VALUE-CENTRIC DESIGN ARCHITECTURE BASED ON ANALYSIS OF SPACE SYSTEM CHARACTERISTICS

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Acronyms and Abbreviations

ACS Attitude Control Subsystem.

ADR Active Debris Removal.

AOCS Attitude and Orbit Control Subsystem.

CBO Congressional Budget Office.

CDF Cumulative Distribution Function.

CDH Command and Data Handling.

CERs Cost Estimating Relationships.

Com Communication mission.

COTS Commercial Off-The-Shelf.

DARPA Defense Advanced Research Projects Agency.

DE Differential Evolution.

DoD Department of Defense.

EOL End of Life.

EP European Parliament.

EPS Electrical Power Subsystem.

ESA European Space Agency.

EU European Union.

FOC Full Operational Capability.

FY Fiscal Year.

GA Genetic Algorithm.

GAO Government Accountability Office.

Gen General mission.

GEO Geostationary Earth Orbit.

GNSS Global Navigation Satellite System.

GPS Global Positioning System.

GS Global Search.

HST Hubble Space Telescope.

Hyb Hybrid mission.

IAC International Astronautical Congress.

IADC Inter-Agency Space Debris Coordination Committee.

IAF International Astronautical Federation.

IAT Integration, Assembly, and Test.

INCOSE International Council on Systems Engineering.

ISS International Space Station.

ISTS International Symposium on Space Technology and Science.

JAXA Japan Aerospace Exploration Agency.

JPL Jet Propulsion Laboratory.

LEO Low Earth Orbit.

LX Laplace Crossover.

M&V Mariner and Voyager.

MCMC Bayesian theory combined with Markov Chain Monte Carlo simulations.

MEO Medium Earth Orbits.

MILP Mixed Integer Linear Programming.

MINLP Mixed Integer Nonlinear Programming.

MIPP Mixed Integer Programming Problem.

MLE Maximum Likelihood Estimation.

MSL Mars Science Laboratory.

NASA National Aeronautics and Space Administration.

Nav Navigation mission.

Obs Observation mission.

OCS Orbit Control Subsystem.

ORS Operationally Responsive Space.

PDF Probability Density Function.

PLS Payload Subsystem.

PM Power Mutation.

PnP Plug-and-Play.

PSO Particle Swarm Optimization.

RAAN Right Ascension of the Ascending Node.

RCD Requirement-Centric Design.

RE RapidEye.

SA Simulated Annealing.

Sci Science mission.

SCS Supplemental Communications System.

SMAD Space Mission Analysis and Design.

SSAD Small Satellite Anomalies Database.

SSCM Small Satellite Cost Model.

SSTL Surrey Satellite Technology Ltd.

STD SpaceTrak Database.

STR Spacecraft Structure.

STS Space Transportation System.

System F6 Future Fast, Flexible, Fractionated, Free-Flying Spacecraft.

TCS Thermal Control Subsystem.

TDRS Tacking and Data Relay Satellites.

Tec Technology demonstration and development mission.

TFU Theoretical First Unit.

TJSASS Transactions of the Japan Society for Aeronautical and Space Sciences.

TTC Telemetry, Tracking and Command.

UCS Union of Concerned Scientists.

USCM Unmanned Spacecraft Cost Model.

VCD Value-Centric Design.

VDD Value-Driven Design.

WSS Overhead of Fractionation and Wireless Sharing Subsystem.

