A Fractionated Spacecraft Architecture
For Earth Observation Missions

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FOR EARTH OBSERVATION MISSIONS

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Abstract

A fractionated spacecraft is a satellite architecture for which the functional capabilities of a conventional spacecraft are distributed across multiple modules (spacecrafts) which fly separately and interact through wireless links. Removing the physical dependencies between the resources, or subsystems, of a spacecraft, brings several attributes, such as flexibility and robustness, which can be exploited for the benefit of the earth observation payloads. In this manner, an infrastructure network which is composed of resource modules can be put into a sun-synchronous orbit for the benefit of such payloads. However, fractionating a spacecraft and letting the different subsystems fly separately leads to several technological concerns which are related to the shared resources within the fractionated spacecraft network. Regarding these technology implications, collision free close proximity cluster flying configurations were evaluated initially. Then a realization approaches were also discussed via system analysis for shared resources, namely guidance, navigation and control, communications, data handling and power. Notional spacecraft architecture was determined in the light of these discussions and the sizing of the modules within this architecture was made based on an incremental launch, or one module per launch, approach. Finally it was concluded that the resource capability increase is more efficient in terms of data storage and processing when compared to wireless power transfer.
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<th>Description</th>
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<tbody>
<tr>
<td>DARPA</td>
<td>Defence Advanced Research Projects Agency</td>
</tr>
<tr>
<td>F6</td>
<td>Future Fast Flexible Fractionated Free-Flying Spacecraft united by Information eXchange</td>
</tr>
<tr>
<td>I&amp;A&amp;T</td>
<td>Integration, Assembly and Test</td>
</tr>
<tr>
<td>JAXA</td>
<td>Japanese Aerospace Exploration Agency</td>
</tr>
<tr>
<td>LEO</td>
<td>Low Earth Orbit</td>
</tr>
<tr>
<td>ENVISAT</td>
<td>The European ENVIronment SATellite</td>
</tr>
<tr>
<td>GNC</td>
<td>Guidance, Navigation and Control</td>
</tr>
<tr>
<td>AOCS</td>
<td>Attitude and Orbit Control System</td>
</tr>
<tr>
<td>C&amp;CDH</td>
<td>Communications, Command and Data Handling</td>
</tr>
<tr>
<td>CDH</td>
<td>Command and Data Handling</td>
</tr>
<tr>
<td>WPT</td>
<td>Wireless Power Transfer</td>
</tr>
<tr>
<td>SPOT</td>
<td>Systeme Probatoire pour l'Observation de la Terre - French remote sensing satellite network</td>
</tr>
<tr>
<td>NEXTAR</td>
<td>NEC Next Generation Star</td>
</tr>
<tr>
<td>GPS</td>
<td>Global Positioning System</td>
</tr>
<tr>
<td>NIR</td>
<td>Near Infrared</td>
</tr>
<tr>
<td>ADEOS</td>
<td>Advanced Earth Observing Satellite</td>
</tr>
<tr>
<td>GOSAT</td>
<td>Greenhouse Gases Observing Satellite</td>
</tr>
<tr>
<td>ALOS</td>
<td>Advanced Land Observing Satellite</td>
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Cluster Flying Nomenclatures

\( R_\oplus \): Earth’s equatorial radius
\( a \): Semimajor axis of the reference orbit
\( e \): Eccentricity
\( i \): Inclination
\( u \): Mean argument of latitude
\( \Omega \): Right ascension of ascending node
\( J_2 \): Geopotential second-order zonal coefficient
\( r_R \): Satellite position in radial direction
\( r_T \): Satellite position in along-track direction
\( r_N \): Satellite position in cross-track direction
\( v \): Satellite velocity
\( T_d \): Orbital period in days
\( T_s \): Orbital period in seconds
\( T_e \): Period of the relative eccentricity vector motion
\( \Delta \): Difference operator
\( \Delta e, i \): Relative eccentricity or inclination vector
\( \delta e, i \): Relative eccentricity or inclination vector modulus
\( \theta \): Relative ascending node
\( \phi \): Relative perigee
\( D \): Relative difference operator \((\Delta_1 - \Delta_2)\)
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CHAPTER 1

Introduction

“But know right from wrong
Or in the flood
You’ll build an Ark
And sail us to the moon”

Sail to the Moon, Hail to the Thief, Radiohead
1. Introduction

Within last 5 years, a novel spacecraft configuration idea was introduced by the Defence Advanced Research Project Agency (DARPA) of United States. The idea was to physically separate the spacecraft subsystems into several modules and operate them as a cluster on orbit using wireless communication and power transfer features. By removing the physical connectivity between the subsystems or resources, such as power, generated by these subsystems, this configuration would enhance the flexibility and robustness throughout the mission lifetime beginning from the design stage to the end of mission. Here flexibility and robustness are defined as the ability of a space system to react and adapt to various forms of uncertainty covering module failures, changes in design and operational requirements, fluctuations in funding and so on. Therefore the risks involved throughout the mission life cycle would be reduced. For instance, when a subsystem module fails, this module can be replaced by launching a new module and the spacecraft can be made operable again. With respect to the given definition of flexibility and robustness, traditional satellite configurations, namely monolithic satellites, are associated with more uncertainty and enhancing these attributes for them are limited with novel operations.

Introducing enhanced capabilities, fractionated spacecraft concept has the potential to bring new research, design and development perspectives. In this thesis, the study of a fractionated spacecraft architecture which is defined for the benefit of earth observation payloads is presented. In this manner, the phases of this study can be listed as follows:

i. Problem Definition
ii. System Analysis
iii. Modelling and Sizing
iv. Synthesis

Problem definition

Physical separation of the resources or constituents of a spacecraft makes it also possible to decouple the design, development and operations of different payloads to be supported by the subsystems, or infrastructure modules, of the fractionated spacecraft. Therefore this spacecraft architecture is very interesting to study in terms of fundamental missions such as earth observations for enhancing the capabilities and the sustainability. If an infrastructure cluster with a long lifetime can be put into sun-synchronous orbit, which is the case for many earth observation missions, then various payloads, as long as the orbit permits, can be launched to this infrastructure and benefit from the generated resources. However the technological and feasibility analyses of such a system are needed. Here the feasibility of a fractionated spacecraft is made through value centric design perspective and this approach requires a value proposition. Then the lifecycle economical analysis is performed in terms of the benefits (through value proposition) and the associated costs of the spacecraft. In this manner, it is decided for this study to analyze the technological aspects, limitations and the realization of an infrastructure concept for earth observation missions. Later, this study can be input to a value centric feasibility analysis.

System Analysis

The system analysis is performed in terms of the identified shared sources such as navigation management, data handling, communications and power generation. The associated technologies are also discussed and the realization perspectives are provided under the same chapter. Exception to this is the cluster flying for which a dedicated chapter is introduced. In the end of the system analysis a notional spacecraft architecture is proposed.
Modelling and Sizing

Based on the system analysis, a modular and parametric bus sizing code was developed. Using this sizing code the infrastructure and mission modules are sized with respect to their individual requirements. This sizing code, associated assumptions and the results are presented within the dedicated modelling and sizing chapter.

Synthesis

System analysis, modelling and sizing results are evaluated and the conclusions are presented together with a future study perspective at the end of this thesis.

The overall design process can be summarized by below figure:

![Flowchart for the design approach](image)

**Fig 1.1 Flowchart for the design approach**

1.1. Fractionated Spacecraft

A fractionated spacecraft is a satellite architecture where the functional capabilities of a conventional monolithic spacecraft are distributed across multiple modules which interact through wireless links (Fig. 1.2). The physical independence between the subsystems in this architecture provides the possibility for a plug and play architecture which increases the value of this concept.

![Fractionated (ref: DARPA System F6) and Monolithic (conventional) Satellites](image)

**Fig 1.2 Fractionated (ref: DARPA System F6) and Monolithic (conventional) Satellites**
By providing the sections below an insight on the basic terminologies, concepts and attributes regarding the fractionation of a spacecraft will be provided. These sections are summarized from various studies which are stated in the references ([1.1] to [1.5]).

**Fractionation**

Fractionation concept emerges on the idea of changing the physical connectivity of the various components within a spacecraft into wireless connections. In this manner, a fractionated space system is characterized by physically independent constituents that may or may not collaborate to provide a benefit or value to a corresponding mission. Here we can define modules, or constituents of a fractionated system, as payload modules which are the ones including mission specific instruments and functionalities; and infrastructure (or resource) modules which support these mission specific payload modules. If a system has its modules physically dependent then the system is considered to be monolithic.

A modular system has constituents which can be developed, tested and integrated concurrently. In this manner, a fractionated system is naturally modular, though a modular system needs not to be fractionated. Understanding this characteristic is important to appreciate the benefits of a fractionated system.

Fractionation can be achieved in two ways which are defined as heterogeneous and homogeneous fractionations. First is the decomposition of a given spacecraft into many dissimilar subsystems or functional elements. This is called heterogeneous fractionation and the functionality of the system is distributed into many distinct and independent infrastructure and payload modules. Second is the decomposition into identical or functionally similar modules which considers the redundancy of a subsystem or decoupling multiple payloads. An example to homogeneous fractionation is GPS satellites for which the payloads are decoupled. A third class called “hybrid” can be also considered by combining homogeneous and heterogeneous fractionation. This would also provide a measure of homogeneity within a heterogeneously fractionated system.

![Homogeneous, Heterogeneous and Hybrid Fractionation](image)

**Fig 1.3** Homogeneous, heterogeneous and hybrid fractionation

If we consider each type of spacecraft as a subsystem of a cluster, then cluster A would be an example for homogeneous fractionation, B for the heterogeneous fractionation and C for the hybrid fractionation.

**Attributes of a Fractionated Space Architecture**

With fractionated spacecraft architectures, new concepts of missions which are not possible to perform with a monolithic spacecraft can be enabled. In addition, these architectures can also offer improvements on the implementations of current space missions. In this manner, some of the attributes of fractionated space architectures are discussed in the previous studies (as mentioned before) in terms of cost, flexibility and robustness in addition to the negative perspectives. Some outlines of these are given below in terms of each:
Cost

A comparison between monolithic and fractionated spacecrafts in terms of lifecycle costs through the phases of a mission is provided through development, fabrication, launch, and operation as:

- **Development Phase:**

  Main advantage of fractionated system during the development phase is the ability of decoupling the individual module requirements. This has three main impacts on the development phase. Firstly, typical pointing, jitter, isolation and other requirements of the payload modules will be only considered for the particular payload module, therefore not on the whole system. This could lead to downsizing of attitude control systems and reduce the integration, assembly and test (IA&T) costs. Secondly IA&T costs could further be reduced as the systems engineering concerns would only be at the individual module level. Technically, concerns would be less on the interactions between the subsystems in terms of vibration, thermal, and electromagnetic interferences. Programmatically, development of a single payload or a module wouldn’t be on the critical path of the program. Thirdly, decoupling of the design reliability and design lifetime would be possible. Therefore each module’s design lifetime can be optimized individually and this ability can provide significant cost savings.

- **Fabrication:**

  Firstly standardization of the development of particular infrastructures, and for some cases payload modules, can reduce recurring labour costs through increase in the individual and organizational knowledge and experience in terms of fabrication of multiple identical modules. Standardization can also lead to the commoditization of multiple items, especially for the infrastructure modules. For instance, automotive and computer industries are good examples of commoditization. This can further provide a volume based cost reductions considering mass production analogy.

- **Launch:**

  Fractionated architectures favour small launch vehicles which are considered to be responsive. (Here responsiveness can be perceived as the ability to react an uncertainty during the mission in a rapid manner.) Although the use of such small vehicles is a potential advantage, the economic efficiency is not well estimated so far. The advances on the development of both such small and responsive launch vehicles in addition to the fractionated architectures are needed. However fractionated architecture modules can still be launched as bundles on conventional launch vehicles and as secondary, or piggyback, payloads if the orbit conditions are likely to be satisfied.

- **Operations:**

  Operating a cluster can be more complex and cost inefficient when compared to operating a conventional satellite. The main difficulty is to provide collision free trajectory planning which induces increased autonomy on the system. Therefore it is stated that the autonomous cluster operations is an explicit objective of the fractionated architectures development. In addition, cost reduction is desired by simplification of ground assets via treating ground segment as another node, or sets of nodes, in the network of a fractionated architecture.

Cost considerations through comparison with respect to a monolithic spacecraft cannot give a clear conclusion. Fractionated spacecraft architectures have both some certain advantages and disadvantages in terms of cost. Therefore the value centric design which compares the benefits and
costs is essential. In addition, to clear the ambiguities on potential advantages and disadvantages, further developments on desired objectives are needed. For instance, launch vehicles and operations are the two areas that fractionated architecture can either suffer or provide more opportunities.

**Flexibility**

Flexibility can be defined as the system’s ability to change on demand. Flexibility can offer to owner, developer, operator or stakeholder taking decisions and actions with respect to the perturbations and/or uncertainties that appear throughout the mission lifetime. The attributes can be considered under the flexibility are scalability, evolvability, adaptability and maintainability.

![Fig 1.4 Flexibility attributes: scalability, evolvability, adaptability, maintainability (ref: DARPA System F6)](image)

**- Scalability:**

One of the advantages of a fractionated architecture is that it would be possible to add new modules to the system. This is expressed as scalability of a system. Hence system’s capability or ability can be enhanced throughout the mission by incremental deployment of individual modules in demand. In addition, the system could be also scaled down during the design and development phase in need.

**- Evolvability:**

Similarly when a certain technology is advanced by a newer one, it would be easier to adapt the use of the advanced technology with a fractionated architecture. This can be also defined as upgradeability. Upgradeability can be achieved by simply changing the module with a new one or even in the development phase without affecting the whole system.

**- Adaptability:**

Adaptability is the ability to reconfigure existing system functionality with respect to the needs and uncertainties. This ability can be also expressed as reconfigurability or versatility of a system.

**- Maintainability:**

In a fractionated spacecraft architecture, the modules that are failed or near end of life can be individually replaced. Therefore this reduces the necessity of performing complex on-orbit servicing to increase the lifetime of a spacecraft.

Flexibility is a unique characteristic of a fractionated architecture. If a monolithic satellite is thought instead, only way to satisfy above demands and tackle uncertainties would be launching another satellite.
Robustness

Flexibility refers to the changes that are made by decision. On the other hand, robustness is defined as the intrinsic capability of a system to maintain its nominal mission in the existence of various internal and external perturbations to the system throughout the mission lifetime. The attributes which constitutes the robustness are reliability, survivability, resilience to fragility and fault tolerance.

- Reliability:

Reliability depicts the probability that the system survives and performs operations under nominal conditions for a fixed system lifetime. There are two options for designer to affect the reliability of a system. These are redundancy and qualification. Redundancy is obtained by duplicating the modules or components distributed over the fractionated spacecraft. Here reliability is also enhanced by enabling modules to share the resources within the system. On the other hand, the qualification procedure is not different from the one for monolithic spacecraft.

- Survivability:

Survivability extends the reliability by taking into account the off-nominal conditions. Therefore it can be defined as the ability of a system to function under off-nominal or unanticipated conditions. For example a satellite that can survive temperature fluctuations, meteorite strike or anti-satellite attack would be termed as survivable. As fractionated spacecraft has a spatially distributed nature, survivability can be achieved in an easier way when compared to the monolithic spacecraft.

- Fragility:

This is the tendency, or frequency, of a system to experience unmodeled, catastrophic and cascading failures. This is related to the complexity of the system. Decomposition of the system reduces the concerns related to effective complexity to module level. The complexity of a fractionated system to be managed is the complexity of software systems, not electro-mechanical.

- Fault Tolerance

Fault tolerance is the gradual loss of system functionality due to one or more failures. This attribute depicts the difference in the response of fractionated and monolithic spacecraft to the conditions that the system turns out to be not robust. For example, the change in functionality for various stimuli and conditions can be dramatic for monolithic spacecrafts while a well designed fractionated architecture should degrade capability incrementally.
**Industrial Effects**

Decomposition of a spacecraft can have industrial effects as well. Firstly, it would be possible to increase collaborations in terms of development and fabrication. Secondly, provided a standard open interface, the competitive opportunities can be increased in the industry. Thirdly, launching fractionated spacecraft modules can lead to a significant market. Finally, the fractionated spacecraft architecture is desired to provide improved Net Present Value (NPV) and shareholder returns for operators by improving the value proposition of fractionated spacecrafts.

**Negative Perspectives**

There are also concerns about the fractionated spacecrafts in addition to their benefits. This innovative concept offers a new paradigm that forces the industry and academy to advance on certain technologies and to change traditional aspects in terms of design, development, launch and operations. Therefore, it is not easy to persuade industry and governments to put effort and investment on a new concept at first. However, academy is the most suitable address to assess the feasibility of such a new concept and also to identify and guide the required technological developments. Therefore in this section, it is more reasonable to mention technological concerns about fractionated spacecraft architecture. Firstly, it was discussed and shown on previous studies that fractionated architectures will have larger mass and in some cases more life cycle costs. Secondly, due to its physically dispersed nature, more complexity is associated; especially in terms launch and mission operations. Also new manufacturing, integration, assembly and test operations are required and they can be difficult to establish. In addition, due to its dependence on new and unproven technologies, there are certain risks of on-orbit failure. These failure risks further increase with the increase of inter-module dependency, e.g. resource sharing. Thirdly, for a given mission, this architecture may offer sub-optimal performance with increasing use of common modules. Finally, in order to appreciate the benefits of this architecture, value-centric perspective which is not common and not widely accepted is needed.

As it can be seen, if there will be some achievements on technological developments and on the assessment of net value for a mission using fractionated architecture then it would be easier to see the future of this concept more clearly. Therefore more study is still needed on the feasibility of fractionated architectures.
CHAPTER 2

Mission

“Up above aliens hover
making home movies for the folks back home.
Take me on board their beautiful ship,
show me the world as I’d love to see it.”

Subterranean Homesick Alien, O.K. Computer, Radiohead
2. Mission Overview

Earth observation missions are fundamental missions. Therefore several earth observation missions and programs were, are being and will be realized. For instance, Japanese Earth Observation Missions are focused on atmospheric, marine and land observations [2.1]. On the other hand, the programs such as SPOT and LANDSAT imply the necessity of sustainability of earth observation missions.

One of the desired outcomes of the fractionated spacecraft concept is enhancing the mission lifetime and spacecraft capability in a more robust way than before. If an infrastructure can be put into a convenient sun-synchronous orbit then it would be possible to launch many mission modules to be used in earth observation missions instead of designing, manufacturing and launching many dedicated big spacecrafts complementing each other. Therefore the main focus of this study is to designate a space system (or an infrastructure) which is capable of supporting at least two distinct mission modules (scientific instruments) used in earth observation missions. If there can be a feasible infrastructure, then it would be possible to establish a “space port” for different payload modules. Hence flexibility and robustness of Earth observation missions would be enhanced as a consequence of establishing such an infrastructure.

