also serve as an entry point for heat, contributing to attitude control. The radiator's surface can be adjusted to achieve high emissivity when heat dissipation is needed, or low emissivity when heat retention is necessary, achieved through the use of louvers. Radiators are designed to dissipate the specific heat generated by the entire satellite into space. They can be compared to "le tende veneziane" (Venetian blinds) where fins or slats are used to cover an emitting surface, exposing a receiving surface. This passive control approach utilizes smart materials and may include rotational mechanisms to open and close surfaces. The use of thermal sensitive materials plays a fundamental role in passive control, and no additional power is required. However, it is essential to consider radiator placement when not in use, as they can have unintended effects on the spacecraft's thermal environment. Careful consideration and design are necessary to ensure optimal thermal performance throughout the satellite's lifecycle.

The phase change material acts as a storage system, capable of either dissipating or retaining heat depending on the thermal requirements. It behaves like a basin, utilizing latent heat to change its state and facilitate heat removal from components or maintain their temperature, all while operating within a passive control framework. In contrast, active control systems come in two types. Heaters, functioning as resistors, are small and flexible, allowing them to be strategically placed as needed. For instance, in non-planar surfaces like spherical or cylindrical tanks, these adhesive heaters can be distributed for effective temperature control. Another active control method employs the Peltier element, acting as a power pump to transfer heat from the hot to the cold part, requiring electrical input. This control mechanism is suitable for specific instruments or detectors due to its low power demand, although it offers only a slight temperature difference. During the design phase, it is crucial to consider the multifunctionality of connections. Any physical connection serves multiple purposes, encompassing thermal, structural, and electrical aspects. Designers must carefully assess and verify these connections to ensure they meet all requirements. For example, even the length of an element can impact its conductivity and overall performance.

In certain cases, it is necessary to minimize conductivity. However, achieving low conductivity can pose challenges in specific scenarios. For instance, if a long beam is used as a structural element, it may compromise stiffness, which is undesirable. Similarly, when structural support requires a large contact area, it can hinder thermal conduction. One approach to address this is to introduce vacuum space between surfaces and create high roughness to limit real contact surface area, as vacuum has low conductivity. Surface finishing also plays a role in controlling conductivity.

Straps are commonly employed to establish thermal paths. These small metal plates are screwed from the hot side to the radiator side, facilitating thermal conduction. For instance, a thermal path can be created from batteries to an external radiator panel.

Heat pipes are another type of control mechanism that falls within the active domain, although they exhibit some characteristics of passive control as well. Heat pipes utilize capillary action, such as those found in personal computers, where fluid moves from the hot to the cold part through phase change. Heat pipes can be considered either passive or active, depending on various factors and perspectives. Combining different types of control is also feasible. For instance, incorporating a compartment of heat pipes near batteries allows heat to travel along the spacecraft through phase change of the fluid until it reaches the radiator. The key message is to minimize reliance on active control types and primarily focus on passive solutions while integrating them in a flexible combination based on specific requirements.

8.5.1 Passive components

Coating

Coating plays a crucial role in thermal control for space applications. The graph on the top left of the slide illustrates the energy entering the system based on wavelength or frequency. At low frequencies, the energy corresponds to high values, representing non-thermal energy like UV or high-energy solar radiation. On the y-axis is the absorbance coefficient (alpha), indicating how much energy is absorbed. As the wavelength increases, the system moves towards thermal emission in the infrared (IR) domain. Effective cooling requires a large emissive coefficient at long wavelengths while warming demands a low emissive coefficient. The slide showcases different surface combinations. For instance, white-painted houses have low absorptivity and high emittance, making them cool since they are mainly found in sunny regions. Greek houses and houses in tropical or equatorial regions are also painted white for the same reason. The bottom of the slide presents graphs depicting the energy distribution in various domains (e.g., solar, albedo) and the corresponding absorbance behavior. The left side represents absorbers, while the right side shows reflectors, both functioning oppositely. Ideal absorbers have high absorbance, close to one, while ideal emitters have an absorbance of zero. Their sum always equals one, but it's essential to consider the wavelength or frequency. The slide also presents useful graphs for preliminary design, as they provide practical engineering insights. Alpha and epsilon (absorbance and emissivity) cannot be assumed as independent variables because they are properties of the same material at different wavelengths. Therefore, for a specific material, if a particular value of alpha is desired, epsilon is also determined accordingly. In the coated sphere equilibrium case, various elements used to fold it are listed. The graph indicates the temperature ratio between absorption and emission of the sources from the highest to the lowest temperature. This aids in making rough selections before detailed computations. When designing a spacecraft, it is essential to consider worst-case scenarios for both cold and hot conditions since the spacecraft's properties cannot be altered during the mission. This explains why ISO (International Space Station) is white-painted and why instruments are mounted with cooling systems, like liquid helium, to maintain optimal temperatures. Beta cloth, white paint, silver Teflon, and anodized aluminum are some of the materials used in spacecraft construction. These materials exhibit different behaviors and optical properties depending on the spacecraft's location and environmental conditions. Datasheets provide information on the end-of-life and aging per year due to the environment. It is crucial not to assume the beginning-of-life values for sizing when materials are listed as out of fabric in the datasheet.