Nomenclature

α	Significance Level
β	Dimensionless Shape Parameter of Weibull Distribution
ΔV	Change in Velocity
η	Time Scale Parameter of Weibull Distribution
$\Gamma(n)$	Gamma Function
μ	Geocentric Gravitational Constant
$\overline{p}$	Mean of PDF
σ	Variance of PDF
A	System Arranging Matrix
$a_{j,ij}$	Element of A_j in Row i, Column j
A_j	Subsystem Arranging Matrix of Subsystem j
$A_j(k)$	k^{th} Column Vector of Matrix A_j
a_t	Semi-major Axis of Transfer Orbit
C	Subsystem Cost Vector
$c_{\text{IAT},f,i}$	IAT Cost of Fractionated Module i
$c_{\text{IAT},m}$	IAT Cost of Monolithic System
c_a	Average Unit Cost

c_d Expected Development and Deployment Cost

c_e Expected Launch Cost

c_j Element of C in Column j

C_l ADR Spacecraft Launch Cost

c_l Launch Cost

C_p ADR Mission Cost

$c_{r,f}$ Fractionated System Replenishment Cost

$c_{r,m}$ Monolithic System Replenishment Cost

C_{rd} ADR Spacecraft De-orbit Package Development Cost

C_{rs} ADR Spacecraft Development Cost

c_s Cost of Launched Satellites

c_{T1} Theoretical First Unit Cost

c_w Cost of WSS

D System Design Matrix

d_{ij} Element of D in Row i, Column j

F System Fractionation Vector

f Launch Failures

$f_{\mathrm{A}}(a)$ PDF of Successful Launches

f_j Element of F in Column j

g_0 Gravitational Constant at Earth's Surface

I_{sp} Specific Impulse

$J(x)$ Objective Function

m	Number of Satellites
m_{fuel}	Mass of Spacecraft Propellant
m_{kit}	Mass of Spacecraft De-orbit Kit
m_{plat}	Spacecraft Dry Mass Excluding Tank and Kits
m_{tank}	Mass of Spacecraft Tank
m_f	Final Mass or Dry Mass of a Spacecraft
m_i	Initial Mass or Launch Mass of a Spacecraft
N	System Configuration Matrix
n	Number of Subsystems
$N(k)$	k^{th} Column Vector of Matrix N
n_e	Number of Expected Launches
$n_{f,i}$	Number of All Possible Interfaces of Fractionated Module i
n_{ij}	Element of N in Row i, Column j
n_j	Number of Arranging of Subsystem j
P	System Duplication Vector
p_j	Element of P in Column j
Q	System Sequence Vector
q_j	Element of Q in Column j
R	Subsystem Reliability Vector
r	Reliability of Launch Vehicle Family
$R(t)$	Cumulative Distribution Function of Reliability
$r(t)$	Probability Density Function of Reliability

$R_{\mathrm{SSAD},j}(t)$ Reliability CDF of Subsystem j in SSAD

$R_{\mathrm{STD},j}(t)$ Reliability CDF of Subsystem j in STD

$R_{EOL}(t)$ EOL Reliability of the Entire System

$R^k_{f,j}(t)$ Reliability CDF of Subsystem j at Degree of Fractionation of k

r_f Final Orbit Radius

r_i Initial Orbit Radius

r_j Element of R in Column j

$R_j(t)$ Reliability CDF of Subsystem j

$R^k_{p,j}(t)$ Reliability CDF of Subsystem j at Degree of Duplication of k

$R_{v,j}(t)$ Reliability CDF of Subsystem j at Degree of Derivation of v_j

$R_{w,j}(t)$ Reliability CDF of WSS for Subsystem j

r_w Reliability of WSS

s Launch Successes

$S(n, k)$ Stirling Numbers of Second Kind

t Satellite Lifetime in Orbit before Failure

U Unit Design Property Matrix

u_{ij} Element of U in Row i, Column j

U_k Unit Design Property Matrix of Generation k

V System Derivation Matrix

v_{ij} Element of V in Row i, Column j

V_k System Derivation Matrix of Generation k

x State Variable

x_i	i^{th} Element of State Variable
x_i^l	Lower Bound of i^{th} Element of State Variable
x_i^u	Upper Bounds of i^{th} Element of State Variable

ABSTRACT OF THESIS submitted by **Qin Xu** for the degree of Doctor of Philosophy at The University of Manchester, entitled "**Value-Centric Design Architecture Based on Analysis of Space System Characteristics**".