2.1. Earth Observation Missions

Earth observation can be defined as the acquisition of information on the physical, chemical, and biological systems that exist on Earth. Making use of spaceborne remote sensing techniques sustains our knowledge on those systems in an advanced way. We can monitor and assess the status of and the changes in the natural and built environment on the Earth continuously with wide area coverage.

Remote sensing techniques allow us to observe size, shape and character of the objects without being in direct contact. This is done by studying the characteristics of the absorbed, reflected and/or radiated electromagnetic radiation which is recorded by on board sensors. Passive and active remote sensing are the two ways to perform spaceborne remote sensing measurements. Passive remote sensing uses the electromagnetic radiation from an external source such as sun light for absorption characteristics and earth (thermal emission from land, oceans, etc.) for emission characteristics. On the other hand active remote sensing uses the reflection of the radiation (usually microwave) emitted by the remote sensing platform itself. This provides ability to make measurements during day and night.

![Fig 2.1 Spaceborne Passive Remote Sensing](image source: Department of Physics, National University of Singapore)
When the EM wave travels through the atmosphere they can experience absorption and scattering with respect to their wavelengths. Some wavelengths are strongly absorbed and blocked by the atmosphere. Therefore there are certain wavelength bands that are commonly used for remote sensing purposes. These are optical, near infrared (NIR) and thermal infrared bands for passive remote sensing and microwave for the active remote sensing.

Below some of the specific earth observation missions conducted by Japanese Aerospace Exploration Agency (JAXA) [2.1] can be found (h: altitude, i: inclination, T: period):

**Land Observations:**

*Example Mission: Advanced Land Observing Satellite (ALOS)*
- **Orbit:** Sun-Synchronous Sub-Recurrent, h: 700 km, i = 98 deg., T = 98.7 min.
- **Recurrent Period:** 46 days
- **Instruments:** Panchromatic Remote-sensing Instrument for Stereo Mapping (PRISM), Advanced Visible and Near Infrared Radiometer type 2 (AVNIR-2), Phased Array type L-band Synthetic Aperture Radar (PALSAR)

**Atmospheric Observations**

*Example Mission: Greenhouse Gases Observing Satellite “IBUKI” (GOSAT)*
- **Orbit:** Sun-Synchronous Sub-Recurrent, h: 667 km, i = 98 deg., T = 98 min.
- **Recurrent Period:** 3 days
- **Instruments:** Thermal and Near Infrared Sensor for Carbon Observation (TANSO) which is composed of Fourier Transform Spectrometer (FTS) and Cloud Aerosol Imager (CAI)

**Advanced Earth Observation Missions**

*Example Mission: Advanced Earth Observing Satellite II (ADEOS II)*
- **Orbit:** Sun-Synchronous Sub-Recurrent, h: 803 km, i = 98.62 deg., T = 101 min.
- **Recurrent Period:** 4 days
- **Instruments:** Advanced Microwave Scanning Radiometer, Global Imager, SeaWinds, POLDER(Polarization and Directionality of the Earth’s Reflectance), Improved Limb Atmospheric Spectrometer II, Technical Data Acquisition Equipment (TEDA)

As it can be seen from the examples, sun synchronous sub recurrent orbits are used commonly for earth observation missions with variety of instruments on board.

### 2.2. Mission Objectives and Requirements

The aim of this study is to designate an infrastructure which is capable of supporting payload, or mission, modules used for earth observation purposes. Usually the type of the mission and related payloads drive the mission requirements. However the fractionated spacecraft architecture brings a different perspective in terms of space systems design. With this perspective, the designer is not obliged to optimize a space system for certain payloads as long as the required resource generation and distribution is achieved throughout the spacecraft network.

On the other hand, a design process is driven by the objectives and requirements. Here the design objectives are the ultimate ones which are aimed to be realized. On the way to realize a fractionated architecture there are already some goals and objectives proposed by [1,4]. Taking them into account, the objectives for this study can be provided below:
### 2.2.1. Design Objectives

<table>
<thead>
<tr>
<th>OBJ-1</th>
<th>Provide a concept for an on-orbit infrastructure network that is capable of accepting new payload modules to be launched later.</th>
</tr>
</thead>
<tbody>
<tr>
<td>OBJ-2</td>
<td>Provide an approach for an optimal distribution of the capability and reliability across the network of spacecraft</td>
</tr>
<tr>
<td>OBJ-3</td>
<td>An infrastructure made of 2 to 3 support modules to which payloads can login</td>
</tr>
<tr>
<td>OBJ-4</td>
<td>Each module should be in the mini scale(^1), for instance they shall have a mass of less than 500 kg.</td>
</tr>
</tbody>
</table>

\(^1\) *Mini satellite scale covers the region of 100 to 500 kg according to [2.7]*

| OBJ-5 | Enhanced robustness and safety through safe cluster navigation |

Taking the above objectives into consideration together with the general requirements for Earth observation missions, the main mission driving requirements can be summarized below:

<table>
<thead>
<tr>
<th>REQ-1</th>
<th>The proposed infrastructure shall be able to perform safe cluster navigation</th>
</tr>
</thead>
<tbody>
<tr>
<td>REQ-2</td>
<td>The proposed infrastructure shall be capable of <em>accepting</em> and <em>supporting</em> at least two distinct mission modules used for Earth observations</td>
</tr>
<tr>
<td>REQ-3</td>
<td>The proposed infrastructure shall provide continuous communication availability with a decent quality</td>
</tr>
<tr>
<td>REQ-4</td>
<td>Data processing and storage shall be capable of supporting the demands of distinct mission modules</td>
</tr>
</tbody>
</table>
2.3. Orbit Considerations

It is important for passive remote sensing to make observations over a region where the illumination is constant. Therefore the collected data will always be under the same condition. This condition can be achieved by keeping the angle between the orbital plane and the sunlight direction constant. With sun-synchronous orbit, the area that the satellite flies over always gets the same sunlight angle. Therefore the earth observation satellites are usually put into sun-synchronous orbit. If a satellite is also desired to observe the specific regions on the earth with certain intervals, the altitude and inclination can be adjusted to make satellite repeat its orbit periodically. This combined orbit is called sun synchronous – sub recurrent orbit. Regarding these considerations the infrastructure shall be accommodated in a sun synchronous orbit.

The orbit characteristics are mainly shaped under the mission and specific payload requirements. Similar to the considerations in the mission requirements case, the resulting orbit must comply with as much general requirement of the payloads used in Earth observations as possible. In our case, orbit must be designed with respect to two main perspectives. Firstly the orbit should be selected such that as much mission modules as possible will be able to perform optimally. Secondly the orbit must also satisfy the requirements for optimum cluster flying.

A good example in terms of satisfying the requirements of distinct mission payloads would be the orbit design considerations of the European Environmental Satellite, ENVISAT. The 8-ton satellite accommodates an array of nine Earth observation instruments performing global and regional land, water, ice and atmospheric observations. The orbit selection considerations of ENVISAT are summarized in a table 2.1 provided below:
### Table 2.1 ENVISAT Orbit Design Considerations [2.3]

<table>
<thead>
<tr>
<th>Consideration</th>
<th>Influence Factors</th>
<th>Orbital Parameter</th>
</tr>
</thead>
<tbody>
<tr>
<td>observation frequency</td>
<td>swath width; revisit time</td>
<td>altitude</td>
</tr>
<tr>
<td>global access</td>
<td>maximum latitude; spacing between ground-tracks</td>
<td>inclination, altitude</td>
</tr>
<tr>
<td>regular ground pattern</td>
<td>synchronous or drifting orbit</td>
<td>altitude</td>
</tr>
<tr>
<td>regular illumination conditions</td>
<td>sun-synchronism</td>
<td>inclination and altitude</td>
</tr>
<tr>
<td>aliasing of solar tides</td>
<td>sun-synchronism</td>
<td>inclination and altitude</td>
</tr>
<tr>
<td>aliasing of all tides</td>
<td>repeat period</td>
<td>altitude</td>
</tr>
<tr>
<td>accessibility of celestial sphere</td>
<td>orbital precession</td>
<td>inclination and alt.</td>
</tr>
<tr>
<td>discontinuities in orbit</td>
<td>orbit maintenance frequency</td>
<td>altitude</td>
</tr>
<tr>
<td>mission lifetime</td>
<td>orbital decay</td>
<td>gross altitude</td>
</tr>
<tr>
<td>instrument spatial resolution / radar</td>
<td>-</td>
<td>gross altitude</td>
</tr>
<tr>
<td>transmitter power</td>
<td>-</td>
<td></td>
</tr>
<tr>
<td>radar pulse repetition frequency (PRF)</td>
<td>-</td>
<td>altitude range</td>
</tr>
<tr>
<td>permanent cold radiator surfaces</td>
<td>sun-synchronism</td>
<td>inclination and altitude</td>
</tr>
</tbody>
</table>

As it can be deduced from above table, fundamental consideration is about the type and altitude of the orbit. Selecting the orbit type as sun-synchronous, altitude can be decided with respect to the required period, and therefore frequency, of the common earth observation missions. For this purpose an approach is needed for identifying the common needs for distinct earth observation mission payloads rather than focusing on specific mission module requirements in order to keep flexibility enhanced. This approach may lead to a perception that the overall system may not provide optimum performance means for a particular mission module. However as long as the particular earth observation mission module is designed with respect to the provided infrastructure, it indeed offers advantages since there will be no conflicting requirement and/or configuration conflicts between distinct mission modules due to the payload isolation. Below table 2.2 gives examples of some of the advanced earth observation satellites orbits with respect to their spatial resolution categories [2.2]:

### Table 2.2 Earth observation satellites orbits with respect to their spatial resolution categories

<table>
<thead>
<tr>
<th>Type</th>
<th>Satellite</th>
<th>Orbit (h: altitude, i: inclination, T: period)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Low Resolution</td>
<td>ADEOS (JAXA)</td>
<td>h: 800 km, i: 98.6°, T: 101 min., Repeat Cycle: 41 days</td>
</tr>
<tr>
<td>Medium Resolution</td>
<td>Envisat (ESA)</td>
<td>h: 800 km, i: 98.6°, T: 100.6 min., Repeat Cycle: 35 days</td>
</tr>
<tr>
<td>High Resolution</td>
<td>Landsat (NASA)</td>
<td>h: 705 km, i: 98.2°, T: 99 min., Repeat Cycle: 16 days</td>
</tr>
<tr>
<td>Very High Resolution</td>
<td>Spot 5 (CNES)</td>
<td>H: 832 km, i: 98.7°, T: 101 min., Repeat Cycle: 26 days</td>
</tr>
</tbody>
</table>
Apart from above examples, if we consider the other missions conducted for each type of mission, we can see that the typical altitude range is from 700 to 900 km. The exception to this range may be the orbits of the very high resolution satellites. Although majority of this type missions are operated at an altitude range of 400 to 500 km., there are still satellites, such as Spot 5 and IKONOS, operating within the above mentioned range. Therefore we can consider the altitude to be around 800 km. initially.

If we consider the orbit requirements for cluster flying, we can focus on four main points. These are intersatellite communication, wireless power transmission, collision avoidance and propellant consumption. First three are more related to system design and has little or no effect on the gross altitude and inclination. However, selecting a naturally stable orbit for cluster flying to reduce the propellant consumption is related to the altitude and inclination of the reference module within a cluster. Therefore corresponding relations shall be considered within the determination of the orbit characteristics.

2.3.1. Orbit Design

In reality orbit design has two parts. One is the absolute orbits of the payload modules with respect to Earth and the relative orbits, or formation flying, with respect to a leader satellite. In this section the absolute orbit design will be considered.

- **Orbital Perturbations:**

  There are several considerations to start with a sun-synchronous orbit design as in the case of ENVISAT. However we can start by considering the perturbations on the orbital elements due to various sources. Perturbations can appear in the form of secular (linear), short period (shorter than orbital period) and long period (longer than orbital period) variations. These can be caused by third bodies, such as Sun and Moon, gravity gradients (due to non-spherical Earth), atmospheric drag and solar radiation pressure. Selecting a LEO orbit with an 800 km of altitude, the perturbations due to atmospheric drag and solar radiation pressure are assumed to be not significant [4.5]. However below this altitude atmospheric drag will be more dominant with respect to solar pressure and vice versa. To maintain a more lifetime it is also important to accommodate a space system into a higher altitude. If we consider that the altitude is still within the range of LEO, the perturbations due to third body can also be neglected. Therefore the only important variations to be considered are the secular drifts due to the gravity gradients caused by the non-spherical Earth.

- **Sun Synchronous Repeating Orbit Design:**

  Sun synchronous orbits have unique coupling between the orbit altitude and inclination as well as providing constant sun viewing angle. The altitude can be selected to give an integer number of revolutions in an integer number of Earth days for ground-track repetitions. In addition sun lighting conditions on the orbit and for observations can be determined by selecting the mean local time of the node crossings. In this manner, an algorithm can be implemented to determine the orbit altitude and inclination by selecting the number of revolution of the satellite and the Earth days for orbit repetition. Such an algorithm is introduced by [2.4]. It is based on the rate of change of the secular drifts due to non-spherical Earth, or namely J2 perturbations. Within the algorithm, the secular motions of ascending node and the argument of perigee together with the mean anomaly are iterated to find the corresponding altitude and the inclination for a quasi-circular sun-synchronous orbit. The flowchart of the algorithm can be seen in the appendices.

In the case of cluster flying, we can determine the orbit of reference module, or the leader, by using this algorithm. Considering that the infrastructure is designed for various earth observation mission modules, we can determine the orbit repeat cycle as 30 days which will be convenient for providing monthly data from the instruments. If the ground track separation between the sequential orbits is
desired to be aligned finely, the total number of orbits completed within this repetition cycle can be chosen as 428. Resulting sun-synchronous orbit characteristics can be provided in below table 2.3

Table 2.3 Orbit characteristics of the reference point for cluster

<table>
<thead>
<tr>
<th>Sun-Synchronous Sub-Recurent Orbit Characteristics of the Reference Point</th>
</tr>
</thead>
<tbody>
<tr>
<td>Altitude [km]:</td>
</tr>
<tr>
<td>Inclination [deg]:</td>
</tr>
<tr>
<td>Period [min]:</td>
</tr>
<tr>
<td>Recurrent Period [days]:</td>
</tr>
</tbody>
</table>

2.4. Mission Modules

The remote sensing instruments are characterized by their wavelength bands, spatial resolution, coverage area and the temporal coverage. Each characteristic has implications both on the orbit and the system design. For instance, high resolution observation is a very important characteristic for land observations and they require high accuracy pointing. However atmospheric and oceanic missions are characterized by the multi-channel observations and therefore they can be considered within medium resolution observations. On the other hand, instruments for thermal infrared observations may require significant cooling. Hence each instrument and payload module has particular requirements to perform its mission properly.

As it is mentioned previously, it is of particular interest to achieve the capability of supporting many distinct mission modules with the proposed infrastructure. Here a mission module can be defined as a module which accommodates one or more specific instrument/s for a specific mission, such as land, water or atmospheric observation mission. For defining our mission modules, the latest two JAXA missions, Daichi - Advanced Land Observing Satellite (ALOS) and Ibuki – Greenhouse Gases Observing Satellite (GOSAT), can be very good references [2.1]. First is a land observation mission for which three instruments are accommodated on board. This mission concerns cartography, regional observation, disaster monitoring and resource surveying. Second is aimed for providing data to ascertain the distributions of carbon dioxide (CO₂) and methane (CH₄) and the emission/absorption characteristics of greenhouse gases. For this purpose, it makes use of an instrument which is composed of two sensors. Below the two mission modules can be shown together with their specific instruments:

<table>
<thead>
<tr>
<th>Land Observation Mission Module</th>
</tr>
</thead>
<tbody>
<tr>
<td>Panchromatic Remote Sensing Instrument for Stereo Mapping (PRISM)</td>
</tr>
<tr>
<td>Advanced Visible and Near Infrared Radiometer type 2 (AVNIR-2)</td>
</tr>
<tr>
<td>Phased Array type L-band Synthetic Aperture Radar (PALSAR)</td>
</tr>
</tbody>
</table>
Mission scenario can be defined more specifically with respect to these two mission modules. For instance, the instruments within the modules can be launched to and supported by the infrastructure independently. Therefore it can be shown that instead of designing, developing and launching two satellites, two mission modules can be developed and launched to the existing infrastructure. In addition, one of the main arguments of fractionation is to provide flexibility in terms of instrument development. For example an instrument can be launched without waiting for the development of other instruments and fractionation would also avoid the requirements and/or configuration conflicts between the distinct instruments.

In terms of mission design and operations, fractionated architectures would also increase the flexibility. For instance, considering that the JAXA’s planned mission ALOS-2, by which it is aimed to improve the capabilities in terms of disaster monitoring, land and climate observations, the importance of such a flexibility provided by fractionated architecture is evident. Instead of designing a new spacecraft, a new payload module can be launched in addition to the flying cluster with the current infrastructure. Therefore many launch and mission operations alternatives can be considered in this way.

In conclusion, to make it easier for assessing the requirements for the system design, we can consider above mentioned two set of instruments/modules to be supported by the proposed infrastructure. The characteristics of each instrument can be found in [2.1] and appendices.

2.4. Launch Vehicle

Launch vehicle is a fundamental part of a mission design which affects the structural design of the spacecraft, as well as operations and the cost. As it is mentioned on [1.1], that responsive launch vehicles are very important within the lifecycle of a fractionated spacecraft. As a good example to responsive launch vehicle, and with a low cost per kg, SpaceX Falcon 1e is chosen as a baseline.

The two burn performance to LEO for Falcon 1 and Falcon 1e can be seen in below figure.

![Fig 2.3 LEO performance of Falcon 1 (dashed line) and Falcon 1e (solid line) launch vehicles [2.6]](image-url)
As we can see from above figure, a payload mass around 550 kg can be put into 100° inclination and 700 km of altitude. This could be taken as a maximum limit when evaluating the sizes of the modules.

The payload compartment of Falcon 1e has a volume of approximately 5.5 m³ and it can be seen on right.

Finally the other specifications of this launch vehicle are provided below.

<table>
<thead>
<tr>
<th>Table 2.4 Falcon 1e Mission Accuracy and Fundamental Frequencies</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Falcon 1e Specifications</strong></td>
</tr>
<tr>
<td><strong>Inclination</strong></td>
</tr>
<tr>
<td><strong>Altitude</strong></td>
</tr>
<tr>
<td><strong>Lateral Frequency</strong></td>
</tr>
<tr>
<td><strong>Axial Frequency</strong></td>
</tr>
</tbody>
</table>

2.5. Uncertainty Analysis

Uncertainty can be defined as imprecisely known or unknown various risks and opportunities which can appear during the development and operational lifecycle of a space system [2.5]. In our case, a risk can be defined as an uncertainty that would cause degradation in the system functionality or capability. On the other hand, opportunity can be defined as an uncertainty that would enhance the functionality or capability of a space system.