Coating degradation effect

This graph depicts the effects of degradation, with aging shown on the horizontal axis and properties (such as absorbance or emittance) on the vertical axis. Several degradation sources should be considered, including UV radiation, charged particles, atomic oxygen, contamination, micrometeoroids & debris particles, and corrosion (at the launch site, for instance). Depending on the environment, the impact of degradation can vary significantly, leading to different operational decisions and mission lifetimes. For example, white paint is mostly affected by UV and charged particles, while thermal emissivity is largely affected by atomic oxygen. Let's consider a specific scenario: in the beginning phase of a spacecraft's mission, the payload can be operated simultaneously without issues, and there is good thermal emission (emittance) due to white paint behavior at 0.8. However, over time, the white paint's behavior changes to 0.5, resulting in reduced power production in the spacecraft. Consequently, operational behaviors for payloads may need to be adjusted, shifting from parallel operation to series operation to manage the thermal load effectively.

SSM/OSR

Secondary Surface Mirrors or Optical Surface Reflectors (SSM, OSR) are passive thermal control components used in spacecraft. These components consist of specific materials carefully chosen to achieve the desired thermal control properties. In this example, the materials used for the SSM/OSR include Polyimide, Teflon, and glass on the upper surface, and Silver, Gold, and Aluminum on the lower surface. The primary goal of using SSM/OSR is to limit the amount of incoming thermal flux while maximizing the emission of heat. This approach helps in achieving the desired thermal balance within the spacecraft. The solution involves a two-layered approach:

- Lower Surface: This layer acts as a high reflector and a low emitter. It reflects incoming thermal energy to limit its penetration into the spacecraft.
- Upper Surface: This layer is designed to have high transmission properties, allowing thermal energy to pass through. Additionally, it acts as a high emitter, effectively releasing the excess heat.

The upper surface is made very thin to ensure that it remains nearly isothermal with the lower surface. This helps in maintaining a consistent and balanced thermal environment within the spacecraft. By carefully selecting the appropriate materials and configuring the SSM/OSR in this manner, spacecraft designers can achieve effective passive thermal control, ensuring that the internal temperature remains within the desired range throughout the mission.

Lowers

Thermally activated shutters play a crucial role in regulating the thermal environment of structural and electronic equipment during spaceflight. These shutter systems, also known as louver assemblies, are designed to sense the temperature of a baseplate or space radiator and react accordingly to control the temperature. The louver assemblies consist of meticulously crafted aluminum blades, which are set within a frame and driven by bi-metallic sensors. These bi-metallic sensors, or actuators, are responsible for initiating the movement of the blades in response to temperature changes. When the temperature rises, the bi-metallic sensor contracts and applies torque to rotate the blades towards an open position. This action allows excess heat to dissipate, preventing the equipment from overheating. Conversely, as the temperature decreases, the actuator expands, causing the blades to rotate towards a closed position. In this configuration, the highly polished blade surfaces act as reflectors, redirecting and retaining heat from the baseplate or radiator. Overall, these thermally activated shutters provide a dynamic and efficient means of maintaining the optimal thermal conditions for structural and electronic components throughout the spaceflight journey. Their ability to adjust and respond to temperature variations ensures the equipment remains protected and operational under varying thermal conditions in space.

MLI

Multi-layer insulation, also known as thermal blanket, consists of a surface at the top and another at the bottom, with several layers in between. The purpose of this insulation is to block and reflect thermal energy exiting the spacecraft and the surrounding environment, ensuring the lowest emittance of each surface on both sides. To achieve this, the layers must be reflective and not in direct contact, allowing radiation between them. Dacron elements are used to keep the layers detached from each other. The wrinkled appearance of the layers is intentional, as it allows for detachment and the creation of a vacuum between them to promote radiation. It is crucial to avoid contact between layers, as conduction between them would compromise the insulation's effectiveness. Therefore, a single continuous piece of blanket is preferred over separate pieces that would require joining. Additionally, when the spacecraft is launched from a planet with an atmosphere, it must be vented upon entering the vacuum of space to prevent explosions. To achieve this, the blanket is equipped with small valves strategically placed to allow air to escape during the launch phase. These valves are designed carefully to maintain thermal decoupling and not compromise thermal protection. In the conceptual modeling of MLI, multi-node analysis considers the different layers, internal and external temperatures, and various connections between the layers representing the view factor, which includes area and emissivity for parallel surfaces. Resistance sizing is achieved through an equivalent factor that incorporates emissivity, and datasheets provide epsilon star values related to the number of layers included in the sizing.

Graphs provide valuable insights into how to optimize the contact between layers and increase the surface area for better thermal balance. Several strategies can be employed during the implementation and mounting of the thermal blanket:

- Crinkled and Embossed Layers: Utilizing crinkled and embossed layers helps increase the effective surface area and limit direct contact between layers, promoting better thermal insulation.

- Netting: Incorporating netting in the thermal blanket design further enhances thermal separation between layers and improves overall performance.

- Joints: Carefully designed joints play a crucial role in maintaining the integrity of the thermal insulation and preventing thermal leaks.

- Venting: Proper venting with small valves is essential during launch to allow air to escape without compromising thermal protection.

The graphs illustrate the effects of these design choices. On the left, effective emissivity and power density losses are shown as a function of the distance from the point of joining the blanket pieces. A significant change occurs due to the transition from an almost adiabatic condition at the beginning to a critical point at the end of the cover. On the right, the graph demonstrates how the number of layers affects emissivity. Having around 10 to 20 layers proves suitable for achieving the desired emissivity. Realistically, an emissivity value of 0.003 is considered instead of 0. The graph also parametrizes discontinuity, highlighting that larger blanket pieces lead to better performance. Furthermore, it is important to note that the flux on the right side of the equation emits nothing, making it a potential viable solution for desperate situations. This approach ensures minimal heat leakage and maximum thermal control when needed most.