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Emerging design concepts such as miniaturisation, modularity, and standardisation, have contributed to the rapid development of small and inexpensive platforms, particularly CubeSats. This has been stimulating an upcoming revolution in space design and development, leading satellites into the era of "smaller, faster, and cheaper New Space".

However, the traditional requirement-centric design methodologies focus on large, complex, and customised systems. The associated labour-intensive development and production process typically spends considerable time and money on the integration and testing. This does not inherently fit with the innovative modular, standardised concepts, and the incorporation of mass-produced technologies that newer and smaller satellite classes are considering. Therefore, there is a significant potential benefit in establishing and adopting a new design architecture to effectively solve the problems rooted in the traditional methodologies and deliver innovative capabilities. This research presents a new categorisation, characterisation, and value-centric design architecture to address this need in both traditional and novel system designs.

Based on the categorisation of system configurations, a characterisation of space systems is proposed, comprised of the degree of duplication, fractionation, and derivation. The three primary characteristics capture the overall configuration features, thus potential hybrid designs are promoted to improve performance or reduce cost.

With the formulation of this characterisation, a value-centric design architecture for the design and development of a wide range of space systems is established. This architecture enables the use of both traditional and innovative technologies, acting as a systematic guideline for quantitative system design and analysis.

The function of the design space is to integrate the cost or intrinsic properties, e.g., mass, reliability, and orbit, from subsystem level to system level, based on configuration designs. Through applying appropriate value models, these properties can be measured in the singular monetary dimension. Different properties can be used for the cost modelling of different lifecycle phases of a space system, e.g., development, launch, operation, and retirement phases. The sum of the costs of these four lifecycle phases, i.e., development, launch, replenishment, and disposal costs, can be further applied as the comprehensive objective function to enable an optimization process of design configurations to minimise the entire lifecycle costs. Thus, different system properties or design requirements can be converted into a standardised dimension, solving the design selection problems by turning the multi-objective optimization into the single-objective optimization.

This design architecture embraces the innovative design concepts of modularity and standardisation, and the use of commercial off-the-shelf (COTS) products. In this condition, the design and optimization of system configurations are realised through the design and optimization of the combination and permutation of standard subsystems or COTS products. Therefore, lowering the difficulties and decoupling the requirements in designing space systems. Meanwhile, this architecture is also applicable to the spacecraft design using traditional design concepts.

Declaration

No portion of the work referred to in the thesis has been submitted in support of an application for another degree or qualification of this or any other university or other institute of learning.

Copyright Statement

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In memory of my elapsed youth!

Chapter 1

Introduction

Innovative design concepts and technologies such as modularity, standardisation, and Commercial Off-The-Shelf (COTS) products have enabled the rapid development of small and inexpensive spacecraft platforms, such as CubeSats. This has been stimulating a revolution in spacecraft design and development, leading satellites into the era of the “New Space” [1], “Space 4.0” [2–4], or “Operationally Responsive Space (ORS)” [5–7]. However, the traditional requirement-centric design philosophy and the corresponding labour-intensive and one-off fabrication process do not inherently fit with these emerging concepts and technologies.

Therefore, a new design paradigm for spacecraft design and analysis is addressed in this chapter to deliver such capabilities effectively in both traditional and novel system designs. In order to achieve this aim, three concrete research objectives are further proposed. Moreover, the structure of this thesis and the related publications are also presented.

1.1 Space System Design

Emerging technologies derived from the electronics industry, such as widely-used micro- and nanotechnology, offer significant opportunities for the miniaturisation of space systems [8]. The concept of modularity associated with commercialisation, not only completely changes the labour-intensive and bespoke nature of the existing space industry by promoting standardisation [9], but also drives the reduction of design, integration and testing time, and associated costs [10]. Thus, these innovative concepts and technologies have enabled the rapid development of small and inexpensive spacecraft platforms, particularly CubeSats, stimulating an upcoming revolution in spacecraft design and development.