Several uncertainties within a spacecraft lifecycle were discussed by [2.5, 1.4, 1.2] previously and these are summarized below:

<table>
<thead>
<tr>
<th>Table 2.5 Lifecycle Uncertainties Leading to Risks and Opportunities</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Category</strong></td>
</tr>
<tr>
<td>Technical</td>
</tr>
<tr>
<td>Environmental</td>
</tr>
<tr>
<td>Operational</td>
</tr>
<tr>
<td>Launch Vehicle</td>
</tr>
<tr>
<td>Demand</td>
</tr>
<tr>
<td>Requirements</td>
</tr>
<tr>
<td>Funding</td>
</tr>
<tr>
<td>Programmatic</td>
</tr>
<tr>
<td>Security</td>
</tr>
</tbody>
</table>
To make these uncertainties easier to understand, some examples can be given with respect to the above mentioned categories as:

### Table 2.6 Lifecycle uncertainty examples

<table>
<thead>
<tr>
<th>Category</th>
<th>Risk Example</th>
<th>Opportunity Example</th>
</tr>
</thead>
<tbody>
<tr>
<td>Technical</td>
<td>Communication system failure</td>
<td>Higher capability processor development</td>
</tr>
<tr>
<td>Environmental</td>
<td>Debris Impact / Solar Flare</td>
<td></td>
</tr>
<tr>
<td>Operational</td>
<td>Unsuccessful Deployment</td>
<td>Module addition</td>
</tr>
<tr>
<td>Launch Vehicle</td>
<td>Launch Failure</td>
<td>Small/responsive launch vehicles</td>
</tr>
<tr>
<td>Demand</td>
<td>Reduction in demand for S/C value</td>
<td>Increase in demand for S/C value</td>
</tr>
<tr>
<td>Requirements</td>
<td>Different thermal requirements</td>
<td>New requirement for a future value</td>
</tr>
<tr>
<td>Funding</td>
<td>Decrease in funding</td>
<td>Increase in funding</td>
</tr>
<tr>
<td>Programmatic</td>
<td>Delays in manufacture line</td>
<td>Combination of Earth Obs. Missions</td>
</tr>
<tr>
<td>Security</td>
<td>Anti satellite attack</td>
<td>Use of virtual private network</td>
</tr>
</tbody>
</table>

Traditional perspective for spacecraft design is meeting certain requirements while minimizing the cost. This approach in turn reduces the designer’s freedom to take actions to increase the capability, therefore the delivered value of the spacecraft. However, in order to achieve more functionality, robustness and responsiveness with a space system, the designer must consider as much uncertainty as possible instead of a requirement driven design [1.4]. This can also increase the capability and therefore the value offered by the system. However, one should note that the mitigations of risks and/or exploitations of opportunities must be justified by the attributes, or values, that they offer.
CHAPTER 3

Cluster Flying

“I jumped in the river and what did I see?
Black-eyed angels swam with me
A moon full of stars and astral cars
And all the figures I used to see

There was nothing to fear and nothing to doubt”

Pyramid Song, Amnesiac, Radiohead


3 Cluster Flying

Fractionating a spacecraft and letting the different subsystems fly separately leads to the formation flying design of a cluster. Slightly different from the common term formation flying, cluster flying which is defined here requires less control effort but a collision free relative orbit design for a multiple satellite configuration. In addition to the safety, propellant consumption and line of sight for inter-satellite link establishment are other considerations for a cluster flying orbit design in the case of fractionated spacecraft (Table 3.1). Reconfigurability is another aspect which induces formation control and manoeuvrability. However this has less importance as it is less frequent and vital when compared to orbit maintenance.

<table>
<thead>
<tr>
<th>Aspect</th>
<th>Implication</th>
<th>Action on Implication</th>
</tr>
</thead>
<tbody>
<tr>
<td>Safety</td>
<td>Distance</td>
<td>Increase</td>
</tr>
<tr>
<td>Inter-module Link</td>
<td>Visibility, Distance</td>
<td>Ensure Visibility, Decrease Distance</td>
</tr>
<tr>
<td>Propellant Consumption</td>
<td>Maintenance Manoeuvre</td>
<td>Decrease</td>
</tr>
<tr>
<td>Reconfigurability</td>
<td>Formation Control</td>
<td>Increase</td>
</tr>
</tbody>
</table>

Previously an exploitation of the eccentricity/inclination vector separation method was provided for a close proximity low earth orbit (LEO) formation flying design [3.1]. In addition, the conditions for safety were clearly identified in the same paper and in a case study performed by the same authors in [3.2]. As the method of parallel separation of eccentricity and inclination vectors also provides decoupling between the in-plane and cross track plane motion, it provides an easy understanding in terms of the visualization and evaluation of a cluster flying design.

Using this method for a collision free cluster flying design, several configurations can be identified based on their geometrical properties. In this part, these configurations are evaluated with respect to the propellant consumption efforts for orbit maintenance and constraints due to safety and visibility to find the best cluster flying configuration alternative in the case of fractionated spacecraft.

3.1 Cluster Flying Design and Optimization

In this section, the methods for designing and optimization of a close proximity LEO cluster flying are presented. Initially the parallel separation of eccentricity and inclination vectors method will be introduced. Then the optimization of the specified configuration will be explained together with the involved assumptions. The nomenclatures are provided in the beginning of the thesis.

3.1.1 Parallel Separation of Eccentricity and Inclination Vectors

This method was previously exploited in the references [3.1] and [3.2] for a two spacecraft formation flying in LEO. In this part, used equations are summarized and the necessary explanations are provided.

3.1.1.1 Relative Motion

As derived in [3.1], it is possible to define the relative motion at any mean argument of latitude, \( u \), in terms of the relative orbital elements, \( \Delta a, \Delta e, \Delta i, \) and \( \Delta u \), by using the linearized equations of relative motion at an epoch. The positions for a given argument of latitude can be provided by the equations (3.1), (3.2) and (3.3) for radial (R), along track (T), and cross track (N) directions.
\[ \Delta r_R = a\Delta a - a\delta e \cos(u - \varphi) \]  
\[ \Delta r_T = a\Delta u - \frac{3}{2}(u - u_0) a\Delta a + 2a\delta e \sin(u - \varphi) \]  
\[ \Delta r_N = a\delta i \sin(u - \theta) \]  
With the assumption of close proximity operations where the altitude and argument of perigee differences are small, i.e. \( \Delta a \approx 0 \) and \( \Delta u = 0 \), these equations can be re-written as (3.4), (3.5) and (3.6)

\[ \Delta r_R = a\delta e \cos(u - \varphi) \]  
\[ \Delta r_T = 2a\delta e \sin(u - \varphi) \]  
\[ \Delta r_N = a\delta i \sin(u - \theta) \]  
Resulting relative orbit is a three dimensional ellipse where the in plane motion is described by relative eccentricity vector, \( \delta e \), and relative perigee, \( \varphi \), and the motion perpendicular to orbital plane is described by the relative inclination vector, \( \delta i \), and relative ascending node, \( \theta \). The clear visualization for two spacecrafts relative motion is provided in Figure 3.1.

![Fig 3.1](image1)

**Fig 3.1** In plane (along track) and cross track plane relative motion of two spacecrafts described by relative eccentricity and inclination vectors (figure ref [3.1]).

### 3.1.1.2 Collision Avoidance

One second of along track navigation uncertainty approximately results in 7.5 kilometres in along track direction for LEO. Therefore a spacecraft must avoid crossing path of any other spacecraft in the along-track direction in the case of close proximity operations. In this manner, the basic assumption for collision avoidance is ensuring that there will be always a separation along the cross track direction and/or radial direction between any two spacecrafts. As described in [3.1], this condition can be achieved by aligning the relative eccentricity and inclination vectors parallel to each other. This is simply obtained by setting \( \varphi = \theta \). For orthogonal eccentricity and inclination vectors, the radial and cross track distances can vanish together leaving a separation only on along track direction. This is what to be avoided for a collision free relative orbit design. These two conditions are depicted in below figure

![Fig 3.2](image2)

**Fig 3.2** Depiction of relative orbits corresponding to parallel and orthogonal relative eccentricity and inclination vectors (figure ref [3.1]).
3.1.1.3 Relative Orbit Maintenance

For close proximity cluster flying relative motion, the long periodic and secular perturbations are the main sources of the changes in relative eccentricity and inclination vectors. The major forces that disturb the relative motion in LEO are mainly due to the non-spherical Earth and differential drag. It is assumed that the effect of differential drag would be much lower, especially for higher altitudes around 800 km., so the perturbations only due to non-spherical Earth (due to J₂) are considered for the relative orbit maintenance.

The J₂ perturbations cause shifts in the directions of relative eccentricity and inclination vectors as depicted in Fig. 3.3

![Fig 3.3 Evolution of the relative eccentricity and inclination vectors due to J₂ perturbations for nearly circular orbits (figure ref [3.1])](image)

The equations describing these shifts can be written as follows

\[ \phi = \frac{2\pi}{T_e} = \left( \frac{3}{2} \frac{\pi}{T} \frac{R_e^2}{a^2} J_2 (5 \cos^2 i - 1) \right) \]  
\[ \frac{d}{dt} \Delta i = -\frac{3\pi}{T} \frac{R_e^2}{a^2} J_2 \sin^2 i \Delta i \]   

Where the term \( \Delta i \) can also be written as

\[ \Delta i = \begin{vmatrix} \Delta i_x \\ \Delta i_y \end{vmatrix} = \sin(\delta i) \begin{vmatrix} \cos(\theta) \\ \sin(\theta) \end{vmatrix} \approx \begin{vmatrix} \Delta i_x \\ \Delta i_y \end{vmatrix} \]  

The relative orbit control equations to account for these shifts are provided for both along track and cross track directions as:

\[ |\Delta V_T| = \frac{v}{2} ||De|| \]  
\[ |\Delta V_N| = v ||Di|| \]   

Where \( V = 2\pi a / T \), \( ||De|| = |\delta e \phi| \) and \( ||Di|| = |\frac{d}{dt} \Delta i| \)

Therefore the final control equations in the along-track and cross-track directions can be rewritten by using the above relations from 3.7 to 3.11 as:

\[ |\Delta V_T| = \frac{\pi}{T_s} \left( \frac{3}{2} \frac{\pi}{T_d} \frac{R_e^2}{a^2} J_2 (5 \cos^2 i - 1) \right) a \delta e \]  
\[ |\Delta V_N| = \frac{2\pi}{T_s} \left( \frac{3\pi}{T_d} \frac{R_e^2}{a^2} J_2 \sin^2 i \right) a \delta i \cos(\theta) \]  

Where \( T_s \) is the orbital period in seconds and \( T_d \) is the orbital period in days.
3.1.2 Optimization

As it can be seen from the relative position and relative orbit control equations, it is possible to design relative orbits by just defining \( a_\delta e, a_\delta i \) and phase angle \( \phi = \theta \). Therefore with respect to these design variables, several configurations can be formed and the propellant consumption due to the orbit maintenance manoeuvres can be evaluated.

Since the propellant consumption is directly proportional to the \( a_\delta e, a_\delta i \) and \( \cos \theta \), a simple cost function, \( \Phi \), can be written for the purpose of evaluation as

\[
\Phi = \sum \Phi_i \text{ where } \Phi_i = C_1 (R_i)^2 + C_2 (\cos (\theta_i))^2 \text{ and } i = 1, ..., N \quad (3.14)
\]

Here \( R_i \) can be either the radius of the relative circular orbit projected on the cross track plane for \( a_\delta e = a_\delta i = R \) or the arithmetic mean of \( a_\delta e \) and \( a_\delta i \). The coefficients \( C_1 \) and \( C_2 \) can be written as \( 1/R_{\text{max}} \) and 1 respectively and \( N \) is the number of spacecrafts. \( R_{\text{max}} \) is a normalization factor which is introduced to make the effects of change in the variables comparable. This factor is defined properly after a few runs of the optimization code with respect to the collision avoidance distance and the dimension of the cluster.

The constraints are defined by the inter-satellite distances and the visibility conditions. A minimum distance between any two satellites is specified with respect to safety considerations and maximum distance is specified with respect to the range of wireless power transfer, relation (3.15). The visibility condition is defined such that any three points which are on the same plane shall not be on the same line, i.e. not collinear. This condition is satisfied for the cluster flying if the points \( P_3, P_2 \) and \( P_1 \) are defined as the projected positions of any three spacecrafts on the cross track plane which are expressed as imaginary numbers, i.e. \( P_i = \Delta r_{R,i} + i \Delta r_{T,i} \), and if the relation (3.16) is true [3.3].

\[
\Delta d_{\text{max}} \geq (\Delta r_{R,j} - \Delta r_{R,j})^2 + (\Delta r_{T,j} - \Delta r_{T,j})^2 + (\Delta r_{N,j} - \Delta r_{N,j})^2 \geq \Delta d_{\text{min}} \quad (3.15)
\]

\[
(P_1 - P_2) / (P_3 - P_2) \in \text{Complex} \quad (3.16)
\]

For the optimization, evolutionary algorithms, namely genetic algorithms, were used as they are efficient to cope with the nonlinear constraint functions. The simulations were made by using the MATLAB Optimization Toolbox for eight spacecrafts. The number of spacecrafts was chosen such that a notional fractionated spacecraft architecture can be represented with three infrastructure modules, such as navigation, communication & data handling and power, and five payload, or mission, modules. In addition to the number of spacecrafts, the altitude was chosen as 800 km to represent a big earth observation satellite orbit, such as ENVISAT. The minimum and maximum distance constraints were chosen as 100 m. and 1 km. respectively and the distances between every two spacecraft were calculated for each instant, or degree, of argument of latitude. Finally, after several runs of the optimization the normalization factor \( R_{\text{max}} \) was specified as 200 m..

3.2 Configuration Alternatives and Optimization Results

The configurations which are discussed in this section are based on the considerations mentioned in Table 3.1 and the representing design variables mentioned in the section 3.1.2. For simplicity, the analysis can be done firstly by assuming \( a_\delta e = a_\delta i = R \). This will provide a circular relative orbit (projected) on the cross track plane. Then design variables will become the radius \( R \) and the phase
angle $\theta$ for each satellite. In this manner, three different configurations can be identified as depicted in Fig 3.4.

![Fig 3.4 Three different types of configurations: line, circular and combination](image)

In the first configuration the spacecrafts are distributed to relative circular orbits with different radii but with same phase angle. As the phase angles are same for all, the spacecrafts move as a line along the corresponding three dimensional elliptical relative orbits. Line configuration is limited in terms of visibility and scalability of the spacecraft network, i.e. addition of new modules. If the satellite is likely to grow with additional modules then the distance between the innermost and outermost modules can exceed the maximum limit. In the second configuration, the spacecrafts are distributed on the same relative orbit but with different phase angles. Therefore this configuration ensures the visibility between the spacecrafts, or modules. With increasing number of modules, the radius of the projected relative circular orbit would grow more and this could introduce propellant consumption inefficiency. Also safety should be considered in detail for this configuration. In the third configuration, both radii of the relative orbits and the phase angles of the spacecrafts are different. Therefore this configuration has the advantages of both configurations and at the same time eliminates the limitations of scalability and visibility. The design variables and the considerations for each configuration can be summarized in Table 3.2.

**Table 3.2 Cluster Flying Configurations and Initial Considerations (i=1,…, N=8)**

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Design Variables</th>
<th>Considerations</th>
</tr>
</thead>
<tbody>
<tr>
<td>Line</td>
<td>$R_i$ and $\theta (=\theta_i)$</td>
<td>Ensures safety, limited in terms of visibility and scalability</td>
</tr>
<tr>
<td>Circle</td>
<td>$R (= R_i)$ and $\theta_i$</td>
<td>Ensures visibility, scaling may increase fuel consumption</td>
</tr>
<tr>
<td>Hybrid</td>
<td>$R_i$ and $\theta_i$</td>
<td>Eliminates the limitations</td>
</tr>
</tbody>
</table>

Several runs were made and the results for each configuration are best summarized in below Table 3.3 and visualized in figures 3.5, 3.6 and 3.7:

**Table 3.3 Initial cluster flying design optimization results**

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Radius [m]</th>
<th>Phase</th>
<th>Max &amp; Min Distances [m]</th>
<th>Cost Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Line</td>
<td>$R_i=0,100,\ldots,700$</td>
<td>$\theta_i = \pi/2$</td>
<td>$\Delta d_{\min} = 100$ &amp; $\Delta d_{\max} = 1565$</td>
<td>35</td>
</tr>
<tr>
<td>Circle</td>
<td>$R = 140$</td>
<td>Fig. 3.6</td>
<td>$\Delta d_{\min} = 100$ &amp; $\Delta d_{\max} = 620$</td>
<td>6.1</td>
</tr>
<tr>
<td>Hybrid</td>
<td>$R_i=0-100-190$</td>
<td>Fig. 3.7</td>
<td>$\Delta d_{\min} = 100$ &amp; $\Delta d_{\max} = 815$</td>
<td>4.87</td>
</tr>
</tbody>
</table>
Fig 3.5 Cross track plane positions at $u = \pi$ and the corresponding relative orbits for line configuration

Fig 3.6 Cross track plane positions at $u = \pi$ and the corresponding relative orbits for circle configuration

Fig 3.7 Cross track plane positions at $u = \pi$ and the corresponding relative orbits for hybrid, or compromise, configuration
For a final assessment, it was assumed that $a_\delta e \neq a_\delta i$ and more design variables were introduced for a fractionated satellite network of $N=8$. Since the feasibility of line configuration was already assessed, optimization was performed only considering the ellipse and hybrid configurations. The results of the final simulations are summarized in Table 3.4 and visualized in Figures 3.8 and 3.9:

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Radius [m]</th>
<th>Angle</th>
<th>Max &amp; Min Distances [m]</th>
<th>Cost Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ellipse</td>
<td>$a_\delta e = 185$</td>
<td>Fig. 3.8</td>
<td>$\Delta d_{\text{min}} = 100$ &amp; $\Delta d_{\text{max}} = 784$</td>
<td>5.45</td>
</tr>
<tr>
<td></td>
<td>$a_\delta i = 100$</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Hybrid</td>
<td>$a_\delta e = 30 \cdots 130 \cdots 160 \cdots 190$</td>
<td>Fig. 3.9</td>
<td>$\Delta d_{\text{min}} = 100$ &amp; $\Delta d_{\text{max}} = 741$</td>
<td>4.5</td>
</tr>
<tr>
<td></td>
<td>$a_\delta i = 3 \cdots 50 \cdots 60 \cdots 75$</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

As it can be seen from the Tables 3.3 and 3.4, an optimum solution in terms of propellant consumption, safety and visibility can be obtained for the circular and hybrid configurations. Scaling the cluster for greater module numbers effects the feasibility of the line configuration significantly.
since the maximum distance constraint is violated and the propellant consumption strongly depends on the distance from the reference point.

From figures 3.5 to 3.9, the trend for minimizing the radius and \( \cos(\theta) \) can be observed. The phase angles converge towards the vicinity of \( \pm \pi/2 \) in parallel to a reduction in the size of relative orbits.

Comparing the results which are presented in Tables 3.3 and 3.4, it is observed that there is an improvement in the cost values for elliptical cross-track plane relative orbit approaches when compared to circular counterparts. With respect to these cost function results, most optimum configuration becomes the hybrid configuration for the same constraints and objective function.