Radiators

Radiators play a crucial role in spacecraft thermal control, as they are responsible for dissipating waste heat to space. There are various types of radiators, including SC structural panels, flat-plate radiators, and deployable radiators. They achieve heat rejection primarily through radiation, which strongly depends on the radiator's temperature. To size a radiator, factors such as the area (A) and emissivity (epsilon) must be fixed, along with the desired operating temperature, which typically remains around 20°C. SC radiators commonly reject waste heat in the range of 100 to 350W per square meter, with masses varying from negligible to approximately 12kg per square meter. Before considering radiators as appendages, it is essential to analyze the required surface area and evaluate if the radiator can be accommodated on the spacecraft's surface. Electrical grounding issues should also be taken into account, ensuring that only the parts housing instruments need to be protected by the radiator. Additionally, Venice curtains can be strategically placed, opening during spacecraft operation and closing when not needed, providing flexibility in thermal control. Appendages, like solar panels, also present challenges concerning mechanisms, flexibility, and pointing. The system engineer must make decisions regarding the appropriate values for the radiator's area (A) and the energy the spacecraft needs to dissipate, which determines the emissivity required. These considerations ensure effective thermal management throughout the spacecraft's mission.

Phase change materials

Phase change materials (PCMs) are substances that undergo a phase transition, utilizing latent heat, between solid and liquid or between liquid and gas states. This property allows them to store and release heat during the phase change process. PCMs are commonly used in cyclically operating components where maintaining a nearly constant temperature is essential. In such components, when the component is ON, heat is stored in the PCM as it undergoes fusion (from solid to liquid). Conversely, when the component is OFF, the heat is released to deep space, causing the PCM to re-solidify (freeze) and maintain a stable temperature during operation. One typical application of PCMs is in satellites orbiting the Earth (LEO) or landing vehicles on planets without an atmosphere. These environments experience cycling thermal variations, with periods of sunlight (eclipse) and darkness. PCMs are used to dampen these temperature variations. During sunlight, the PCM absorbs and stores heat, and during eclipse, it releases the stored heat, helping to regulate the component's temperature. It's important to note that PCMs are not suitable for all applications and have limitations. For example, once a PCM undergoes a phase change from liquid to gas, it cannot revert to its liquid state through condensation. Therefore, the energy exchange between a volume of helium and the spacecraft is crucial to maintain a specific temperature, and the mission's success depends on this energy transfer. PCMs are specific technologies primarily used for payloads and may not be relevant for cooling down the entire spacecraft, which requires different thermal control methods. Therefore, the use of PCMs depends on the specific thermal requirements and operational conditions of the mission.

Filters

Passive control of conduction can be achieved by manipulating the roughness of the surfaces in contact. When two surfaces are not perfectly polished and have some roughness, there would be air between them on the ground side and vacuum in space, or another medium can be introduced depending on the specific requirements. The choice of material between the surfaces becomes a design parameter that influences conductivity. By selecting a suitable material to be placed between the surfaces before screwing them together, the contact surface area can be increased, enhancing thermal conduction. It is crucial to choose thermally conductive materials that align with the system's requirements. For instance, consider the scenario of an electrical box in contact with a support plate. A practical approach is to exploit the multifunctionality of the elements. The contact surface area can be increased by using fillers placed on the contact surface, or alternatively, washers can be employed to reduce the contact area and control the conduction. These design considerations help optimize the thermal performance and efficiency of the system.

- Doublers
- Washers: Thermal fillers are used to enhance conduction in thermal systems. However, this choice involves a trade-off with electrical and EMI considerations, especially in applications with small gaps, like in IRES noise generator antennas. The fillers are typically electrically and thermally conductive materials. On the other hand, thermal washers are used to reduce conduction and provide structural support. They are made of thermally isolating materials, such as plastic, to prevent heat transfer. The use of thermal washers may be limited by their mechanical properties. Thermal doublers are employed to enlarge conduction and spread the heat across the in-plane thermal conductivity of the material. This helps improve the overall thermal performance and heat dissipation in the system.

Bold, straps, braids

In thermal control systems, bolts, straps, and braids play a crucial role when alternatives like fillers or doublers are not sufficient to effectively remove all the heat that needs to be transported to radiators. These elements are primarily used to establish a conductive connection between the internal components and the external part of the system. Bolts are used as mechanical fasteners to secure components and ensure good thermal conductivity between them. They provide a strong and reliable connection that facilitates the transfer of heat from the internal components to the external surfaces. Straps are flexible and flat conductive elements that serve as thermal paths to conduct heat away from critical components. They are particularly useful when dealing with irregular or non-planar surfaces where traditional rigid connections may not be practical. Braids, similarly to straps, are also used for conducting heat, but they offer additional advantages, such as flexibility and the ability to be routed in various configurations. This makes them suitable for complex thermal control requirements where rigid connections are not feasible. Together, bolts, straps, and braids are essential components in ensuring efficient thermal management, enabling the safe and effective transfer of heat from sensitive internal systems to the outer

radiators for dissipation into space.
Straps and Braid:

- Straps and braid are used to ensure conductive coupling between two parts in a thermal system.
- In certain situations, it is important to achieve thermal coupling while maintaining structural decoupling.
- Copper braids, though heavy, are commonly used in test setups for their effectiveness in conducting heat.