New design concepts have also promoted significant breakthroughs in system capabilities, e.g., modularity, standardisation, and fractionation. Some even overturn the conventional wisdom of space design and development. Distributed systems refer to a cluster of satellites cooperating as a virtual satellite in the form of spatial separation, with each sharing the communication, processing and payload [11]. These distributed systems offer an ideal carrier for modularity, mass production and the use of COTS products. Such a configuration is considered to be flexible, robust, and cost-effective throughout the lifecycle [12], filling the vacancies of flexibility and robustness in traditional systems and creating additional value for space systems.

To enable the use of these design concepts and technologies, innovative space paradigms have been addressed. The “New Space” [1], is a novel paradigm for aerospace companies cooperating to develop “smaller, faster, and cheaper” [13–15] satellites, which is primarily driven by commercial motivations. The “Space 4.0” [2–4] refers to a potential revolution for space applications to be safe, secure, and easily, readily, economically, and sustainably accessible over time. The “ORS” [5, 6] was proposed by the Department of Defense (DoD) [7] to loosen the time constraints of spacecraft design and keep updated with ever-changing mission demands, environmental conditions and advanced technologies.

Since these novel paradigms have been the latest driving factors for spacecraft design, new analysis and design methodologies are urgently required to deliver the above innovative capabilities effectively. However, the current requirement-centric design methodologies, focused on designing a bespoke monolithic spacecraft with complex structures, do not naturally fit with the innovative modular, standardised, and mass-produced technologies.

Due to the small quantity and bespoke architecture, certification through production has become a typical feature of traditional spacecraft development and production processes. To ensure the reliability, considerable time and money is invested in the integration and testing of large, complex, and customised systems [16–18]. Such labour-intensive design architecture results in a number of inherent problems, such as capability uncertainty, cost overruns, and schedule delays.

Therefore, a new design architecture for spacecraft design and analysis is required to effectively solve the above problems of traditional design methodologies and deliver the capabilities provided by both traditional and innovative system designs.

1.2 Thesis Aims and Objectives

Given the motivating factors discussed previously, the aim of this research is the development of a design architecture to improve the current practice of system configuration design during the concept phase, which solves the problems inherent in the traditional design methodologies and provides a vehicle for both traditional and innovative design concepts and technologies. To achieve this aim, it is necessary to integrate the quantitative analysis of system configurations into the design process of space systems. Fundamental to this endeavour will be the formulation of system design configurations and the establishment of an appropriate design space.

The development of the architecture for the integrated design of system configurations enables the analysis of system configurations as part of the overall design process. The exploration of the design space helps understand the relationship between system design configurations and selected value or property objectives, providing supporting knowledge for the decision making process. By applying optimization methods, the optimal set of solutions can be identified to achieve improved system configuration designs. This allows the design process to be driven by system design objectives, and overturns the labour-intensive and customised design process utilised in traditional design methodologies, which rely on the manual selection of appropriate alternatives to meet system requirements.

More specifically, the development of this design architecture can be implemented by the following research objectives, forming the individual contributions of this research:

Research Objective I: To develop a methodology which supports the configuration design of a wide range of space systems in the concept phase.

Through the formulation of the design configurations of different space systems, this methodology can be used to analyse and evaluate their characteristics. Optimization methods can also be applied to explore the design space and identify the optimal set of solutions within the feasible domain.

Research Objective II: To enable the integration of the system configuration design into the wider design process of space systems.

Through the analysis of the intrinsic properties or value of space systems, the mapping relations between system design configurations and intrinsic properties or value can be established. This integration ensures the design of space system configurations to be considered during the overall design process and to support the exploration of the design space for the top-level design parameters of system properties or value.

Research Objective III: To apply optimization methods to perform the exploration of the design space of system design configurations and identify improved design configurations.