Based on the above discussions, without the presence of a significant navigation uncertainty for the motion perpendicular to along-track direction it is concluded that a cluster flying design which ensures collision free relative motion is possible for a network of fractionated spacecraft. Even though a passive collision avoidance approach is an objective, there is still need for manoeuvring capability in terms of close proximity cluster flying to correct the relative eccentricity and inclination vector shifts caused by the \( J_2 \) perturbations as discussed previously.

Finally, for the ease of communication and navigation during the normal operations, the configuration which is depicted on figure 3.7 is favourable since the relative orbits are coplanar. However, the configuration may still be changed to one of previously mentioned alternatives with respect to different operational modes.
CHAPTER 4

Spacecraft Architecture

“You can try the best you can
If you try the best you can
The best you can is good enough”

Optimistic, Kid A, Radiohead
4. Spacecraft Architecture

This chapter is divided into two sections: system analysis and notional spacecraft architecture. Various technologies are studied and the realization strategies are discussed within the system analysis section. Later notional fractionated spacecraft architecture is proposed for an earth observation cluster based on the system analysis.

4.1 System Analysis

The fractionated spacecraft architecture is composed of two types of modules. These are infrastructure and mission, or payload, modules. Since the key to the fractionated spacecraft is resource sharing, the modules with the ability of producing more resources are considered to be resource, or infrastructure, modules.

In this section the shared resources, related technologies and realization of the corresponding subsystems will be analysed. The infrastructure modules which have the ability to share the resources together with the corresponding key technologies can be identified as follows:

- shared resources → infrastructure modules
  - guidance, navigation & control (GNC)
  - communication
  - power
  - command and data handling

- key technolgy and strategy analysis to realize the concept
  - cluster flying
  - autonomous cluster navigation
  - inter-satellite communication and optical links
  - wireless power transfer

Since the cluster flying analysis is provided in Chapter 3, the rest of the shared resources, technologies and realization strategies will be discussed one by one.

4.1.1. Guidance, Navigation and Control (GNC) System

Guidance, navigation and control can be referred as the determination and control of the orbit and orientation of a spacecraft. If more than one space system depend on each other and fly together then the relative orbit and orientations must also be considered. In this manner, a general strategy for the management of above tasks should be defined initially by considering the basic requirements of a cluster flying with respect to the fractionated spacecraft architectures.

In the case of formation flying, the satellites need to be controlled precisely to keep a specific configuration. However cluster flying does not require a strict relative position control as long as collision avoidance is ensured. This can reduce the frequency of the control actions for the cluster. Though minimizing the propellant consumption is an indispensable goal, the propellant consumption does not have to be the same for all modules as the lifetime and operational time interval for each module can differ from the others. However the propellant percentage of a single module can still be considered while assigning the manoeuvres to increase the lifetime of its functionality.

A fractionated architecture can be formed by either launching the modules incrementally or together as a bunch to the desired initial orbital positions. However in order for the acquisition, in the case of
initialization or reconfiguration, and maintenance of the desired orbital positions within a cluster, orbit determination and control should be considered as a minimum requirement for each module. Hence orbit control induces attitude control requirements to direct the thrusters for performing the required manoeuvres.

In addition to the absolute orbit management, relative navigation issues must be considered while defining the GNC strategy. In this manner, the functions for guidance, navigation and control management can be defined. A flight management system of a cluster requires monitoring, planning, commanding and fault detection (FD) functions [4.1].

**Monitoring:**

The aim of this function is to collect information from all members of the cluster and provide past, present and future of the absolute and relative motion of individual modules and the cluster as a whole.

**Planning:**

Basically, planning function will be responsible for the determination of desired orbital positions in the case of maintenance and reconfiguration of the cluster. Therefore the related tasks are generating the required trajectories in case of nominal operations such as orbit corrections, and orientations for an inter-module link establishment and in case of contingency operations such as collision avoidance, module addition, failure or another uncertainty that causes the cluster to change its nominal operation to a relevant special mode.

**Commanding:**

Duty of this function is distributing the required manoeuvres, or control actions, through the cluster by evaluating the trajectory outputs of the planning function.

**Fault Detection (FD):**

Basic tasks of this function will be the detection and identification of the possible risks, which are the uncertainties that will cause degradation and/or loss in the functionality of the system such as collision risks, command or manoeuvre failure and so on. Therefore risk identification will be performed both at module and cluster level. These risks should be notified to the planning function and the necessary recovery actions can be determined either in collaboration with the planning function and/or in some cases with ground.

The overview of the flight management with respect to these functions can be given as follows:

![Functional Block Diagram of the Flight Management](image)

As it can be deduced from the above functional block diagram, outputs of the monitoring function will be provided to both FD and planning. Then the required actions are decided within the planning function. These required actions are then forwarded to the command functions which is responsible for the distribution of those actions to the corresponding members of the cluster.
4.1.1.1 Cluster Flying Management

In order to perform autonomous operations, above functions have to be performed with some degree of autonomy. Here the question is how to distribute and realize this autonomy with respect to the mission specific and basic requirements.

Basically there are two extremes for distribution of flight management through the cluster [4.2, 4.3]. These are centralised and decentralised GNC. For centralised flight management, one module, which could be named as GNC module, performs all previously mentioned GNC functions itself. If the commands are not decided on-board then it relays the commands which are sent from ground. However the GNC functionality relies highly on the existence of this GNC module. On the other hand fully decentralised GNC implies that all this management is distributed to the modules within the cluster. This could improve the reliability via system redundancy. However, taking a reconfiguration or safety decision could be complicated in case of an abnormal event during the mission lifetime. In addition, decentralised GNC would require high inter-module communication abilities and significant complexity for the individual module GNC system. A compromise between these two extremes is also possible. For instance, the computational effort can be distributed over the cluster and a leader can be assigned to take decisions and distribute the tasks to the rest of the cluster.

In our case, it is desired to perform most of the basic functions by the infrastructure (here it is GNC module). Therefore this approach favours a centralized flight management for the mission modules. Since the total number of modules would be less than 10, the centralised approach can be realized. If the network is likely to grow then the cluster should be divided into teams which are managed in a centralised manner by a captain [4.2]. Here a GNC module can be designed such that it performs all the functions of autonomous cluster navigation and becomes the reference point for the others. The advantage of a centralized system is that the inter-satellite communications and computational load for each module, except GNC module, in the cluster would be minimized in terms of flight management. However the main disadvantage of the centralized system is that it is prone to single point failure. Therefore redundancy within the infrastructure or contingency operations (with respect to the number and configuration of the cluster) can be considered in the case of GNC module loss.

In the case of nominal operations, such as orbit maintenance, the frequency of the orbit control is much less than the attitude control for each module. Therefore only the orbit control will be centralised and attitude control will be the responsibility of the individual modules. However the monitoring function will still be providing the relative attitudes between the modules. Therefore in case of an operation which requires an inter-module link, the GNC module will be responsible for commanding the desired absolute attitudes for each module at particular time instants to establish the corresponding link.

This centralised architecture can be shown as follows:
In the above architecture all modules are in contact with the GNC module and they provide both navigation and risk information (such as hardware loss) to the monitoring and FD functions. In addition to the modules, ground is also included as another node within the whole system. With the above flight management architecture, ground will be able to monitor the cluster and uplink either desired positions and/or trajectories for the cluster or directly the commands to a specific module.

**Monitoring Function**

It will collect absolute orbital motion and attitude data from each module, store them and perform coordinate transformations to the relative motion frame. Hence it will be able to provide past and present absolute and relative navigation information. If onboard motion propagators are included then it will be also possible to provide future information. Absolute and calculated relative motion data is provided continuously to the planning function for orbit maintenance and FD function for collision or evaporation inspection. Absolute and relative motion data is provided periodically or when needed to the ground. This function also maintains the information of member addition, removal or leader change.

**Planning Function**

It is responsible for the generation, evaluation and assignment of the trajectories to be followed by the modules in order to keep the cluster operating as desired. There can be three reasons to perform these functions. These are undesirable navigation error, assignments from ground and contingency operations such as collision avoidance triggered by FD.

i – *Orbit Maintenance*:

Planning function receives the motion data continuously to identify if the navigation errors are within the tolerated margins. If the errors are greater than the pre defined margins then it performs trajectory generation using the guidance law to reduce this error into reasonable values.

Orbit correction can be either for a single module or for the whole cluster. In case of a single module correction then planning function will generate trajectories for only this module and the corresponding manoeuvre will be commanded. On the other hand orbit correction of a cluster can be

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**Fig 4.2 Functional Block Diagram of the Centralised Flight Management**

![Functional Block Diagram of the Centralised Flight Management](image-url)
performed with some steps. Initially the reference or GNC module will perform the correction then it assigns the corrections to the other modules.

ii – *Ground Interruption:*

If the ground decides to reconfigure the positions within the cluster it can either uplink the positions/trajectories to be followed by the planning function or the commands to be distributed to the modules. An example could be the position assignments from ground in case of a reconfiguration resulting from a module addition.

iii – *Contingency Operations:*

In the case of nominal operations, there may be no need for continuous collision monitoring. In this case, collision avoidance can be considered when there is a reconfiguration or a failure event. However evaporation monitoring is needed continuously by inspecting the inter module distances to make sure a link can be established. When a risk, which requires a safety operation, is identified by FD then it triggers the planning function to generate corresponding trajectories.

Whatever the reason is, once a trajectory is determined with safety concerns, it is sent to the FD function and evaluated if there is a possible risk or not. After this confirmation, the resulting trajectories are sent to the command function. If there is a need for link establishment, then required orientations for each module is calculated and provided to the command function.

**Command Function**

The basic duty of command function would be the generation of manoeuvres and timelines required by the given trajectories from planning function and distribution of those to the corresponding modules. Manoeuvre planning can be performed with respect to the amount of ∆V and manoeuvre duration minimization. In the case of link establishment between the modules or aligning the thrusters in required firing directions for manoeuvres, it will also calculate the most optimum attitude motion and command the absolute attitudes to corresponding modules.

When several trajectories are assigned to the command function, a cost analysis is performed in response to the required ∆V values and manoeuvre duration. These values may be determined differently for each module. Therefore control laws of each module are provided to the commanding function. Then the command function determines the ∆V and manoeuvre time values itself and evaluates the corresponding costs. Finally optimal manoeuvres are commanded as time tagged ∆V values such that minimum possible cost is achieved.

A cost function for each module with respect to a given manoeuvre can be written as follows [4.2]:

\[ C_{ij} = f^{-x} \times \Delta V_{ij} \times \left(\frac{T_i}{T_{min}}\right)^y \]  

(4.1)

- i : is the module number
- j : manoeuvre number
- f : propellant percentage of the module (the amount may be different for each module)
- x : a parameter which indicates the importance of fuel consumption and is greater than 0
- Ti : Time required for the manoeuvre
- Tmin: minimum manoeuvre time defined by the operator
- y: adjustable parameter to define the importance of manoeuvre duration

Here minimum ∆V is obtained by setting y=0 which corresponds to the longest possible manoeuvre duration. In addition, one should note that the desired attitude values are commanded only in the case of orbital manoeuvres and inter-module link establishment.
**Fault Detection**

This function will be capable of propagating the relative motion to identify collision and evaporation risks from a given initial conditions provided by monitoring function and trajectories/desired positions provided by planning function. If the relative motion propagation is performed by monitoring function then it will only perform the risk evaluation for nominal flight. This function will also inspect the housekeeping data provided by the modules if the attitude and orbit control systems are alive and capable of performing control actions. Therefore both at module and cluster level the possible risks would be identified. Once a risk is identified, corresponding operation mode or action will be decided and FD will trigger the planning function to generate the required trajectories.

### 4.1.1.2 Control Modes

At this point some of the cluster flying modes can be defined. The modes can be categorized as individual and cluster (collective) modes.

**Individual Modes**

- **Module Cluster Acquisition Mode (MCAM):** This mode can be active in two cases such as orbit acquisition and reconfiguration. As an example, after the separation from launch vehicle each module performs sun acquisition and orbit control itself to get into desired positions within the cluster. Once the cluster is formed, orbit control is centralised and nominal operations are performed by the GNC Module.

- **Module Pointing Mode (MPM):** As pointing requirements for each module can be different, the control of attitude is decentralised and performed by each module itself. Inertial and target pointing can be included within this mode.

- **Module Stand Alone Mode (MSAM):** A survival mode for the spacecraft until the recovery action is commanded in case of unforeseen events such as link losses for a long period. Constraints are derived from the power and thermal control.

- **Module Link Mode (MLM):** A special individual mode for the establishment of an inter-module link. This is activated by the GNC module regarding Link Mode operation for the corresponding modules.

**Cluster Flying Modes**

- **Cluster Nominal Mode (CNM):** This mode is a coarse formation control mode which only performs relative and absolute orbit maintenance operations. It is managed by the GNC module with a centralised architecture as explained previously and individual pointing modes are active.

- **Cluster Link Mode (CLM):** This mode is only considered when there is a precise attitude control requirement due to an inter-module link establishment. The control is performed only by the corresponding modules therefore not by the whole cluster.

- **Cluster Reconfiguration Mode (CRM):** Module addition/removal, geometry and formation size changes are considered within this mode. It is planned and commanded either by GNC module or ground. Depending on the configuration, it can be active for one or more modules.

- **Cluster Collision and Evaporation Avoidance Mode (CCEAM):** It is managed by GNC mode and is similar to reconfiguration mode.

These modes can be summarized with respect to the corresponding operations in Table 4.1:
Table 4.1 Individual and cluster modes with respect to the operations

<table>
<thead>
<tr>
<th>Operation</th>
<th>Module GNC Mode</th>
<th>Cluster GNC Mode</th>
</tr>
</thead>
<tbody>
<tr>
<td>Acquisition / Reconfiguration</td>
<td>MCAM</td>
<td>CRM</td>
</tr>
<tr>
<td>Safety</td>
<td>MCAM, MSAM</td>
<td>CCEAM</td>
</tr>
<tr>
<td>Nominal</td>
<td>MLM, MPM</td>
<td>CNM, CLM</td>
</tr>
</tbody>
</table>

4.1.1.3 GNC Systems Overview

The overview of the GNC systems and their interactions can be shown in the below figure.

Cluster flight management is already described previously. Module flight management is nothing but the management of the attitude and orbit control in single unit level. Here AOCS hardware is composed of the actuators and sensors. These actuator and sensors must be selected with respect to the functional and mission specific module requirements.
The selection and trade off for the AOCS hardware should be performed with respect to below considerations:

- thrust range and specific impulse
- pointing and measurement accuracy
- mass of the hardware
- propellant requirements
- power requirements
- cost (including development or acquisition and operation)
- associated risk
- mean time between failure (MTBF)
- onboard software and extra hardware requirements
- operational implications
- system level impact of hardware configuration

Attitude control system is composed of a reference, or attitude, a rate sensor and the actuators such as reaction wheels and magnetic torquers. Given a pointing and control frequency requirement, the sensor and actuators can be selected. In this manner, some typical performance intervals of such can be seen in below tables.

### Table 4.2 Possible sensors and actuators with typical performance measures [4.5]

<table>
<thead>
<tr>
<th>Attitude sensor</th>
<th>Rate sensor</th>
<th>Attitude accuracy</th>
<th>Rate accuracy</th>
</tr>
</thead>
<tbody>
<tr>
<td>STT</td>
<td>FOG</td>
<td>1arcsec～100arcsec</td>
<td>1e-5rad/s ～1e-4rad/s</td>
</tr>
<tr>
<td>STT</td>
<td>Inertial gyro</td>
<td>1arcsec～100arcsec</td>
<td>1e-2rad/s ～1e-3rad/s</td>
</tr>
<tr>
<td>Sun Sensor, Magnetometer</td>
<td>Inertial gyro</td>
<td>1deg～10deg</td>
<td>1e-2rad/s ～1e-3rad/s</td>
</tr>
<tr>
<td>Magnetometer</td>
<td>Inertial gyro</td>
<td>10deg</td>
<td>1e-2rad/s ～1e-3rad/s</td>
</tr>
<tr>
<td>Magnetometer</td>
<td>Magnetometer</td>
<td>10deg</td>
<td>1e-2rad/s ～1e-3rad/s</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Actuator</th>
<th>Typical Performance Range</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hot Gas Thrustes</td>
<td>0.5 to 9000 N</td>
</tr>
<tr>
<td>Cold Gas Thrusters</td>
<td>&lt;5N</td>
</tr>
<tr>
<td>Reaction and Momentum Wheels</td>
<td>0.1 to 1 Nm</td>
</tr>
<tr>
<td>Control Moment Gyros</td>
<td>0.01 to 1000 Nm</td>
</tr>
<tr>
<td>Magnetic Torquers</td>
<td>0.01 to 0.1 Nm</td>
</tr>
</tbody>
</table>
In addition, depending on the torque and momentum storage capabilities as well as the operational lifetimes, the reaction wheels can be selected individually for each module. In this manner the specifications of two types of commercial reaction wheels are given as follows [4.9]:

<table>
<thead>
<tr>
<th></th>
<th></th>
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<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>MW1000</td>
<td>1.4</td>
<td>2 to 9</td>
<td>1.1</td>
<td>0.03</td>
<td>±10000</td>
<td>5</td>
</tr>
<tr>
<td>MW4000</td>
<td>2.5</td>
<td>2.5 to 15</td>
<td>4</td>
<td>0.1</td>
<td>±10000</td>
<td>10</td>
</tr>
</tbody>
</table>

For the orbit control, mono propellant thrusters can be accommodated on a spacecraft and we can make use of GPS receivers for the position determination. The specifications of a commercial thruster are given below:

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>1N Mono Propellant Hydrazine Thruster [4.10]</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>0.32 to 1.1</td>
<td>220</td>
<td>0.29</td>
<td>~10</td>
</tr>
</tbody>
</table>

Apart from the attitude and orbit control hardware, we also need to evaluate the possible relative navigation sensors which could be accommodated on a spacecraft within the cluster. At this point regardless of their accuracies, mass and power consumptions, some possible relative navigation sensors are listed for a given measured parameter.

<table>
<thead>
<tr>
<th>Measured Parameter</th>
<th>Alternatives</th>
</tr>
</thead>
<tbody>
<tr>
<td>Absolute Position and Velocity</td>
<td>GPS</td>
</tr>
<tr>
<td>Relative Position and Velocity</td>
<td>CDGPS</td>
</tr>
<tr>
<td>Time</td>
<td>GPS</td>
</tr>
<tr>
<td>Relative Attitude</td>
<td>CDGPS</td>
</tr>
<tr>
<td>Range</td>
<td>Radar (RF)</td>
</tr>
<tr>
<td>Range Rate</td>
<td>Radar (RF)</td>
</tr>
<tr>
<td>Line of Sight Direction</td>
<td>Radar (RF)</td>
</tr>
</tbody>
</table>

\(^1\) Carrier Phase Differential GPS, \(^2\) Cross Link Transceiver, \(^3\) Intelligent Vision Based Relative Navigation
Since it is not the primary requirement to determine the attitudes with a high frequency, it is more important to determine relative positions and velocities. In this manner, accommodating a CDGPS receiver would allow us to determine absolute positions with a frequency of 1 Hz and relative positions with accuracy of ~2-5 cm and 1 cm/s in real time [4.3]. As it will be mentioned while the communication system is described, there will be optical links within the cluster which are using lasers. Therefore this optical system can also be used for ranging and line of sight direction detection if the signal to noise ratio within the link would be sufficient for a good accuracy.