Thermal structures

In this slide, we have some relevant numbers that correspond to the material characteristics discussed earlier. These characteristics are expressed in terms of two key aspects:

- Thermal Conductivity for Conduction: This aspect is represented along the horizontal line. When selecting the structural material, it's essential to consider its thermal conductivity, which influences how well it can conduct heat.
- Coefficient of Thermal Expansion (CTE): The vertical line represents the thermal expansion characteristics of the material. Ideally, a material with low CTE is preferred, especially if it is close to zero or uniform across different materials, as this allows for effective coupling and proper contact between them.

To ensure proper functionality, it's important to choose materials with similar CTE values, preferably in the order of micrometers, rather than millimeters or higher. This approach will help achieve the desired thermal and structural performance of the components.

Heat pipes

Heat pipes are a versatile thermal control mechanism that can be classified as both passive and active control systems. Conceptually, heat pipes work by utilizing a fluid that moves from one side to another within the pipe due to the presence of internal grooves. These grooves facilitate the phase change of the fluid, allowing it to absorb heat at one end and release it at the other. The fluid inside the heat pipe undergoes a phase change depending on its location within the system. Ammonia is a common liquid used in heat pipes due to its favorable properties for heat transfer. Heat pipes are typically small in diameter, usually around $3-4$ mm, making them compact and efficient for thermal control applications. A typical application of heat pipes involves nesting them inside a radiator. In this configuration, the hot unit is placed inside the spacecraft, and the heat is transferred through the heat pipes to the radiator where it is dissipated into space. This serves as an effective transportation method for managing thermal loads. Alternatively, heat pipes can be coupled in parallel to enhance their heat transfer capacity, providing a flexible and effective solution for various thermal control needs.
A heat pipe is a passive thermal control device that operates on the principle of two phase cooling loops involving isothermal phase change processes. It consists of a hermetically sealed cylindrical tube with a capillary wick structure lining the inner wall, which is filled with a working fluid. The effectiveness of a heat pipe depends on the thermohydraulic properties of the selected liquids. To quantify the performance of a heat pipe, a figure

of merit called "G" is considered, which is defined as $G = K\tau/\eta$, where: K represents the heat of vaporization (increasing its value improves performance) ρ denotes density (higher density leads to better performance) τ represents surface tension (higher surface tension is beneficial) η indicates dynamic viscosity (lower dynamic viscosity is preferred) For various thermal control applications, different combinations of materials and working fluids are utilized. Some common examples include:

1. Aluminum-ammonia, which is a classical choice for space applications and operates within a temperature range of −40°C to +80°C.
2. Aluminum-propylene, suitable for extreme low-temperature scenarios.
3. Copper-water, ideal for temperatures ranging from 100°C to 200°C.

8.5.2 Active components

Heaters

Heaters and thermostats are essential active components in thermal control systems. Heaters are used to provide additional heat to specific equipment or to compensate for heat dissipation when the equipment is switched off. They are constructed using a flexible printed circuit that incorporates a metal resistor, enclosed between two layers of Kapton material. The power density for Kapton heaters reaches a maximum of 5W/cm2, with an area resistance ranging from 100 to 400 Ω/in2. To ensure safe operation, a typical margin of 25% is applied to the supplied power. These heaters can handle temperatures up to 250°C (T_{melt}) and can be effectively controlled using thermostats. The dissipated power ($Q - htr$) is calculated using the equation $Q_{htr} = V_{bus}^2/R_{htr}$, where V represents the voltage supplied, and Rhtr is the resistance of the heater. By utilizing heaters and thermostats in thermal control systems, engineers can precisely regulate temperatures, ensuring optimal performance and thermal stability for the equipment in various conditions. Heaters are essential components of active thermal control systems, serving as resistors to generate or dissipate power according to Joule's equation in order to regulate temperatures effectively. These heaters come in various sizes and are flexible, making them easy to distribute and position precisely where needed. For example, they can be strategically placed in the avionics bay to maintain strict temperature boundaries, ensuring that the temperature remains within the desired range of −5°C to +30°C. Another application of heaters involves placing a thermal blanket over specific components. This approach allows for the decoupling of the internal part from the rest of the spacecraft, enabling independent temperature regulation.

TEC

Peltier elements offer a unique thermal control solution that operates on the principle of the inverse Seebeck effect, known as the thermo-electric effect. Unlike traditional methods that convert thermal energy to electrical power, Peltier elements work in reverse, utilizing an electric current to induce a heat transfer motion from the cold to the warm junction. These elements come with both advantages and limitations. On the positive side, Peltier elements are compact, lightweight, and vibrationless. They do not require refrigerating fluids, making them more reliable and suitable for certain applications. However, their efficiency is relatively low, and the amount of thermal energy removed (Q_m) depends on factors like power (P), electric current (i), and time (t). The system must be effectively cooled down and connected to radiators to dissipate the heat removed from the hot junction. Peltier elements can be designed with stacked stages, forming a pyramid-like structure, which allows for increased cooling capacity as the heat load rises. One practical example of Peltier element usage is the Hubble Space Telescope Imaging Spectrograph (STIS), where it efficiently cools the cold junction to approximately −80°C while rejecting 17.7W of power, with the hot junction at 20°C. In summary, Peltier elements offer an intriguing approach to thermal control, particularly in scenarios where compactness, low mass, and reliability are crucial factors. However, their lower efficiency compared to other cooling methods should be considered when selecting the appropriate thermal control solution for a given application.