Apart from the relative navigation sensors, an interesting relative navigation actuator system is developed by Space Systems Laboratory (SSL) of Massachusetts Institute of Technology (MIT) [4.6]. This system is mentioned as electromagnetic formation flying. In this system, the magnetic fields are created by sending a current through coils of wire. By controlling the direction of the magnetic dipoles between two spacecraft (or more), attraction, repulsion and sheer forces can be created. If reaction wheels are added to the system, then any manoeuvre can be performed as long as the centre of mass of the formation is kept constant.

This technology is still being developed and there is no demonstration on space until now. Therefore there is no reliable performance data available. Though it minimizes the use of consumables, this concept is more applicable to the free flying multi-spacecraft formations where the centre of mass is not needed to be controlled. Depending on its mass, volume and power requirements, it can be thought for an emergency collision avoidance actuator.

4.1.2 Communications, Command and Data Handling

Traditionally communication subsystem functionality is routing the payload and housekeeping data from spacecraft to ground and commands from ground to spacecraft. The difference to traditional communications architecture is that there is a need for more than one link design due to the distributed architecture. In addition, one step further would be combining the command & data handling system (CDH) with communications subsystem. This is mainly because the communication and CDH systems are closely coupled to each other in terms of collecting and routing the data. CDH and communication modules can still be realized physically separately, however, this may cost for additional high speed links other than payload to communications module communication. The combined Comm. & CDH system can form a point to point high speed bi-directional architecture which can be reconfigured with respect to payload and infrastructure module addition or removals. This strategy is based on so called SpaceWire [4.11] data handling architecture which is a standard for high speed links and networks for easing the intercommunications between sensors, mass memories, processing units and downlink telemetry subsystems. Only difference is that multiple agents within the cluster are forming a virtual bus altogether via wireless communications. Such a virtual mission bus and autonomous task management are discussed by [4.21] and [4.22] in detail.
4.1.2.1 Communications Architecture

The overall architecture is depicted in below Figure 4.6. Within the fractionated spacecraft architecture, there are four types of link to be designed separately due to the different requirements. In addition, since the system is likely to be less than or around 10 spacecrafts each spacecraft can communicate with each other. In turn, this would increase the robustness of the overall system due to distribution of communication abilities.

Fig 4.6 Overview of the communications architecture (gif source: DARPA System F6)

The links are identified with names Link 1, 2, 3, 4 and they can be described one by one. Here one should note that there exists links (link 3) between the payload module and infrastructure modules though it is not depicted in the figure explicitly.

**Link 1: Module <-> Communication and CDH Module (C&CDH)**

This is a two way communication between the C&CDH module and any other module (mentioned as simply module here) within the cluster. From a module to C&CDH module there can be housekeeping and payload data flow. In the opposite direction, from C&CDH to a module, commands arriving from ground and/or necessary housekeeping data would be sent. Housekeeping and command data flow can be handled with a relatively very low data rate (compared to mission payload) in the range of 2 kbps. However payload data flow may require high speed links in the range of 240 Mbps and if the payload module is not desired to have an onboard processor then this rate may increase to 1 Gbps. The housekeeping and command data flow can be realized by low gain antennas or optical terminals. Low cost, low mass, low power and compact optical links for this purpose can be designed and implemented [4.14]. However in case of a communication module failure, it may not be possible to communicate with ground by using optical terminals optimized for inter-module communication. Therefore low gain RF antennas are more favourable in terms of redundancy and safety operations considerations. On the other hand, only optical link can answer such a high speed link demand in the case of payload data flow. Therefore in addition to low gain antennas, an optical link terminal shall also be included in a payload module.
**Link 2: Ground \(\rightarrow\) C&CDH**

This is the classical ground to spacecraft communication link which makes use of a dedicated C&CDH to ground directional RF antenna. Downlink includes housekeeping and payload data and uplink includes the commands. Data rate is likely to be 120 Mbps with respect to the possible mission payload requirements. However with the advances in high rate downlink transmitters, it is possible to raise this up to 500 Mbps in X-band (≈ 8 GHz) with QPSK modulation, low mass and power consumption [4.12]. Therefore an X-Band link can be designed for the purpose of space to ground link for high data rate mission modules.

**Link 3: Module \(\leftrightarrow\) Module**

This link is a direct communication link between any two modules excluding the C&CDH module. The data flow would likely include housekeeping and navigation data to be sent to GNC module for cluster monitoring and power transfer link establishment. The low gain RF antenna with a wide beam, or omni-directional antenna, can be used within this link.

**Link 4: C&CDH\(\rightarrow\) Relay Satellite**

For a real time communication between the cluster and ground, it is necessary to include a link to a relay satellite located at GEO. This may be especially important in case of a disaster monitoring operation. The link to be used should be an optical link to achieve such a high speed data transfer rate with a relatively low mass and power requirements when compared to an RF link.

The possible links defined above include radio frequency (RF) and optical communications. Based on the discussion above we need to define 2 types of inter-satellite optical systems, 1 RF inter-satellite system and 1 X-Band space to ground link system.

### 4.1.2.2 Optical Communication Systems

As mentioned previously, there is a need for two optical links which are for inter-satellite communication within the fractionated spacecraft cluster and inter-satellite communication between data handling module and a relay satellite. In this part these two different optical communication systems will be analysed.

To start with inter-satellite links, we can first analyze the optical terminals for microsatellite swarms defined by [4.14] previously. The specifications of the terminals are provided as follows:

| Optical Intersatellite Link Terminal for Microsatellite Swarms [4.14] |
|---|---|---|---|---|---|---|
| Peak Power | Wavelength | Receiver Aperture | Weight | Power | Ranging Accuracy |
| 160 mW | 980 nm | 5 mm | 900 g | 5 W | 10 m |

This system is optimized for data rates around 100 kbps and a cluster dimension of 1 km\(^2\). This is similar to our case except that there may be a transfer rate requirement of 1 Gbps for a payload data which is to be processed in data handling module. Therefore the existing system should be modified and these two systems should be compared. For the feasibility evaluation, commercially available software [4.16] is used. The analysis on the software is based on eye diagrams and the results are given with a maximum Q factor. Here the relationship between Q factor and bit error rate (BER) can be provided as follows:

\[
BER \approx \frac{1}{\sqrt{2\pi Q}} \exp \left( -\frac{Q}{2} \right) \quad (4.2)
\]

Referring to above equation, the performance of an optical link is better with a higher Q factor. With the mentioned software, the optical link is simulated with the characteristics of existing system as:
<table>
<thead>
<tr>
<th>Case</th>
<th>Transmitter</th>
<th>Transmission Medium</th>
<th>Receiver</th>
</tr>
</thead>
<tbody>
<tr>
<td>ISL with 100 kbps</td>
<td>EDFA(^1) Pump Laser</td>
<td>Wavelength: 980 nm</td>
<td>APD(^3)</td>
</tr>
<tr>
<td></td>
<td>Power: 160 mW</td>
<td>Optics Efficiency: 0.63</td>
<td>Gain: 100, (D_{\text{receiver}}: 5) mm</td>
</tr>
<tr>
<td></td>
<td>NRZ(^2) modulation</td>
<td>Distance: 1000 m</td>
<td>Responsivity: 0.13 A/W</td>
</tr>
</tbody>
</table>

1 Erbium Doped Fiber Amplifier (EDFA), 2 Non-Return to Zero (NRZ), 3 Avalanche Photo Diode (APD)

The resulting eye diagram is presented in below figure. What we see is a very clear diagram with a Q factor around 530. To evaluate the quality we need to look at the point on top horizontal line corresponding to the middle of the eye which is the best sampling instant. The width of the top horizontal line indicates the amount of distortion at the sampling instant which is related to the signal to noise ratio. Therefore the region from the horizontal axis (middle of the eye on x-axis) to the bottom of top horizontal line will provide an idea on how much noise the signal can tolerate. So below figure represents a very robust link for a rate of 100 kbps.

![Eye diagram of 100 kbps optical inter-satellite link](image)

**Fig 4.7** Eye diagram of 100 kbps optical inter-satellite link

For the case of 1 Gbps inter-satellite link, the system is evaluated without interrupting with the transmitter but doubling the receiver diameter. Then the resulting case is summarized below:

<table>
<thead>
<tr>
<th>Case</th>
<th>Transmitter</th>
<th>Transmission Medium</th>
<th>Receiver</th>
</tr>
</thead>
<tbody>
<tr>
<td>ISL with 1 Gbps</td>
<td>EDFA Pump Laser</td>
<td>Wavelength: 980 nm</td>
<td>APD</td>
</tr>
<tr>
<td></td>
<td>Power: 160 mW</td>
<td>Optics Efficiency: 0.63</td>
<td>Gain: 100</td>
</tr>
<tr>
<td></td>
<td>NRZ modulation</td>
<td>Distance: 100 mm</td>
<td>(D_{\text{receiver}}: 10) mm</td>
</tr>
</tbody>
</table>

The eye diagram which is obtained after the simulation is depicted in below Figure 4.8 and we can see the distortions on top horizontal line which indicates a lower signal to noise ratio. The Q factor is around 27 which also indicates this significant reduction. However the amount of noise that the
signal can tolerate, which is the distance from the middle of the eye to the bottom of the top horizontal line (or band), is still acceptable as it is around 80%.

Fig 4.8 Eye diagram of 1 Gbps optical inter-satellite link

To be able to size the system, the scaling method provided in Chapter 9.5.3 of [4.17] is used. Doubling the diameter of the receiver would increase the receiver system size, or volume, by a factor of 8. Then with respect to this reference, the weight and power requirement are estimated from the existing system as

\[ M_{new} = K R^3 M_{existing} \]  \hspace{1cm} (4.3)

\[ P_{new} = K R^3 P_{existing} \]  \hspace{1cm} (4.4)

Where \( R \) is the scaling factor, i.e. the ratio of demanded diameter to existing diameter, and \( K \) is a margin factor. \( K \) is specified as 2 where \( R<0.5 \) and 1 for the other cases. With \( R=2 \), \( K \) becomes 1 for our case. Since the increase is only applied to the receiver and hence the housing, this scaling factor is applied to the mass and power requirement of this portion of the system. In the reference [4.14] the receiver (together with electronics) and housing is given with a mass of 400 grams and a power requirement of 1 Watt. Therefore the mass increase would be 3.2 kg and power increase would be 8 Watts. Finally, the specifications of an optical terminal for an inter-satellite link becomes

<table>
<thead>
<tr>
<th>Modified Optical Intersatellite Link Terminal</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Peak Power</strong></td>
</tr>
<tr>
<td>160 mW</td>
</tr>
</tbody>
</table>

The possibility of ranging with these optical inter-satellite link terminals is also discussed by [4.14]. It is mentioned that the accuracy strongly depends on signal to noise ratio therefore the possibility of ranging with the modified system should be investigated in detail.

Apart from the optical links between the modules, an optical link between the communication module and a relay satellite is also necessary. In this manner, an optical system is discussed by [4.18] and the specifications of this system and the transceiver are provided below:
<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Weight [kg]</th>
<th>Volume [m³]</th>
</tr>
</thead>
<tbody>
<tr>
<td>On board control unit</td>
<td>10</td>
<td>0.018</td>
</tr>
<tr>
<td>Power supply unit</td>
<td>10</td>
<td>0.016</td>
</tr>
<tr>
<td>Telescope and beacon</td>
<td>8</td>
<td>0.064</td>
</tr>
<tr>
<td>Acquisition, Pointing and Tracking (APT)</td>
<td>8</td>
<td>0.018</td>
</tr>
</tbody>
</table>

Table 4.8 Terminal Specifications for Optical Relay Satellite Link

This optical transceiver which is addressed by the authors is based on using the EDFA technology and it is developed by Kongsberg Defense and Aerospace (Norway) [4.19].

Then the specifications of the transceiver are as follows:

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>EDFA Optical transceiver</strong></td>
<td></td>
</tr>
<tr>
<td>Total Mass</td>
<td>2.1 kg</td>
</tr>
<tr>
<td>Data Rate</td>
<td>2.5 Gbps</td>
</tr>
<tr>
<td>Wavelength</td>
<td>1544.5 nm</td>
</tr>
<tr>
<td>Bit Error Rate</td>
<td>$10^{-5}$</td>
</tr>
</tbody>
</table>

Table 4.9 Transceiver Specifications for Optical Relay Satellite Link

4.1.2.3 RF Communication Systems

As specified previously there are two types of RF communication system for the X-band space to ground link and wide beamwidth inter-satellite links. Initially, the inter-satellite system is selected as 2 hemispherical antennas with 220 degrees of beamwidth for each as mentioned in the Table 10.23 and Table 13.16 of ref [4.17]. Then the specifications of a single system are given as follows:

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Mass [kg]</th>
<th>Power [W]</th>
</tr>
</thead>
<tbody>
<tr>
<td>S-Band antenna</td>
<td>1.2</td>
<td>0</td>
</tr>
<tr>
<td>Diplexer</td>
<td>1.2</td>
<td>0</td>
</tr>
<tr>
<td>Receiver</td>
<td>1.8</td>
<td>4</td>
</tr>
<tr>
<td>Transmitter</td>
<td>2</td>
<td>4.4</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>6.2</strong></td>
<td><strong>8.4</strong></td>
</tr>
</tbody>
</table>

Table 4.10 Hemispherical Communication Unit Specifications

This system is selected as 2 antennas so that they could provide an omni-directional pattern together. Therefore they can communicate with other modules regardless of their position within the cluster. Since the desired use of this communication channel is exchanging the housekeeping and position data between the modules, the low data rates are sufficient. In the case of communication module failure, the spacecrafts can also receive commands and downlink the telemetry to the ground with these antennas.
On the other hand the X-band system is based on the transmitter and sized with respect to the given example system on Table 11.26 of [4.17]. Firstly the specifications of the transmitter developed by [4.12] are provided below:

<table>
<thead>
<tr>
<th>X-Band Transmitter</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass</td>
<td>1.1 kg</td>
</tr>
<tr>
<td>Power</td>
<td>30 W</td>
</tr>
<tr>
<td>Data Rate</td>
<td>500 Mbps</td>
</tr>
<tr>
<td>Frequency</td>
<td>8.1 GHz</td>
</tr>
<tr>
<td>RF output power</td>
<td>6 to 20 W</td>
</tr>
</tbody>
</table>

It is mentioned that the data rate of the transmitter is limited to 500 Mbps due to QPSK coding.

The other constituents of the X-band communication system are the receiver, diplexer, switch, multiplier and the antenna system. Antenna system includes the main parabolic antenna, 2 hemispherical antennas and a waveguide antenna. The mass and power estimations of these units are provided below:

<table>
<thead>
<tr>
<th>X-Band Communication System</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Units</td>
<td>Mass [kg]</td>
</tr>
<tr>
<td>Receiver</td>
<td>1.1</td>
</tr>
<tr>
<td>Diplexer, switch, multiplier, etc.</td>
<td>1.5</td>
</tr>
<tr>
<td>Parabolic Antenna</td>
<td>~1</td>
</tr>
<tr>
<td>Hemispherical antenna</td>
<td>1.2</td>
</tr>
<tr>
<td>Waveguide antenna</td>
<td>1.4</td>
</tr>
<tr>
<td>Transponder for the redundant units</td>
<td>4</td>
</tr>
</tbody>
</table>

The parabolic antenna mass is obtained within the sizing code.

### 4.1.2.4 Multiple Access Scheme

Physical communication medium is formed by the spatial dimensions of the cluster and frequency bands. Since there are many links to share this medium and without any interference, there is a need for multi access technique to make every link to be established independently. Among the several multi access methods, the most suitable cases to fractionated spacecraft are frequency division multi access (FDMA) and code division multi access (CDMA). This is mainly because simultaneous multiple crosslink transmissions are desired. In this manner space division multi access (SDMA) can also be considered. However if the pointing, acquisition and tracking system is desired to be kept simple and therefore high beamwidths are required then the use of this technique would be limited to clusters with a small number of spacecrafts. If a comparison is done between the FDMA and CDMA, below table can be provided by [4.13].
Table 4.13 Comparison of Multiple Access Schemes

<table>
<thead>
<tr>
<th>MA Method</th>
<th>Principle</th>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>FDMA</td>
<td>Unique frequencies are assigned for each link</td>
<td>- Simultaneous access</td>
<td>- Greater frequency band allocation with respect to the number of spacecrafts</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>- Cost increase due to frequency variation in cross link design</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>CDMA</td>
<td>Cross link signal is randomly spread across a portion of the frequency band via pseudo noise (PN) codes (each signal is introduced a signature)</td>
<td>- Simultaneous access</td>
<td>- Total number of links is limited by the code noise floor</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Flexible and reconfigurable network</td>
<td>- Enlarged transmit bandwidth</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Possible simultaneous range measurements</td>
<td>- Complex signal processing</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Data flow can be encrypted</td>
<td>- Cost increase due to complexity</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

As it can be seen from table, CDMA is more favourable with respect to its unique advantages such as flexibility and re-configurability. All disadvantages hold for FDMA with a growing cluster and the noise level of CDMA is not significant in a small cluster. Therefore the code division multiple access should be used in terms of inter-module communications. This will also make it possible to encrypt the signals.

4.1.3 On Board Data Handling

Data handling is another shared resource among the fractionated spacecraft cluster. Its functionality is to gather, process, store, format and route the data (commands, housekeeping and payload) either for downlink or onboard use. Most of the data storage and processing required within the cluster can be performed by the data handling module which has a higher storage capability and a high performance computer. Therefore the storage and processor requirements for other modules can be reduced significantly. The baseline capability for a regular module (other than data handling module) could be the housekeeping (navigation data included) and command data handling. Therefore if the data handling module is lost then any module can be capable of orbiting around the earth by processing the commands and navigation data until a new data handling module arrives to cluster.

Owing to the SpaceWire standard [4.11], the data handling module must contain high capacity memory, context saving memory, data compression module, control processor, dedicated processor (for high data rate compressions), digital signal processor, telemetry formatter/encryption module and required routers (for controlling the data traffic).