Radio isotope

The Radioisotope Heating Unit (RHU) is a thermal control component used in spacecraft, which operates based on the decay of plutonium. Its primary purpose is to provide heat directly to warm the spacecraft or generate electricity through Radioisotope Thermoelectric Generators (RTGs) to power heaters. Currently, there are no European manufacturers of RHUs or RTGs, but both the United States and Russia have successfully developed and utilized these devices for their deep-space missions. A single RHU typically features the following specifications:

- Mass (M): 40 grams
- Size: 3.2x2.6 cm
- Power Output (P): 1 Watt

On the other hand, a US General Purpose Heat Source (GPHS), which is a type of RTG, has the following characteristics:

- Mass (M): 1.4 kilograms
- Size: 10x5 cm

These radioisotope-based heating units and generators play a vital role in providing reliable and long-lasting power and heat sources for spacecraft, enabling them to function effectively in the harsh and remote conditions of deep space missions. The power is small and they are in localized points. They cannot be stopped once they star emitted.

Chapter 9

OBDH and Configuration

OBDH stands for "On-Board Data Handling." It refers to the system or subsystem within a spacecraft or satellite that manages and processes data generated by various onboard sensors, instruments, and systems. OBDH has several key responsibilities and functions, including:

1. Data Acquisition: It collects data from various sensors and instruments onboard the spacecraft or satellite.

2. Data Processing: It processes the acquired data, performing calculations, algorithms, and filtering to extract useful information.

3. Data Storage: It stores data for future access or transmission to the ground.

4. Control and Monitoring: It oversees the status of onboard systems, including sensors, actuators, and subsystems, to ensure the proper operation of the spacecraft.

5. Data Communication: It sends processed data and important information to ground communication systems for further analysis or mission control purposes.

6. Autonomous Control: It can make autonomous decisions based on received data, such as resolving minor issues or adapting to unforeseen situations.

9.1 Design steps

Information Required:

- Assessment of the environmental factors encountered during the mission, with a focus on radiation.
- Examination of mission phases and operational modes.
- Evaluation of data storage capacity and onboard data processing operations.
- Assessment of the available electric power profile.

Task Objectives:

- Identification of essential data management functionalities.
- Derivation of mission and system requirements.

- Conducting trade-off analyses to determine criteria for decision-making.
- Comprehensive analysis of space segment components and requirements.
- Identification of potential alternative architectural approaches.
- Estimation of the required onboard memory size.
- Evaluation of the necessary Performance Control Unit (PCU) capabilities.
- Definition of the Data Handling (DH) architecture.
- Refinement of the onboard software needs within the design.
- Selection of appropriate components and data bus.

Expected Outputs:

- Development of the system and subsystem (s/s) architecture.
- Compilation of an equipment list for the power control unit (PCU) and memory.
- Creation of power and mass budget assessments.
- Establishment of data budget projections.

9.1.1 Defining requirements

Mission requirements:

- Costumer needs (military, civil, scientific)
- number of satellites (single, multiple, costellation)
- programmatic (risk, schedule, cost)

System level processing requirements:

- For the costumer needs, it's important to define: functional capabilities and processing partitioning
- for the other it's important to: determine the physical features and the communication protocol.

Computer level requirements: For this it's important to define: throughput, memory, radiation hardness, development tools.

9.1.2 Partitioning Functionalities

The functionalities are generally partitioned between processing, performed:

- in space or on ground: in space for unbearable processing delays and allow to limit the download bandwidth; in-ground if the human interaction with processing is mandatory. This means space/ground tradeoffs are autonomy, time criticality, downlink bandwidth and uplink bandwidth.
- in HW or SW: in HW if high performance is needed; SW especially if changes in processing need to be made after HW acquisition or if the processing complexity exceeds that available on HW. This means HW/SW tradeoffs = special purpose HW, algorithmic complexity.
- processing can also be allocated between SM (Service Module(and P/L (or not) based on the level of accountability of P/L performance or if processing are different between P/L and bus.
- processing could also be allocated along organizational lines (or not) if there are geographical or other impediments to effective inter organizational commands or the interface control documents is managed within a defined organization.

9.1.3 Architectural building blocks

On-board Data Systems comprise an extensive array of essential building blocks, which encompass:

1. Telecommand and Telemetry Modules
2. On-Board Computers
3. Data Storage and Mass Memory
4. Remote Terminal Units
5. Communication Protocols and Buses

These components collectively form the core elements of the system's architecture. There are different alternatives in data flow architecture, which refers to how data is transmitted and managed within a system. Here are the alternatives:

- Central Processor: In this architecture, a central processor plays a key role. Data is transferred through point to point interfaces, which are direct connections between the central processor and various devices or components in the system. Communication in this setup necessitates dedicated wiring and specialized software to facilitate the data exchange.
- Ring: The ring architecture involves the use of a closed-loop or circular structure for data transmission. It employs established arbitration mechanisms, such as token-passing. Token-passing is a method where a "token" or permission is passed around the ring, allowing the device with the token to transmit data.

- Bus: The bus architecture employs a shared communication channel or "bus" through which processors and devices communicate. Data transmission in this architecture relies on protocol software for transmission control. Protocols define the rules and procedures for data exchange. It also implements standard interfaces to ensure compatibility between different components within the system.

The choice of architecture depends on factors such as the system's requirements, performance needs, and the desired level of flexibility and compatibility.

Centralized architetture

Centralized architecture is based on a central processor (CPU) to which all data collected by the spacecraft's sensors and instruments are routed, through point-to-point interfaces, for processing and analysis. This type of architecture is best suited for well-defined systems in which all components interact only with the central computer. It is highly reliable, and any eventual failure affects only one communication line. The advantages of Centralized Architecture are:

- Reliability: This system is highly reliable because all data traffic passes through a single central point. This simplifies data monitoring and control.
- Specialization: It is suitable for systems that require strong specialization of the central processor, for example, when high computational power or centralized data management is needed.