For sizing data handling system, the relations given by [4.17] in Tables “11.28” and “11.29” can be used to estimate size, weight and power of the CD&H hardware. The system complexity which is required for the estimations can be determined with respect to a given module. To be able to determine the degree of complexity precisely TOPSIS method is used. TOPSIS stands for “Technique for Order Preference by Similarity to Ideal Situation”. To use this method we need to define a decision matrix, A, which provides the relationships between the alternatives and evaluation factors, the importance weights and a positive-negative (PN) definition of the evaluation factors. Here the evaluation factors are Command Rate, Telemetry Rate, Special Functions, Bus Constraints, Reliability, Radiation Environment and Schedule. On the other hand, the alternatives for a data handling system complexity are Simple, Typical and Complex. Therefore we can form the 3x7 decision matrix for each module depending on the complexity of the functions to be performed. A representative decision matrix is formed for the CDH module as follows:

<table>
<thead>
<tr>
<th>MA Method</th>
<th>Principle</th>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>FDMA</td>
<td>Unique frequencies are assigned for each link</td>
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<td>- Greater frequency band allocation with respect to the number of spacecrafts</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>- Cost increase due to frequency variation in cross link design</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>CDMA</td>
<td>Cross link signal is randomly spread across a portion of the frequency band via pseudo noise (PN) codes (each signal is introduced a signature)</td>
<td>- Simultaneous access</td>
<td>- Total number of links is limited by the code noise floor</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Flexible and reconfigurable network</td>
<td>- Enlarged transmit bandwidth</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Possible simultaneous range measurements</td>
<td>- Complex signal processing</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Data flow can be encrypted</td>
<td>- Cost increase due to complexity</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Therefore we can write the $A$ matrix and weight vector, $w$, as

$$A = \begin{bmatrix}
1 & 1 & 1 & 1 & 1 & 1 \\
1 & 1 & 5 & 5 & 9 & 9 \\
9 & 9 & 9 & 9 & 1 & 1
\end{bmatrix}$$

$$w = [0.15 \ 0.25 \ 0.2 \ 0.1 \ 0.1 \ 0.1]$$

The decision matrix is filled by using the numbers 1, 5 and 9 though these numbers are not mandatory. They are selected such that it is possible to reflect the complexity associated with each alternative for a given module in a more precise way. For example if the CDH subsystem is a mixture of single and/or distributed multiple units then we can define the complexity between typical and complex for bus constraint. In the example above the architecture is composed of more distributed units therefore it is more close to “Complex” option. If it was composed of more single units then we would exchange the 9 and 5. In addition, an evaluation factor may not have the same influence on the complexity of CDH system as the others. To reflect this, a weight vector is introduced. Each alternative is given a weight and the sum of these weights must be equal to one. In the above example, Telemetry processing has the greatest influence on the complexity of the CDH system.

After several steps of calculations the decision matrix and the weight vector are used to evaluate the scores of alternatives. This score indicates the closeness of an alternative to the ideal solution which is the degree of complexity in our case. Using this score, we can determine the complexity and therefore the mass and power requirement of the system precisely. The code for the TOPSIS method and the code for CDH system mass and power determination can be seen in appendices.

Finally, the storage of the large amounts of data can be performed by solid state recorders. In this manner, a commercial solid state recorder is chosen with the specifications below.

<table>
<thead>
<tr>
<th>Table 4.14 Solid State Recorder Specifications</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Solid State Recorder [4.20]</strong></td>
</tr>
<tr>
<td>Mass</td>
</tr>
<tr>
<td>Power</td>
</tr>
<tr>
<td>Data Rate</td>
</tr>
<tr>
<td>Capacity (EOL)</td>
</tr>
<tr>
<td>Technology</td>
</tr>
</tbody>
</table>
4.1.4 Power Management

Power is one of the most explicit resources to be shared. Similar to the other resource management strategies, a dedicated power module can be designed to generate, store and distribute (via wireless power transfer) an excess power which is required by other modules in addition to a baseline (survival) power requirement. In this manner, power generation and storage can be reduced for other infrastructure and mission (payload) modules. Here power control and regulation is not considered as it is inherent to the individual module.

4.1.4.1 Power Generation

In a sun-synchronous orbit, the most suitable and mature way to generate power is the use of solar energy. Therefore power generation will be modelled with respect to solar cell arrays. The power generation equations can be given as follows:

- Required Power [4.17]
  \[
  P_{\text{gen}} = t_{\text{day}} \left( \frac{P_{\text{ecl}}}{X_{\text{ecl}}} + \frac{P_{\text{day}}}{X_{\text{day}}} \right)
  \] (4.5)
  
  $P_{\text{gen}}$: Overall required power to be generated by solar arrays
  $t_{\text{ecl,day}}$: Eclipse (~35.2 min) and daylight (~65.8 min) durations in an orbital period
  $P_{\text{ecl,day}}$: Required power during the eclipse and daylight periods
  $X_{\text{ecl,day}}$: Path efficiencies (for direct energy transfer ≈ 0.65 for eclipse and 0.85 for day time)

- Required Solar Array Area [4.17]
  \[
  A_{\text{SA}} = \frac{P_{\text{gen}}}{P_{\text{EOL}}} = \frac{P_{\text{gen}}}{\eta S I_d L_a \cos(\theta_s)}
  \] (4.6)

  $A_{\text{SA}}$: Required solar array area
  $P_{\text{EOL}}$: Power generated at the end of life
  $S$: Average solar power flux in the vicinity of the earth
  $\eta$: End of life solar cell efficiency
  $I_d$: Inherent degradation factor due to integration and other imperfections
  $L_a \approx (1 - D)^t$ where $t$ is the mission lifetime in years and $D$ is the yearly degradation
  $\theta_s$: Worst case sun angle

Using multi-junction GaAs solar cells, the parameters above can be listed as follows [4.17]:

<table>
<thead>
<tr>
<th>$S$</th>
<th>$H$</th>
<th>$I_d$</th>
<th>$D$</th>
<th>$\theta_s$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1367 W/m²</td>
<td>0.22</td>
<td>0.77</td>
<td>0.005</td>
<td>30°</td>
</tr>
</tbody>
</table>

- Finally the mass of the solar array can be approximated for with respect to [4.17] as:

  \[
  M_{\text{SA}} = (0.025)A_{\text{SA}}
  \] (for a specific performance of 40 W/kg) (4.7)
4.1.4.2 Power Storage

Power storage is performed by onboard rechargeable batteries. The use of Li-ion batteries on board instead of Ni-Cd or Ni-H2 ones is an effective contributor to the power storage performance and mass reduction. Li-ion batteries provide great improvements in terms of specific energy, energy efficiency and thermal management. They are also modular as they can be put in parallel. Therefore Li-ion batteries should be used on board as secondary batteries. Commercially available and space proven (LEO) Li-ion battery characteristics [4.29] can be given in below table:

<table>
<thead>
<tr>
<th>Table 4.15 Li-ion Battery Characteristics</th>
</tr>
</thead>
<tbody>
<tr>
<td>LEO Satellite Size</td>
</tr>
<tr>
<td>Mean / End of Charge Voltage [V]</td>
</tr>
<tr>
<td>Energy [W.h]</td>
</tr>
<tr>
<td>Specific Energy [Wh/kg]</td>
</tr>
<tr>
<td>Weight (one battery) [kg]</td>
</tr>
</tbody>
</table>

The relation for battery capacity and the mass can be given as follows:

\[ M_b = \frac{P_{\text{stored}} \cdot T_{\text{duration}}}{\text{DOD} \cdot \eta \cdot C_r} \tag{4.8} \]

DOD: Depth of discharge
Cr: Battery specific energy in [W.hr/kg]
\( \eta \): Battery efficiency
\( P_{\text{stored}} \): The total energy to be stored by the batteries
\( T_{\text{duration}} \): The total period that the stored energy is used/needed

4.1.4.3 Wireless Power Distribution (WPT)

One of the unique advantages of the fractionated spacecraft is the physically independent modular architecture. Therefore any failed module can be replaced with a new one. Since any kind of power transfer by means of cables would bring physical dependency, the wireless power transfer (WPT) is an essential part of the fractionated spacecraft architecture.

The concept of wireless power transfer dates back to Nicola Tesla’s early works towards the end of 19th century. Later radio and microwave power transfer was studied. Currently the latest concepts are the power beaming via lasers and electromagnetic induction. The idea of wireless power transfer in space applications is also not new. In late 60s Peter Glaser proposes beaming the solar power captured in space to earth and this is recognized as the first description of solar power satellites. Currently the wireless power transfer is also studied within the development of a space elevator [3].

The means of wireless power transfer can be named as induction (electromagnetic, electrodynamic and electrostatic) for short ranges (in the order of meters); radio/microwave or laser power beaming, electric conduction and concentrated/reflected sunlight for long ranges (in the order of kilometres). The use of magneto inductive resonantly coupled WPT systems developed by Lockheed Martin for the purpose of fractionated spacecraft is not viable over 10 m distances due to the efficiency fall off as a function of range and requirement for extremely high power sources [4.25]. Since it is likely that the cluster dimensions could reach over a few hundred meters to perform safe cluster navigation,
long range power transfer means deserve more attention. Among the long range power transfer means, microwave and laser power beaming are the most considered ones among the discussions for space applications. The most basic associated hardware can be enumerated for both systems as:

- **Microwave/Radio Transmission**: Transmitting antenna and rectenna which converts electromagnetic wave into DC

- **Laser Power Beaming**: Laser diode array (transmitter), microlenses and solar arrays (receiver)

Each of these two power transfer methods has advantages and disadvantages. Although the efficiencies would be higher for the microwave transmission, it is discussed by [4.26] that the laser power beaming is favoured in order to avoid the drawbacks of microwave transmission such as side lobes & spikes, more integration complexity, higher cost, higher mass and sizing requirements of transmitting elements (up to a factor of 50) compared to laser system. As the hardware constraints also favour the laser power beaming, and there would be already solar arrays on board the modules, the power transfer should be performed via lasers.

The overview of the power transmission link can be provided as follows:

![Diagram of wireless power transfer](image)

**Fig 4.9** Illustration of the wireless power transfer

The generated power by the solar arrays is converted into laser beam, directed and transmitted by the power module towards the solar arrays of the receiving module. Therefore this can be modelled as a power transmission link with emphasize on the increased efficiency. Here a penalty would be an additional pointing mechanism to direct the laser beam, however, this wouldn’t increase the mass significantly.

Based on commercially available laser power beaming systems [4.27,4.28], power transfer of several hundred watts over a range of 1 km is possible with an overall efficiency around 26%. For short ranges, it is possible to deliver around 1 kW levels continuously. This corresponding system would have a power density of 1 kW/kg and the overall link efficiency can be given in below table:

**Table 4.16** Approximate efficiencies of wireless power transfer

<table>
<thead>
<tr>
<th>Approximate Efficiencies [4.27]</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Power Conversion</td>
<td>80 %</td>
</tr>
<tr>
<td>PV Efficiency (receiver efficiency)</td>
<td>40 %</td>
</tr>
<tr>
<td>Tracking</td>
<td>90 %</td>
</tr>
<tr>
<td>Optical Transmission</td>
<td>90 %</td>
</tr>
<tr>
<td><strong>Overall</strong></td>
<td>26 %</td>
</tr>
</tbody>
</table>
In future, the focus of the main research and development in terms of laser power beaming will likely be on improved solar array efficiencies and cost reduction [4.28].

4.1.4.4 Power Management Strategy

As it is desired to have the capability to provide as much extra resource as possible by infrastructure modules the failure of an infrastructure module is the limiting case. In the case of such a power module failure, an individual module within the cluster should be able to satisfy a minimal baseline power requirement in order to survive until a new power module starts to operation. In this manner, the baseline power generation and storage (which every module would perform) should be based on the power consumption of the required hardware for GNC collision avoidance / safe mode. To estimate the power required for a module, the Table 10-9 of [4.17] can be used. The power requirements for payloads are approximated with respect to the spacecraft sizes as follows.

| Typical Power Requirement Percentages of Payloads with respect to Overall Operating Power |
|-----------------------------------------------|-----------------------------------------------|-----------------------------------------------|
| Small (<100 W)*                                | Medium (~200 W)*                               | Large (>500W)*                                |
| % 20 to % 50                                   | % 40                                         | % 40 to % 80                                  |

*Numbers in the brackets represent the overall operating power

Therefore we can initially assume that the spacecraft should produce around % 60 of total power requirement by its own solar cells as a baseline. Regarding the excess power required within the modules (such as for payload) there must be an optimization between the increased solar cells (generation), increased battery capacity (storage) and/or increased power transfer. Regarding above discussions below table could be beneficial to summarize the specific characteristics of each with respect to the means used:

<table>
<thead>
<tr>
<th></th>
<th>Generation (Solar Cell)</th>
<th>Storage (Battery)</th>
<th>WPT (Laser)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Efficiency</td>
<td>25%</td>
<td>96%</td>
<td>25% ((P_n, to P_{transferred}))</td>
</tr>
<tr>
<td>Mass</td>
<td>(~(0.01)P_{generated})</td>
<td>(~(0.026)P_{stored})</td>
<td>(~(0.001)P_{transferred})</td>
</tr>
</tbody>
</table>

As it can be seen from the table that if the excess power (=total required power – baseline required power) is generated by the power module and transferred continuously to the receiver modules then the solar array area (to generate the required power) of the power module would be increased 4 times (due to 25% efficiency in the transmission) corresponding to a reduction in the solar array area of a receiver module. Therefore to find the exact feasibility, whole energy flow throughout system should be considered together with the cost and the value (benefit) of power transmission.

The overview of the energy flow within the system is as follows:
While optimizing this energy flow, the goal should be achieving the required power for both eclipse and daylight periods by increasing the net value of power management. The value of power transmission would come from the increase in continuous power supply ability where power generation or storage would not be possible within a receiver module.

The power supply during the daylight and eclipse periods can be formulated as follows:

- **Daylight:**
  
  \[
  P_{\text{used}} = C_1 \cdot P_{\text{generated}} + C_2 \cdot P_{\text{transferred}}
  \]

  \[
  P_{\text{stored}} = (1 - C_1) \cdot P_{\text{generated}} + (1 - C_2) \cdot P_{\text{transferred}}
  \]

  \[
  P_{\text{required, day}} = P_{\text{used}} + P_{\text{stored}} = P_{\text{generated}} + P_{\text{transferred}}
  \]

- **Eclipse:**

  \[
  P_{\text{required, eclipse}} = P_{\text{used}} = P_{\text{stored}} + P_{\text{transferred}}
  \]

Where the coefficients

- \( C_1 \): the percentage of the onboard generated power to be used directly
- \( C_2 \): the percentage of the transferred power to be used directly

The resulting optimization problem for a given power requirement can be written as follows:

- **Objectives**
  - Reduce: Mass, cost, power generation (receiver), power storage (receiver)
  - Increase: Value ( power transfer (power module) , power generation (power module) )

- **Variables:** Generated power, transferred power, \( C_1 \), \( C_2 \)

- **Constraints:**
  
  \[
  P_{\text{required, day}} = P_{\text{used}} + P_{\text{stored}} = P_{\text{generated}} + P_{\text{transferred}}
  \]

  \[
  P_{\text{required, eclipse}} = P_{\text{used}} = P_{\text{stored}} + P_{\text{transferred}}
  \]
4.2 Notional Spacecraft Architecture

In parallel to the discussions previously, the notional virtual spacecraft architecture, or module network, involves infrastructure modules and mission, or payload, modules. Infrastructure modules are Guidance, Navigation and Control module (GNC), Communication and Data Handling Module (CCDH) and Power Module. Mission modules are the ones which benefit from the resources generated by the infrastructure modules. In this manner, the summary of functionalities of the modules can be seen in below table:

<table>
<thead>
<tr>
<th>Table 4.17 Functionalities and resources of spacecraft modules</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>GNC</strong></td>
</tr>
<tr>
<td>---------------------------------------------------------------</td>
</tr>
<tr>
<td>Responsible for cluster navigation management</td>
</tr>
<tr>
<td><strong>Resource:</strong></td>
</tr>
<tr>
<td>Absolute and relative navigation processing</td>
</tr>
</tbody>
</table>

As it is mentioned previously the baseline requirement for any module is the ability of self absolute navigation. Therefore each spacecraft must have the minimum capabilities of attitude and orbit control, communications and data handling, power management and thermal control. Since fractionating these subsystems would require significant technological breakthrough [4.21] it is more viable to fractionate the resources.

An excellent example for a mission module is the satellite bus system, NEXTAR, produced by NEC Company in Japan [4.31]. This system is depicted with various earth observation payloads in Fig. 4.12

![NEXTAR Bus System](image)

**Fig 4.12** Bus system which is adaptable for various mission payloads, figure ref: [4.31]

The specifications of the bus system [4.31] are as follows:

<table>
<thead>
<tr>
<th>Table 4.18 NEXTAR Bus System Specifications</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Bus Dry Mass</strong></td>
</tr>
<tr>
<td><strong>Bus Power</strong></td>
</tr>
<tr>
<td><strong>Bus Dimensions</strong></td>
</tr>
<tr>
<td><strong>Payload Mass</strong></td>
</tr>
<tr>
<td><strong>Payload Power</strong></td>
</tr>
</tbody>
</table>
The system block diagram is also provided in below figure:

![System block diagram of NEXTAR](image)

**Fig. 4.13** System block diagram of NEXTAR, (figure ref [4.31])

This bus system can be also taken as the baseline system with some exceptions. In this manner, the baseline bus system elements for the modules within the cluster can be shown by the below table:

### Table 4.19 Baseline bus specifications

<table>
<thead>
<tr>
<th><strong>Attitude Control</strong></th>
<th>4 Reaction wheels, 3 Magnetic torquers, rate and reference sensors</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Orbit Control</strong></td>
<td>6 Monopropellant 1N Hydrazine Thrusters and CDGPS</td>
</tr>
<tr>
<td><strong>Communications</strong></td>
<td>2 Hemispherical S-Band antennas, 1 cluster optical link terminal</td>
</tr>
<tr>
<td><strong>Power</strong></td>
<td>Multi-junction GaAs solar arrays, Li-Ion batteries</td>
</tr>
<tr>
<td><strong>Data Handling</strong></td>
<td>Basic computer for flight and data routing (storage is module dependant)</td>
</tr>
<tr>
<td><strong>Thermal Control</strong></td>
<td>Passive control with radiators, active control if necessary</td>
</tr>
</tbody>
</table>

Taking this as a baseline, either the payload or the resource generation capability is added to above bus system. Then the additional capabilities can be summarized for each infrastructure module as:

### Table 4.20 Additional Capabilities for Infrastructure Modules

<table>
<thead>
<tr>
<th><strong>Additional Capability</strong></th>
<th><strong>GNC</strong></th>
<th><strong>Comm &amp; DH</strong></th>
<th><strong>Power</strong></th>
</tr>
</thead>
</table>
As it can be seen from above table, the capability increase for communication and data handling module is more than the others. In order to increase the robustness of the infrastructure cluster, we can add more communication and data handling capabilities to GNC and Power modules. For example, we can introduce an additional advanced processor, solid state recorders and an X-band communication system to GNC module. On the other hand, an additional relay satellite communication system can be introduced to the power module. Then the final capabilities of the modules can be proposed as follows:

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>GNC</strong></td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td></td>
</tr>
<tr>
<td><strong>Comm &amp; Data Handling</strong></td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
</tr>
<tr>
<td><strong>Power</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>*</td>
<td>*</td>
</tr>
</tbody>
</table>

From above it can be seen that if all the resources are doubled then the capability of the entire system would be enhanced. In addition, robustness would be achieved due to redundancy, or distribution of the resources. For example if GNC module fails then communication and data handling module can be responsible for the management of the cluster navigation until a new GNC module joins the network. This is an example to the graceful degradation of the overall system capabilities. Here the only resource that there is no redundancy for is the power distribution unit. This is simply because power module would probably have large solar arrays and therefore it may not be possible to fit it in a small and responsive launch vehicle with parabolic antennas.