The disadvantages of Centralized Architecture:

- Scalability: Adding new nodes or devices requires changes to the software and hardware of the central node. This can be complex and costly.
- Cable Complexity: Since each node must have a dedicated transmission line for sending and receiving data to the central node, centralized architecture requires significant cabling and can become complicated on complex spacecraft.

Disrtibuted architeture

Distributed architecture refers to an arrangement in which the various components or nodes of a system are connected using specific structures, such as a ring or a bus.

1. Ring Architecture: In a ring architecture, an arbitration mechanism is established to control the bus (the communication pathway). It typically involves a smaller wiring harness that is distributed throughout the spacecraft (s/c). It has the advantage of limited impact when new nodes are added to the system, as it doesn't require major changes to the existing infrastructure. However, it tends to have lower reliability because each node is in line, and if one node fails, it can disrupt the entire ring.
2. Bus Architecture: Bus architecture involves processors and devices communicating via a shared bus. It relies on protocol software for transmission control. It implements standard interfaces for compatibility.

The advantages of Distributed Architecture (Ring and Bus) are:

- Scalability: Distributed architectures like the ring and bus are typically more scalable than centralized architectures. Adding new nodes is generally straightforward and doesn't require extensive changes to the existing system.
- Reduced Wiring: These architectures often require smaller wiring harnesses because the communication pathways are distributed, which can lead to reduced complexity and weight.

The disadvantages of Distributed Architecture (Ring in this case) are:

- Reliability: Ring architectures can have lower reliability compared to centralized systems. If one node fails, it can disrupt the entire ring, potentially causing communication issues.

In a bus architecture, processors and devices within a system communicate using a shared communication channel called a "bus." This bus is responsible for transmitting data between various components. To control this communication, a protocol software is used. There are two main types of bus architecture:

- Federated Bus: Federated bus architecture is a hybrid approach that combines elements of both centralized and distributed systems. Buses in this architecture can employ command-response protocols and rely on traffic arbitration to manage communication. It ensures deterministic transmission, which means that data transmission occurs predictably and consistently, reducing troubleshooting time and increasing reliability. However, one drawback is that all components within this architecture need to be developed with specific interfaces, both physical and electrical, which can add complexity during the design and development phase.
- Distributed Bus: In a distributed bus architecture, all software (SW) components are stored in non-volatile memories. During different phases of operation, the SW components executed by the system may vary. This architecture offers a high level of redundancy, as multiple processing units can be used to execute different SW components, which can enhance system reliability. However, the testing and management of such a system can be quite complex due to the dynamic nature of SW component execution and the potential for redundancy.

9.1.4 HW redudancy

Electronic Risk in Harsh Radiation Environment: In the context of electronic systems operating in a harsh radiation environment, one significant concern is Single Event Effects (SEE) caused by radiation. SEE refers to unexpected behavior or failures of electronic components due to exposure to radiation. To mitigate electronic risk in such environments, there are several solutions:

- Redundancy: Redundancy involves implementing identical backup units of hardware (HW) or software (SW). In the case of hardware redundancy, an identical backup unit is used. This approach doesn't require additional software, and testing is straightforward since the backup unit is identical to the primary one. However, it adds extra mass to the system and requires decision-making processes for in-flight usage. An example of hardware redundancy is triple modulator redundancy, where each sensor, computer, and actuator is replicated three times for fault tolerance. In the case of **software redundancy**, identical data transmission occurs three times for fault tolerance. Each word of data is transmitted three times, and as a result, data transfer is slowed down by a factor of three. A voting logic compares the three versions and selects the transmitted version.

- Distributed Processing: Distributed processing involves distributing computational tasks across multiple nodes, each equipped with its own processor, memory, and resources. This approach reduces the overall mass of the system, optimizes power consumption, and preserves system performance. However, it requires additional software to manage the distributed processing, and the overall system becomes more complex to test and validate.

9.2 CPU

he Central Processor Unit (CPU) is a critical component of a computer or spacecraft's onboard control system, and it performs several key tasks:

1. Execute Stored Programs: The CPU is responsible for running the program or software instructions that are stored in memory. This program contains the set of tasks and operations the CPU needs to perform to control various aspects of the spacecraft.

2. Interpret and Execute Commands: The CPU interprets and executes commands that are received from the spacecraft's command system. These commands are instructions sent from ground control or other control systems to perform specific actions or tasks on the spacecraft.

3. Maintain System Status and Health Data: The CPU continuously monitors and maintains the status and health data of the spacecraft. This includes checking the performance of various subsystems, sensors, and components to ensure they are functioning correctly.

4. Format Subsystem Telemetry: The CPU is responsible for formatting telemetry data generated by different subsystems on the spacecraft. Telemetry data includes information about the spacecraft's status, measurements, and other critical data. The CPU prepares this data for transmission to the spacecraft's telemetry system or for storage on a solid-state data recorder.

5. Delegate Tasks to Peripheral Processors: In some cases, the CPU may delegate specific tasks to special-purpose peripheral processors. These peripheral processors run tasks in parallel with the CPU's primary functions, helping to offload certain computational workloads and enhance the efficiency of the overall system.

9.2.1 Sizing steps

CPU sizing involves determining the appropriate characteristics and capacity of the Central Processor Unit (CPU) for a specific system or application. Here are the steps and considerations involved in CPU sizing:

1. Identify Functionalities and Partitioning: First, identify the functionalities that the CPU will need to handle. Partition these functionalities among hardware (HW), software (SW), and firmware. Firmware typically resides in non-volatile memory, such as ROM (Read-Only Memory), and is used for critical processes.