Finally, this architecture would be a good example for scaling the capability of the infrastructure network up by incremental launch. For example launching only the communications and data handling module would be enough to make a payload operable. Then power module can be launched and finally a fully operational system can be achieved by launching the GNC module.
CHAPTER 5

System Modelling and Sizing

“I know all the things around your head
and what they do to you.
what are we coming to?”

Black Star, The Bends, Radiohead
5. System Modelling and Sizing

5.1 The Parametric Bus Sizing Code

The fundamental elements of a spacecraft system design are based on the payload requirements and the spacecraft bus is sized with respect to these requirements. In the case of fractionated spacecraft, the payloads could be either the mission specific instruments or the subsystems producing more resources for the rest of the cluster and these instruments. Therefore each module has specific requirements with respect to its task within the cluster. Based on the technological state and the system analysis discussed previously, the overall sizing code is parameterized so that different modules can be sized with respect to their different requirements. Besides the payload requirements, there are also mission specific inputs such as launch altitude, inclination and mission lifetime.

The basic elements, or the functions, of the bus sizing code are orbit, guidance, navigation and control (GNC), command and data handling (CDH), communication, power, structure and thermal. These models of the subsystems and their interactions are summarized in below table. The points on upper triangle indicate the feed-forward information from the related subsystems and the ones on lower triangle indicate the feedback information. The information flow is from left to right.

<table>
<thead>
<tr>
<th>Model</th>
<th>Payload</th>
<th>Orbit</th>
<th>GNC</th>
<th>CDH</th>
<th>Comm</th>
<th>Power</th>
<th>Structure</th>
<th>Thermal</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td></td>
</tr>
<tr>
<td>Orbit</td>
<td>*</td>
<td></td>
<td></td>
<td></td>
<td>*</td>
<td>*</td>
<td></td>
<td>*</td>
</tr>
<tr>
<td>GNC</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>*</td>
<td>*</td>
<td></td>
<td>*</td>
</tr>
<tr>
<td>CDH</td>
<td></td>
<td></td>
<td>*</td>
<td>*</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Communication</td>
<td></td>
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<td>*</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Power</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>*</td>
<td></td>
<td>*</td>
</tr>
<tr>
<td>Structure</td>
<td>*</td>
<td>*</td>
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<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Thermal</td>
<td></td>
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<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

As a starting point the inputs to the overall sizing code are represented as follows:

<table>
<thead>
<tr>
<th>Module and Payload Inputs</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Mission</strong></td>
</tr>
<tr>
<td><strong>Orbit</strong></td>
</tr>
<tr>
<td><strong>GNC</strong></td>
</tr>
<tr>
<td><strong>CDH</strong></td>
</tr>
<tr>
<td><strong>TT&amp;C</strong></td>
</tr>
<tr>
<td><strong>Power</strong></td>
</tr>
<tr>
<td><strong>Structure</strong></td>
</tr>
</tbody>
</table>

These inputs are either payload requirements or initial guesses and all are fed forward to the related subsystem sizing functions in the order presented above. Then the overall code is iterated for more than 5 times to have the optimized solution. The constituents of the sizing code, modelling methodology and the assumptions are explained one by one in the following sub-sections.
5.1.1 Orbit

This function includes the sub-functions orbit, cluster and orbit ∆V budget. The overall scheme of the information flow within the orbit function code can be seen in below figure. The bold written ones are the inputs provided to the overall “orbit” function.

![Diagram of orbit section of the sizing code](image)

**Fig 5.1 Overview of the orbit section of the sizing code**

The orbit sub-function calculates the orbital parameters such as semi-major axis (a), altitude (h), inclination (i), period (T), orbital speed (V) as well as the day and eclipse periods using the algorithm [2.4] provided in the appendices. Here the eclipse and day periods are calculated with respect to worst case conditions. Then for a given mission time, launch altitude and cluster positions; the total ∆V budget of the spacecraft is calculated with respect to the cluster acquisition, orbit maintenance and disposal phases. For acquisition, we can think of two cases: one is the direct launch to the desired orbit and the other is the launch to lower altitudes and inclination. If the case is the direct launch then the ∆V is calculated with respect to the cluster acquisition approach which is described in the appendices. If it is not a direct launch then Hohmann transfer is applied. The absolute orbit maintenance calculation is based on the drag compensation since the effects of solar radiation pressure and third body perturbations are relatively low. Here the ballistic coefficient (Cf) is initially estimated for the worst case by using the dry mass (for estimating the gross mass), spacecraft density (for estimating the volume and therefore drag area), power requirement (for estimating the solar array area) and the drag coefficient (Cd≈2.2) inputs as follows

\[
C_f = M_{\text{gross}} / \{C_d(A_{\text{drag}} + A_{\text{solar array}})\} 
\]

(5.1)

Then this is updated in the structure function by using the updated gross mass and drag areas. Drag area is estimated as the diagonal cross sectional area of the cubic satellite. Finally, ∆V for the absolute orbit maintenance is calculated as:

\[
\Delta V_{\text{maintenance}} = \pi / C_f \rho a V N_{\text{rev}} t_{\text{mission}} \]

(5.2)

Where \(\rho\) is the atmospheric density at the orbit altitude, \(V\) is the orbital velocity \(N_{\text{rev}}\) is the revolution per year and \(t_{\text{mission}}\) is the mission lifetime.

Cluster maintenance is mainly due to J_2 perturbations and the required equations are provided in the cluster flying section. Finally, the last part of the ∆V budget is for the disposal at the end of the lifetime and it is calculated such that the spacecraft is brought to 150 km of altitude. The equation to calculate this final ∆V is given below:
\[ \Delta V_{\text{disposal}} = V \left( 0.5 \left( H_i - H_e \right) / \left( 2 R_E + H_i + H_e \right) \right) \]  

(5.3)

Where \( R_E \) is the radius of the earth, \( H_i \) is the initial altitude and \( H_e \) is the re-entry altitude.

Then the total \( \Delta V \) budget becomes the sum of the acquisition, maintenance and disposal constituents. Finally, this information is forwarded to the GNC function to calculate the required propellant mass with respect to the thrusters’ specific impulse and the dry mass of the spacecraft.

### 5.1.2 Guidance, Navigation and Control (GNC)

For a given set of requirements such as \( \Delta V \), pointing, slew and rate information, the GNC function calculates the mass and power of the attitude and orbit control system in addition to the propellant mass for the given thrusters. The overview of the GNC sizing code is provided below. Here the bold written parameters are given as inputs to this function.

---

**Fig 5.2 Overview of the GNC sizing code**

Initially based on the pointing and angular rate information requirements, a set of attitude sensors are selected. One set includes the reference sensors such as star tracker, sun sensor and magnetometer and the other set includes the rate sensors such as fiber-optic gyro, inertial gyro and magnetometer.
Then the reaction wheels are selected such that the slew torque and angular momentum storage requirements are met. Here angular momentum storage requirement is evaluated based on the pointing requirement and disturbance torques. Disturbance torques are calculated with respect to the gravity gradient and magnetic field of the Earth. As a first assessment, these two cases are considered since they are assumed to be stronger than the solar radiation pressure and aerodynamic drag effects for a small spacecraft at 800 km altitude. For the gravity gradient disturbance calculation, the difference between the maximum and minimum inertias is considered and the worst case deviation angle is specified as 45 degrees. On the other hand, the magnetic disturbance is calculated in accordance with the assumption on Table 11-9B of reference [4.17]. In accordance with the same table, it is assumed that the typical residual spacecraft magnetic dipole is around 1 A.m² for an uncompensated small spacecraft.

Finally the momentum storage requirements are calculated for both pointing requirement and disturbance torques. Based on the bigger angular momentum storage value, the reaction wheel is selected from the 1 Nms and 4 Nms alternatives which are mentioned previously in system analysis section. As a second set of actuators, magnetic torquers are sized with respect to the disturbance torques and with some additional margin. After the selection of attitude sensors and actuators, the propellant requirement for the momentum dumping of reaction wheels is calculated for the given thruster. Here it is assumed that this operation is carried under every day with a single pulse for operating 3 reaction wheels. The margin used here was 50% for directing the thrusters to perform cluster manoeuvres.

After sizing the attitude control system and finding the associated propellant mass, the same is done for the orbit control system. Based on the ∆V requirements, specific impulse of the thrusters and dry mass of the spacecraft the total propellant mass for the orbit control is calculated with a %30 margin. Finally this is added to the attitude control propellant to obtain the GNC, or spacecraft, propellant mass budget. The equations used for sizing the GNC system are provided in appendices and the references are also given in the code where it is used.

5.1.3 Command and Data Handling

Command and data handling system is sized with respect to a given historical data [4.17] by defining the complexity of the data handling system. In addition, for a given link type a solid state recorder is added if required. The overview of the code is as follows

![Fig 5.3 Overview of the CDH sizing code](image-url)
Given the telemetry data processing rates and initial decision making matrix, $A_{\text{initial}}$, the decision (A) matrix is updated based on the telemetry data rate. Then together with the weights of the evaluation factors, this final matrix is put into TOPSIS evaluation and the individual scores of the alternatives, i.e. simple, typical and complex, are obtained. Then these individual scores ($C$), or the measures, of the complexity are used to determine the mass and power of the CDH subsystem.

In addition, if the module is holding a high data rate X-band downlink then this also accommodates a solid state recorder. Therefore solid state recorders are also added regarding the link type and data volume. Finally, the required data volume is calculated with respect to an average ground contact time of 10 minutes and data rate. Then this data volume is doubled in case of a ground contact miss.

5.1.4 Communications

The communication function is the most complex but with the least input. Since communication architecture is defined and the systems are almost sized, it calculates the total masses and power requirements of the RF and optical units for a given specific module. The specific module is identified by the link type and the overall architecture of the code is shown in Figure 5.4.

Within the code there are four link types identified. These are called as Link 1, Link 14, Link 2 and Link 24. These names refer to the links which were described in detail previously. However the communication systems involved with these links can be seen below:

<table>
<thead>
<tr>
<th>Link Types and Related Communication Systems</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Link 1</strong></td>
</tr>
<tr>
<td><strong>Link 14</strong></td>
</tr>
<tr>
<td><strong>Link 2</strong></td>
</tr>
<tr>
<td><strong>Link 24</strong></td>
</tr>
</tbody>
</table>

If the specified link type includes a space to ground link, i.e. either 2 or 24, then initially this X-band link hardware is sized under “RF size” sub-function which includes “RF-link” and “RF-antenna” sub-functions. The transmitter power is fixed to 20 W and sizing of the antenna is based on its diameter. Then the rest of the X-band RF system sizing is based on the transmitter mass. Depending on the relay satellite optical communication capability, i.e. if the link is 24, this system is also included.

If a module doesn’t have a dedicated high data rate space to ground link, then the communication function sums the associated hardware mass and power with respect to the specific link type. The hardware includes RF and optical inter-satellite links and may include optical relay satellite link accommodated in the power module.

Finally total mass and power budgets are obtained and forwarded to the structure and power functions.
Fig 5.4 Overview of the communication system sizing code
5.1.5 Power

Power sizing code includes 3 main sub-functions, Figure 5.5. These are generation, storage and distribution (or wireless power transfer). The code initially calculates the mass and power requirements associated with the wireless power transfer. Here a module can either receive \( (P_{\text{rec}}>0, P_{\text{dis}}=0) \) or distribute \( (P_{\text{rec}}=0, P_{\text{dis}}>0) \) power. If the module is power module, therefore \( P_{\text{dis}}>0 \), then the required power generation to transfer the amount of \( P_{\text{dis}} \) is calculated as:

\[
P_{\text{dis,gen}} = \frac{P_{\text{dis}}}{\eta_{\text{WPT}}}
\]

where \( \eta_{\text{WPT}} \) is the overall efficiency of the wireless power transfer which is around 26%. In other words, the solar array needs to generate 4 W in order to be able transfer 1 W of power. The resulting mass is calculated with respect to the received and transmitted power since the receiver and transmitter have different specific weights defined as kg/W.

If the module is a receiver one then the amounts of power to be generated for the eclipse and day times are reduced by the amount of the received power.

In the generation part, it can be seen from the bold inputs that the solar array is sized with respect to eclipse and day time power requirements as well as power generation requirement regarding wireless power transfer. The additional power requirements (such as wiring) are also added before the solar arrays are sized.

Finally, batteries are sized with respect to storage requirements and other power subsystem mass contributors (power control units, wiring, etc.) are included together with the solar array mass and WPT mass (either receiver or transmitter) in the overall mass budget. Here additional units are estimated with respect to given relations in Table 10.27 of [4.17].

5.1.6 Structure

Structure function, which is shown in figure 5.6, provides the overall mass budget by summing Payload, GNC, CDH and Power subsystem masses and estimating the thermal control and structure masses. The estimations are done such that the structure mass is 20% and thermal control system mass is 5% of the dry mass referring to Table 10.31 of [4.17]. After these estimations, the overall mass budgets are obtained as bus mass, dry mass and gross mass (=dry + propellant). Then using the given spacecraft density, the volume, linear dimensions and body area are calculated for a cubic satellite. Here spacecraft density is defined as 200 kg/m\(^3\) with respect to the NEXTAR bus system example which is explained in previous chapter.

The mass moment of inertias are determined by including the solar array contributions expressed by the relations in Table 10.9 of [4.17]. The inertias are also calculated with respect to launch configuration and resulting fundamental frequencies are calculated by using the equations provided in Table 11.42 (case C and D) of [4.17]. These equations are also given in appendices. Finally, using the diagonal cross sectional body area and solar array area, the ballistic coefficient, \( C_f \), is updated with respect to the equation provided in the orbit function.
Fig 5.5 Overview of the power system sizing code
5.1.7 Thermal

The thermal function evaluates the thermal cycle of the spacecraft to find the equilibrium temperatures achieved after a number of orbital periods (around 10 days). To do this a relatively safe initial condition is defined as 15 °C. Then the heat sources are identified as Sun, Earth and Albedo in addition to the internal dissipations within the spacecraft. The mean of heat rejection is the radiators whose area is estimated from the thermal mass estimation. For radiator, 5 mil Silver Teflon is chosen with the emissivity ($\varepsilon$) of 0.78 and end of life absorptivity ($\alpha_{eol}$) of 0.1. The equation for thermal cycle for a spacecraft with a specific heat capacity of $C$ can be given as follows:

$$T_{i+1} = T_i + \frac{Q_{in} - Q_{out}}{M_{gross}C\Delta t}$$  \hspace{1cm} (5.5)
Here $Q_{in}$ changes during the day and eclipse durations. Though the radiator area, $A_r$, is fixed, $Q_{out}$ is also changing since the spacecraft is cooling and warming up through its thermal cycle. The equations for both can be written for these as

$$Q_{in,\text{day}} = Q_{s,\text{max}} + \alpha_{eol} S A_s + \varepsilon I_R A_e + \varepsilon S \rho_{\text{albedo}} A_e \tag{5.6}$$

$$Q_{in,\text{ecl}} = Q_{s,\text{min}} + \varepsilon I_R A_e \tag{5.7}$$

$$Q_{out} = \varepsilon \sigma A_r T_i^4 \tag{5.8}$$

The coefficients used are can be seen in below table:

**Table 5.4** The coefficients used in thermal sizing code

<table>
<thead>
<tr>
<th>$Q_{s,\text{max}}$ [W]</th>
<th>$Q_{s,\text{min}}$ [W]</th>
<th>$S$ [W/m$^2$]</th>
<th>$I_R$ [W/m$^2$]</th>
<th>$\sigma$ [W/m$^2$K]</th>
<th>$\rho_{\text{albedo}}$</th>
<th>$C$ [J/kgK]</th>
</tr>
</thead>
<tbody>
<tr>
<td>$P_{\text{day}}$</td>
<td>$P_{\text{ecl}}$</td>
<td>1367</td>
<td>218</td>
<td>$5.6704 \times 10^{-8}$</td>
<td>0.25</td>
<td>900</td>
</tr>
</tbody>
</table>

The areas $A_s$ and $A_e$ are assumed to be same and taken as the diagonal cross section area of the spacecraft. After the simulation final maximum and minimum equilibrium temperatures and the experienced maximum and minimum temperatures through the cycle are determined. Then these are checked whether they are within the acceptable temperature intervals or not. The operational temperature interval for a spacecraft is assumed as 10 to 30 °C with respect to Li-ion batteries and hydrazine tanks and lines [4.17].

![Fig 5.7 Overview of the thermal sizing code](Image)

5.2 Sizing Results

For sizing and evaluation, 4 types of modules were considered. These are 3 infrastructure modules, i.e. GNC, C&CDH and Power, and the payload modules. For each type of module a set of individual requirements, or inputs, were given to the sizing code and associated mass, volume and power consumption were obtained. All the inputs and sizing codes are provided in the appendices.

For GNC and C&CDH modules, the total data rate that is to be handled is increased gradually and the mass, power and volume of the spacecraft is estimated for each case. The diameter of the parabolic antenna to be used for X-Band link is also increased gradually from 0.1 meters to see if link margin is satisfied. Since the only difference between the GNC and C&CDH modules is the optical relay satellite link terminal, only the C&CDH module results will be presented. However the overall mass, power and propellant budgets will be presented for both at the end of this section.

The increase of data handling capability and its effect to module mass and bus power can be seen in below figure

![Figure 5.8 C&CDH module gross mass and bus power with respect to handled data rate](image)

A data rate of 5.5 Gbps can be handled by a C&CDH module with a gross mass of 380 kg and power consumption of 430 W. Every 1.6 Gbps data rate capability increase introduces another solid state recorder with a mass of 15 kg and this addition causes an increase in the gross mass around 35 kg. The power increase is equivalent to the power consumption of a solid state recorder and it is around 50 W. With this configuration the C&CDH module accommodates 4 solid state recorders which in total provide a data volume of 16 TB. Since there is still volume and allowable weight margin for the launch, the capability of the module can be increased. However this configuration would already support 5 payloads with a payload data rate of 1 Gbps.

The X-band parabolic antenna diameters and the associated link margins can be seen below

<table>
<thead>
<tr>
<th>$D_{ant} (m)$</th>
<th>0.15</th>
<th>0.2</th>
<th>0.25</th>
<th>0.3</th>
<th>0.35</th>
<th>0.4</th>
<th>0.45</th>
<th>0.5</th>
<th>0.55</th>
<th>0.6</th>
</tr>
</thead>
<tbody>
<tr>
<td>$LM (dB)$</td>
<td>1.55</td>
<td>4.05</td>
<td>5.98</td>
<td>7.57</td>
<td>8.91</td>
<td>10.07</td>
<td>11.09</td>
<td>12.01</td>
<td>12.84</td>
<td>13.59</td>
</tr>
</tbody>
</table>

An antenna diameter of 0.2 to 0.25 would be sufficient for the X-Band link where ground station antenna was estimated as 6 m.