2. Evaluate Instruction Set Architecture (ISA): ISA refers to the machine code format related to a specific processor. It defines the interface between software developers and hardware. ISA can be of two types: General Purpose ISA, which supports various types of processing but may be less performant, and Customized ISA, which is dedicated to specific processing tasks and offers higher performance.

3. Select Software Languages: Decide which software languages to use. Options include assembly language, which is more efficient and compact, and high-level languages, which are cheaper, easier to develop, maintain, and test.

4. Assess Software Needs: Categorize processing tasks into different classes. For example: Central system software handles instructions from the Attitude Control and Determination System (ACDS) and must be timely and highly intensive, system management software processes instructions in case of failure and manages Payload (P/L) logistics, often dealing with simple instructions, mission data software deals with the processing of large amounts of data, such as Ephemerides, operating software manages computer resources and computes basic instructions.

5. Estimate Software Size: Estimate the size of the software based on factors like throughput estimation, which considers the number of instructions relative to time. The number of instructions depends on the required functions, while time depends on the clock speed. If different types of instructions are used, consider the instruction mix as well.

6. Margin: Consider adding a margin to the estimated software size, typically 400%. This margin is gradually reduced until it reaches 100% at the launch stage.

7. Software Sizing Process: The software sizing process begins with defining memory requirements in terms of words, which is language-agnostic. Then, it progresses to Source Lines of Code (SLOC) inclusion. The final outputs include CPU time (based on instruction count and clock rate) and MIPS (Million Instructions Per Second).

8. CPU Time Calculation:

$$CPUTime = InstructionCountCPI(CyclesPerInstruction)/ClockRate \tag{9.1}$$

$$MIPS = ClockRate/CPI \tag{9.2}$$

In general, the goal is to keep CPI (Cycles Per Instruction) low to minimize power consumption and heat production by the CPU. Accurate CPU sizing is crucial to ensure that the CPU can handle the workload efficiently and effectively for a given application or system.

9.3 Memory

There are two main types of memory used within the On-Board Data Handling (OBDH) system:

- RAM (Random Access Memory): RAM stands for Random Access Memory. It is a volatile type of memory, which means that if the power to the devices using RAM is interrupted or turned off, all the data stored in RAM are lost. RAM is known for its high speed and ability to read and rewrite data quickly. In RAM, any word (a unit of data) can be accessed at any time. Typical RAM capacities in OBDH systems range from 4 megabytes (M) to 256 megabytes (M) of data storage.

- ROM (Read-Only Memory): ROM stands for Read-Only Memory. It is a non-volatile type of memory, meaning that even if the electrical power is removed or turned off, the data stored in ROM are not lost. ROM is used to store data that is permanent and does not need to be modified or rewritten by the system. Typical ROM capacities in OBDH systems range from 1 megabyte (M) to 64 megabytes (M) of data storage, and the data in ROM is typically organized in 8-bit words.

9.4 Mass storage

In OBDH systems, it's common to have a need for mass data storage, especially since RAM (Random Access Memory) has limited capacity. Mass storage is essential for storing a significant volume of data. There are several main types of mass storage options, and the text mentions three of them:

- Disks: Traditional hard disks are one form of mass storage. They use spinning disks to read and write data. While they offer high capacity, they are not commonly used in space applications due to their mechanical nature, which makes them more vulnerable to shock and vibration.

- Integrated Circuits (ICs): Integrated circuits can be used for mass storage. However, the term "ICs" is quite general and can refer to various types of semiconductor devices, including flash memory and other memory technologies.

  Solid-State Mass Storage: Solid-state mass memory is the most commonly used type of mass storage in OBDH systems. It is composed of large banks of random-access memory (RAM) and integrated circuits that are interconnected to implement mass storage. Key characteristics of solid-state mass storage include:
  1)High power demand: It consumes relatively more power but offers high-speed data access and storage density.
  2) Storage capacity: Solid-state mass memory can have capacities as large as 4 gigabytes (Gb) or more.
  3) Cost: The cost of RAM memory used in solid-state mass storage is relatively high compared to some other storage technologies.
  There are two memory technologies used for solid-state data recorders:

  1. Synchronous Dynamic Random-Access Memory (SDRAM): This technology is used for various types of data storage and offers certain advantages.

2. Flash Memory: Flash memory is another technology used for mass storage. It is known for its high density and speed. However, it has limitations in terms of total-dose radiation immunity. It is commonly used for storing Payload Data, Platform Data, and Safe Guard Memory.

9.5 Microcontrollers

A microcontroller is a highly integrated computer system that is designed on a single chip. It comprises several essential components, including a central processor, and it incorporates various support functions such as program memory, input/output (I/O) interfaces, and analog-to-digital (A/D) converters. Microcontrollers are designed to be compact and highly integrated, meaning that they contain multiple functions and components within a single chip. These components include a central processor unit (CPU), memory, and interfaces for connecting to external devices. In addition to the CPU and memory, microcontrollers include support functions such as program memory, which stores the software that controls the microcontroller's operations. They also have I/O interfaces that allow them to interact with other devices and systems, as well as analog-to-digital converters that enable them to process analog signals from sensors. Microcontrollers are widely used in various applications where software-based data acquisition and simple control are required. They offer higher flexibility and autonomous capacity compared to pure hardware solutions. Common applications of microcontrollers in spacecraft subsystems include Power Systems (PS), Electrical Power Systems (EPS), antenna pointing systems, Attitude and Orbit Control Systems (AOCS) and Guidance, Navigation, and Control (GNC) sensors, as well as Remote Terminal Unit (RTU) control.