On the other hand, the volume increase of the module and the maximum and minimum equilibrium temperatures can be seen below.
With a fixed spacecraft density of 200 kg/m$^3$, the increase in volume is proportional to the mass increase of the spacecraft. From the thermal control point, we can see that the temperature change between the eclipse and daylight conditions is around 5 °C for each condition and each module operates within the acceptable temperature range.

In the case of power module, the effect of power transfer capability increase was investigated. The sizing code was iterated for 100 times and power transfer was increased by 30W. The power dissipation due to the inefficiencies of wireless power transfer was also considered within the thermal equilibrium calculations. These power dissipation sources are the conversion, tracking and distribution units. On the other hand, the solar array volume for the launch configuration was also estimated and added to the overall spacecraft volume. For launch configuration solar array volume, the array area was multiplied by 5 cm. to compensate the folding mechanism.

The results of the first iteration are given as follows:

From above figures we can see that the mass increase is more limiting than the volume increase. Regarding 550 kg of mass limitation due to launch vehicle, it is possible to transfer 1300 W of power. The 28 m$^2$ solar array generates 5.25 kW of power which is pointed in below figure. The thermal equilibrium curves and the generated power with respect to the distributed power can be seen below:
Fig 5.11 Generated power and equilibrium temperature plots for power module

Though the power dissipation due to wireless power transfer is not low, the module shows a cooling trend with increasing mass. This is due to the low power consumption of the bus with respect to the increasing size of the module. Since the corresponding equilibrium temperature interval for the 550 kg module is from 5 to 9 °C the radiator area can be reduced a little to increase this temperature interval.

The sizing of the mission modules was performed for two cases. Initially no power transfer was considered and the maximum allowable mass within the spacecraft was found. Then the power transfer was applied and corresponding allowable module masses and power requirements were found. The assumed lifetime for a payload module was 5 years.

For each case, the payload mass is increased by 2 kg for each iteration and with a corresponding payload power increment of 6W. This ratio is assumed with respect to the NEXTAR example. The results of this first case can be shown below.

Fig 5.12 Total mass and bus mass of a payload module with respect to increasing payload mass (without wireless power transfer)

From above figures it is observed that a maximum payload mass around 200 kg can be accommodated by a payload module bus of 288 kg. Except the bus mass, allowable payload mass is the same as the NEXTAR example. In the next figure, required solar array areas and the equilibrium temperatures are shown.
For the case without power transfer, the payload module has a solar array area of 6.7 m². The equilibrium temperature interval is from 4 to 7 °C. Therefore the radiator area can be reduced to increase the operational temperature interval to a higher level.

Though the bus weight is larger than the NEXTAR example, the payload module can benefit from the advanced data processing, real time communication and power transfer at the same time with the fractionated spacecraft architecture.

The final evaluation is based on the power transfer. Assuming a cluster with a 5 payload units, the distributed power per payload can be specified as 260 W. The results of this case are shown below.

As it can be seen from the points located on the figures, the allowable payload mass is increased by 20 kg. The bus mass is determined as 262 kg which is 26 kg below the previous case. Finally the solar array area and thermal equilibrium plots are provided as
We can see from above figure that the wireless power transfer reduced the required solar array area significantly. It is seen that we can launch payloads with a mass of up to 75 kg without any solar power generation.

Finally the precise mass, propellant and power budgets of the modules considered previously are summarized in below table:

<table>
<thead>
<tr>
<th>Module</th>
<th>$M_{\text{gross}}$ [kg]</th>
<th>$M_{\text{dry}}$ [kg]</th>
<th>$M_{\text{prop}}$ [kg]</th>
<th>$M_{\text{payload}}$ [kg]</th>
<th>$P_{\text{day,ecl}}$ [W]</th>
<th>Volume [m$^3$]</th>
<th>Lifetime [years]</th>
</tr>
</thead>
<tbody>
<tr>
<td>GNC</td>
<td>341.3</td>
<td>275.7</td>
<td>65.6</td>
<td>-</td>
<td>391.2</td>
<td>1.7</td>
<td>10</td>
</tr>
<tr>
<td>CDH</td>
<td>382.6</td>
<td>317.1</td>
<td>65.5</td>
<td>-</td>
<td>431.4</td>
<td>1.91</td>
<td>10</td>
</tr>
<tr>
<td>Power</td>
<td>546.6</td>
<td>460.1</td>
<td>86.5</td>
<td>-</td>
<td>137.3</td>
<td>3.47</td>
<td>10</td>
</tr>
<tr>
<td>Mission$^1$</td>
<td>551</td>
<td>484</td>
<td>67.1</td>
<td>196</td>
<td>629.6</td>
<td>2.75</td>
<td>5</td>
</tr>
<tr>
<td>Mission$^2$</td>
<td>548.6</td>
<td>482</td>
<td>66.6</td>
<td>220</td>
<td>701.5</td>
<td>2.74</td>
<td>5</td>
</tr>
</tbody>
</table>

$^1$Without wireless power transfer, $^2$With wireless power transfer

Above fractionated infrastructure system results in a cluster of mini satellites where the mass of spacecraft range from 300 to 500 kg with respect to a one module per launch approach. This was favoured since the launch risk will be distributed across a number launches.

Regarding the results presented in above table and previous figures, it can be seen that the capability increase in terms of data handling is achieved with a less mass penalty. Increasing power distribution capabilities within the size limits of micro satellite scale, i.e. less than 100 kg, is very difficult since even a power module of 300kg can distribute around 500 W in total. If the mass is desired to be kept less than 300 kg then the feasibility of such power module could be questionable. However this should be investigated through the cost – benefit analysis of micro scale payload modules which consumes around 100 W of power.

Finally, with an infrastructure of approximately 1270 kg it is possible to provide 1300 W of power, a downlink data rate of 2 x 500 Mbps and 2 x 5.5 Gbps of data handling capability for the benefit of mission modules. In addition, although the mission modules were sized with respect to the given launch vehicle, they can still be launched with another launch vehicle with more mass capability and as bundles of small sized modules.
6. Conclusions

When compared to monolithic spacecraft, fractionated spacecraft may have additional costs due to its natural redundancy. However this architecture can be superior with a value centric design approach and it offers many advantages in terms of flexibility and robustness which were described in detail in the introduction section.

In Chapter 2, earth observation missions were considered and the value proposition was made through the sustainability of the infrastructure modules which provide additional resources for the mission modules, or payloads, in the spacecraft network. Associated orbit design considerations were also discussed and the absolute orbit for the infrastructure was provided.

For the fundamental goal of safe cluster navigation of fractionated spacecraft, several cluster flying configurations were discussed by using a previously developed method for close proximity LEO formation flying in Chapter 3. It was concluded that collision avoidance was possible though there is still need for manoeuvring capability to satisfy the collision avoidance condition by correcting the shifts on relative inclination and eccentricity vectors.

Several key technologies and realization approaches were discussed within the spacecraft architecture chapter through the shared resources which stems from the subsystems of a conventional satellite as guidance, navigation and control, communication, command and data handling, and power generation, storage and distribution. At the end of the Chapter 4 a notional fractionated spacecraft architecture was proposed in the light of the realization discussions for the benefit of earth observation payloads.

The optimum cluster configuration was investigated for 4 types of modules which include 3 types of infrastructure modules and the mission module with an incremental launch approach, i.e. one module per launch. The modules were sized such that the maximum allowable mass and volume was reached for the given launch vehicle while increasing the capability of a single module. At the end of the chapter it was concluded that the resource capability increase is more efficient in terms of data processing and storage when compared to wireless power transfer. In the end of the sizing, it was found out that the infrastructure is capable of providing 1300 W of power, 2 x 500 Mbps of downlink rate and 2 x 5.5 Gbps of data handling.

Apart from the above capabilities, this infrastructure requires ready to use satellite busses which enhances commoditization, responsiveness, flexibility and robustness. With the resource sharing ability and distributed architecture, there is a potential that these evolving fractionated spacecrafts will replace the large monolithic spacecrafts which have long term developments and are prone to single point failures.

However, to be able to assess the feasibility of such an infrastructure concept precisely, the economical cost – benefit analysis, or value centric design, should be performed through these attributes. Then the design outcome shall be the one with the lowest risk and maximum value. This would be the complementing part of this study to be reserved as a future work.

In conclusion, with this thesis, an insight to collision free cluster flying and technological aspects of fractionated spacecraft was provided. Referring to the system and technology analysis performed, a notional fractionated spacecraft infrastructure was proposed for the benefit of earth observation missions.
References


[2.3] ENVISAT homepage, European Space Agency (ESA), www.esa.int


[4.18] De Carlo, P. M., et. al., “Intersatellite link for Earth Observation Satellites constellation”, ASI.


[4.21] LoBosco, David M., et. al, “The Pleiades fractionated space system architectureand the future of national security space”


[4.27] Nugent, T. J., Kare, J. T., “Laser Power Beaming on a Shoestring”, LaserMotive LLC


Appendices
A. Orbit

A1. Orbit Design Algorithm

Sun-synchronous orbit design algorithm provided by [2,4]
A2. Cluster Orbit Acquisition

Considering that the launch vehicle brings the module(s) into the orbit with a certain altitude error and the first propulsive action can be taken after the commissioning phase, there will be certain along track and inclination separations between the cluster and the corresponding module. In this manner, the orbit acquisition strategy for in plane motion is to first stop the initial drift caused by the launch error and then applying the required control actions to acquire the desired along track position in the cluster formation. Therefore there are three steps for the in plane orbit acquisition which are first impulse for stopping the drift, second impulse for reaching the formation and final impulse to stop the motion caused by the second impulse. On the other hand, inclination change can be done with a single impulse.

The resulting equations for the orbit acquisition $\Delta V$ calculations can be given as follows:

$$\Delta V_T = \frac{v}{2a} \sum_{i=1}^{3} |\Delta a_i|$$

Where $\Delta a_i$ are the required relative semi major axis changes.

**In Plane:**

$$|\Delta a_1| = |\Delta a_{err}|$$

$$\Delta a_2 = -a \left( \Delta u_{target} - \Delta u_1 \right) \left( \frac{2a}{3v\Delta t} \right)$$

$$\Delta a_3 = -\Delta a_2$$

Where $\Delta T$ is the desired interval to accomplish the manoeuvre.

**Out of Plane:**

$$\Delta V_N = v|\Delta i|$$

where $a\Delta i = a \left\{ \Delta l_{sep} \Delta \Omega_{sep} \sin i \right\}$, $\Delta l_{sep}$ and $\Delta \Omega_{sep}$ are caused by the launch error.

**Propellant Mass**

Given the total $\Delta V$ budget for the formation acquisition and keeping, the total required propellant mass can be found from the below equation:

$$m_{prop} = m \left( 1 - e^{\frac{-\Delta V}{g\bar{i} \eta_{thr}}} \right)$$

Where $m$ is the module mass, $g$ is the gravitational acceleration, $\bar{i}$ is the specific impulse value for the corresponding propellant and $\eta_{thr}$ is the efficiency of the thruster.
### B1 DAICHI Payload Characteristics:

<table>
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<th><strong>PRISM Characteristics</strong></th>
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<tbody>
<tr>
<td>Observation Band (μm)</td>
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<tr>
<td>Number of Optics</td>
</tr>
<tr>
<td>Base / Height Ratio</td>
</tr>
<tr>
<td>Signal to Noise (S/N)</td>
</tr>
<tr>
<td>Spatial Resolution</td>
</tr>
<tr>
<td>Swath Width</td>
</tr>
<tr>
<td>Pointing Angle</td>
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<tr>
<td>Data Rate</td>
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<td>Data Compression</td>
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<table>
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<th><strong>AVNIR 2 Characteristics</strong></th>
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<td>Observation Band (μm)</td>
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<td>0.27</td>
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<td>Target Species</td>
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<td>CO₂, CH₄</td>
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<tr>
<td>CO₂, H₂O</td>
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<td>CO₂, CH₄</td>
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<td>Instantaneous Field of View</td>
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<td>10.5 km</td>
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<td>Single-scan data acquisition time</td>
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<td>Band 4: 1.56 – 1.68</td>
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</table>
C. GNC System Equations

Below equations are obtained from Table 11.12 and 11.13 within the textbook “Space Mission Analysis and Design (SMAD),” 3rd Edition.

Disturbance Torques

- Gravity Gradient:

\[ T_D = \frac{3\mu}{2R^3} |I_x - I_y| \sin(2\theta) \]

Where \( \mu = 3.986 \times 10^{14} \text{ m}^3/\text{s}^2 \) is the Earth’s gravity constant, \( R \) is the orbit radius, \( I \) is the moment of inertia and \( \theta \) is the maximum deviation from the local vertical in radians.

- Magnetic Field:

\[ T_D = D \cdot B \]

Where \( D \) is the residual magnetic dipole of the spacecraft in [A. m²] and \( B \) is the Earth’s magnetic field in [Tesla]. \( B \) can be approximated for a polar orbit as

\[ B = \frac{2M}{a^3} \]

Where \( M = 7.96 \times 10^{15} \text{ [Tesla.m}^3\text{]} \) and \( a \) is the semi-major axis in [m]

GNC System Sizing Equations

- Slew manoeuvre torque requirements:

\[ T = 4\theta I / t^2 \]

Here, \( T \) is the required torque for a desired angle of \( \theta \) which is to be achieved within the desired time interval \( t \) and \( I \) is the moment of inertia of the spacecraft with respect to the particular axis of motion.

- Angular momentum requirements:

The angular momentum storage of the reaction wheel caused by the disturbance torques is averaged over an orbital period and given as follows:

\[ h = T_D \frac{\text{Orbital Period}}{4} (0.707) \]

where \( T_D \) is the disturbance torques.
The momentum storage on a momentum wheel with respect to accuracy requirement can be written for an allowable angular motion of $\theta_a$ in radians as:

$$h = \frac{T_D \cdot \text{Orbital Period}}{4 \cdot \theta_a}$$

- Thrusters sizing:

In case of a momentum damping:

$$F = \frac{h}{Lt}$$

Where $h$ is the stored momentum on reaction wheels, $L$ is the moment arm and $t$ is the burn time.

Then the amount of propellant for momentum dumping can be calculated with respect to the below equation:

$$M_{p, attitude} = \frac{\text{Total Impulse}}{I_{sp} \cdot g}$$

- For the orbit control, the total amount of propellant can be calculated with respect to a particular $\Delta V$ as:

$$M_{propellant} = = M_{S/C} \left(1 - e^{-\frac{\Delta V}{I_{sp} \cdot g}}\right)$$
D. Link Design
Given: Altitude/distance, bit rate, bit error rate, modulation, ground station min. elevation angle, frequency, transmitter antenna efficiency, noise temperature at receiver, Boltzmann constant
Achieved: Bandwidth, Max distance, max free space path loss, receiver and transmitter antenna characteristics (diameter, opening angle, gain and power), \( (C/N)_{\text{achieved}} \)

- Bandwidth with Quadrature Phase Shift Keying (QPSK) digital modulation:
  \[
  B_{\text{QPSK}} = 0.5 \frac{R}{2} \text{ where } R \text{ is data rate } [4.17]
  \]

- Required Received Power:
  \[
  \left( \frac{C}{N} \right)_{\text{req}} = \frac{R}{B} \ln \left( \frac{1}{2\text{BER}} \right)
  \]
  Where \( C/N \) is the carrier to noise ratio and BER is the bit error rate

- Achieved Received Power:
  \[
  \left( \frac{C}{N} \right)_{\text{ach}} = \frac{P_tG_tG_r}{kTB} \left( \frac{\lambda}{4\pi d} \right)^2
  \]
  Where
  \( P_t \): Transmitter power
  \( G_t \): Transmitter gain
  \( G_r \): Receiver gain
  \( \lambda \): signal wavelength
  \( k \): Boltzmann constant
  \( T \): Noise temperature
  \( L \): Link losses
  \( d \): max. distance between transmitter and receiver

- Gain and Beamwidth:

<table>
<thead>
<tr>
<th>Antenna</th>
<th>Application</th>
<th>Gain</th>
<th>( \Theta_{3\text{dB}} ) (deg)</th>
<th>Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>Parabolic</td>
<td>Narrow Beam</td>
<td>( \eta(\pi D/\lambda)^2 )</td>
<td>70° ((\lambda/D) )</td>
<td>D: diameter</td>
</tr>
<tr>
<td>Helix</td>
<td>Wide Beam (below 4 GHz)</td>
<td>15.3 ( \frac{L}{\lambda} ) ( (\pi D/\lambda)^2 )</td>
<td>16.5° ((\lambda/D) \sqrt{\lambda/L} )</td>
<td>D: diameter, L: length</td>
</tr>
<tr>
<td>Circular Horn</td>
<td>Wide Beam (above 4 GHz)</td>
<td>( \eta(\pi D/\lambda)^2 )</td>
<td>70° ((\lambda/D) )</td>
<td>D: diameter</td>
</tr>
<tr>
<td>Biconical</td>
<td>Omni-directional</td>
<td>0 - 5 dB, ( \eta=0.65 )</td>
<td>60° - 360°</td>
<td>-</td>
</tr>
</tbody>
</table>
E. Structure Equations

- Mass Moment of Inertia for a Cubic Body

\[ I_{x,y,z} = \frac{1}{6} m_{\text{gross}} s^2 \]

Where “s” is the linear dimension of the cube.

- Moment of Inertia estimation for Solar Arrays

Solar array offset: \( I_a = 1.5 s + 0.5 \left( A_{SA}/2 \right)^{1/2} \)
Perpendicular to array face: \( I_{ax} = m_{SA}(L_a^2 + A_{SA}/12) \)
Perpendicular to array axis: \( I_{ay} = m_{SA}(L_a^2 + A_{SA}/24) \)
About array axis: \( I_{aa} = m_{SA} \left( A_{SA}/24 \right) \)

Where \( m_{aa} \) and \( A_{aa} \) are the solar array mass and area.

- Area moment of inertia for the launch configuration

\[ I_{LV} = \frac{1}{12} b h^3 + A d^2 = \frac{1}{12} s^4 + s^2 (s/2)^2 = \frac{s^4}{3} \]

- Fundamental Frequencies for Launch configuration:

\[ f_{\text{lateral}} = 0.56 \sqrt{\frac{EI_{LV}}{m_{\text{gross}} L^3}} \]
\[ f_{\text{axial}} = 0.25 \sqrt{\frac{AE}{m_{\text{gross}} L^3}} \]

Where \( A \left( = s^2 \right) \) is the cross sectional area of the spacecraft, \( L \left( = s \right) \) length of the spacecraft and \( E \) is the Young’s modulus, \( E = 71 \times 10^9 \) [N/m²]
F. Sizing Code