9.6 Microprocessors

A microprocessor is a central processing unit (CPU) designed as a general-purpose processor with a complex architecture. Microprocessors consist of several essential components, including:

- Arithmetic Logic Unit (ALU): This is responsible for performing arithmetic and logical operations.

- Control Unit: It manages the execution of instructions and controls the flow of data within the microprocessor.

- Registers: These are small, high-speed storage locations used for temporarily holding data during processing.

- Cache Memory: Cache memory is a high-speed, small-capacity memory used for storing frequently accessed data to improve processing speed.

- Various Instruction Sets: Microprocessors support a wide range of instruction sets, which are sets of machine-level commands that the processor can execute.

Microprocessors are capable of executing a wide range of tasks and running complex operating systems. They are versatile and can handle various computing tasks, making them suitable for applications where general-purpose processing is required. Microprocessors typically require external components to function effectively. These components may include:

- Memory Modules (RAM, ROM): External memory is used to store data and program code.
- Input/Output Interfaces (I/O Ports): These interfaces allow the microprocessor to communicate with external devices.
- Timers: Timers are used for tasks such as measuring time intervals or generating clock signals.
- Peripheral Devices: Additional devices may be required for functions like communication, data storage, and interfacing with external devices.

They are often programmed using high-level programming languages, which offer a more abstract and human-readable way to write software. Microcontrollers are integrated systems with simplified architectures, including all necessary components on a single chip. They are often programmed using low-level languages and are suitable for embedded systems and real-time applications. Microprocessors, on the other hand, are more versatile and require external components for operation.

9.7 BUS

The onboard command and control bus in a spacecraft serves several essential functions related to the acquisition of data from sensors, commanding of actuators, data transfer between onboard instruments and control computers, and the distribution of time information. Here are the key functions and considerations regarding this bus:

- Acquisition of Data from Sensors: The bus is responsible for collecting data from various sensors placed on the spacecraft. These sensors may measure parameters such as temperature, pressure, radiation levels, and more.
- Commanding of Actuators: Actuators are devices that carry out specific actions or operations based on commands received from the bus. The bus is in charge of sending commands to these actuators to control various spacecraft functions.
- Transfer of Data Between Instruments and Computers: The bus facilitates the exchange of data between onboard instruments and control computers. This data transfer is crucial for monitoring and controlling the spacecraft's systems and instruments.
- Distribution of Time Information: Time synchronization is critical for coordinating various operations on the spacecraft. The bus ensures that accurate time information is distributed to all relevant components and systems.
- Distributed Processing: The architecture of the flight control system is evolving toward a distributed processing approach. This means that specialized microcontrollers are used to control intelligent actuators through digital communication. This distributed architecture can enhance spacecraft efficiency and flexibility.

To meet these functions and requirements, the bus must have certain capabilities, including: ability to acquire synchronous data frames from sensors and transmit them synchronously to actuators and the ability to transfer asynchronous and isochronous data packets (data packets with a consistent time interval between them) between nodes. There are several alternatives for implementing such onboard command and control buses. Two common options are:

1. MIL-STD-1553B: This is a widely used standard for spacecraft communication buses. It operates at a fixed data rate of 1 megabit per second (1 Mbit/s). It is suitable for various spacecraft applications. CAN (Controller Area Network):

2. CAN is another communication protocol used in spacecraft. It supports lower to medium speeds and offers data rates of up to 5 megabits per second (5 Mb/s). CAN is known for its reliability and is commonly used in aerospace and automotive applications.

9.8 RTU

The Remote Terminal Unit (RTU) is a critical component in the operation of a spacecraft. It serves specific functions related to data acquisition and actuator control, and it operates as part of a distributed control system. The primary role of the RTU is to offload the On-Board Computer (OBC) from certain tasks, particularly those related to analog and discrete digital data acquisition and actuator control. By taking on these responsibilities, the RTU helps distribute the control functions across the spacecraft system. The RTU exemplifies a distributed control system, where specific units or components handle certain tasks independently. In this context, the RTU is responsible for tasks related to data acquisition and control, allowing for more efficient and flexible spacecraft operation. Typically, the RTU is a non-intelligent unit, which means it does not host microprocessors or microcontrollers inside. Instead, it performs its functions based on predefined rules and commands received from the central control system. The RTU is responsible for several critical tasks, including:

- Gathering Analog and Digital Telemetry: It collects data from various sensors and units on the spacecraft. This telemetry data includes measurements of various parameters.

- Controlling AOCS Actuators and Sensors: The RTU manages and controls actuators and sensors associated with the Attitude and Orbit Control System (AOCS). These systems are crucial for spacecraft orientation and maneuvering.

- Controlling PS (Power Systems): It oversees and controls components of the spacecraft's Power Systems (PS), ensuring the distribution of electrical power to various subsystems.

- Controlling SA Equipment: The RTU is involved in controlling and managing equipment related to the spacecraft's communication or scientific instruments (SA equipment).

- Distributing Power: It plays a role in distributing electrical power to heaters and active loads on the spacecraft. This is essential for regulating temperature and powering various systems.

The RTU is typically found on medium to large-sized spacecraft, where the complexity and volume of data acquisition and control tasks necessitate its presence. Smaller spacecraft may have simpler control systems and may not require a dedicated RTU.