In order to solve the problems of manual and customised design processes in the current requirement-centric design methodologies, the proposed design architecture allows the application of optimization methods to seek the optimal set of design solutions. Since both traditional and innovative design concepts are considered, the optimization solutions may provide new insights for space system designs.

1.3 Thesis Structure

This research presents a categorisation, characterisation, and value-centric design architecture to solve the problems rooted in traditional spacecraft design methodologies, and to provide a vehicle for both traditional and innovative design technologies. The overall structure of this thesis is organised as shown in Fig. 1.1. There are 7 chapters in the thesis, where Chapter 3 defines and describes the qualitative analysis process and Chapter 4, 5, and 6 develop and implement the quantitative analysis methodology using the proposed value-centric design architecture.

Chapter 2 provides a review of the current trends and the promising technologies, which may stimulate an upcoming revolution in spacecraft design and development. However, the drawbacks inherent in the traditional requirement-centric design philosophy are found to be at odds with these trends and innovative technologies. More specifically, the main design constraints and effects of different lifecycle phases are

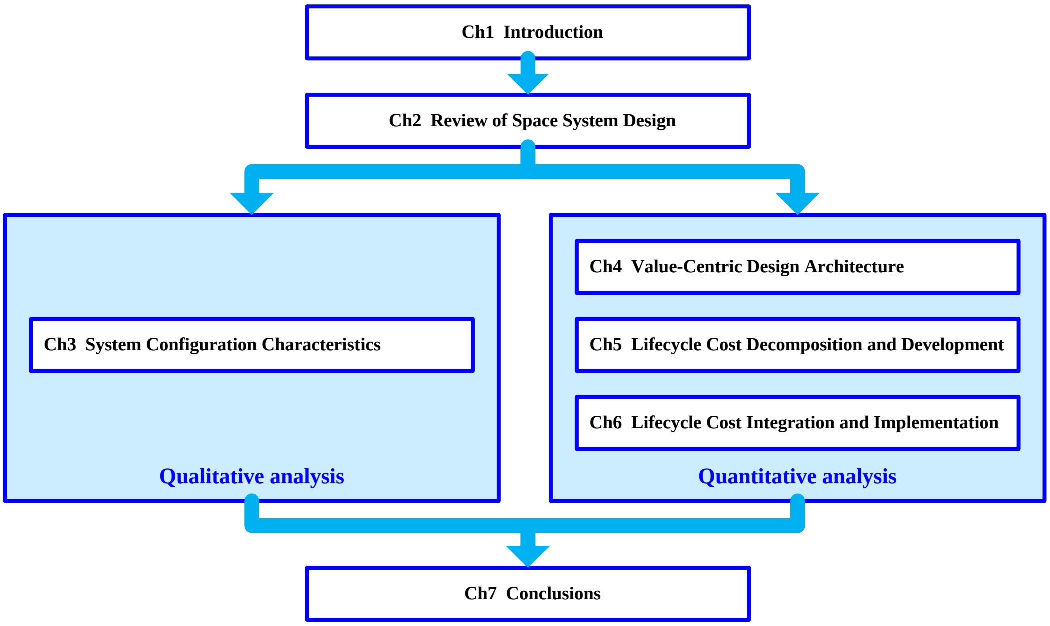


Fig. 1.1: Thesis structure

analysed, which significantly influence the system designs and are the key causal factors for problems associated with traditional design methodologies.

Chapter 3 presents a new categorisation and characterisation for both traditional and innovative space system designs. Considering the features of different design configurations, the categorisation of space system is proposed, consisting of monolithic spacecraft, constellations of identical spacecraft, fractionated spacecraft, and hybrid spacecraft.

This acts as a basis to develop the system characterisation, i.e., the system characteristic space. The system characteristic space is comprised of the degree of duplication, fractionation, and derivation. These three characteristics represent different system qualities, thus design configurations can be effectively explored using the proposed design space. The definitions of the three characteristics are illustrated by a qualitative analysis of a series of typical space missions. Therefore, the connection between system designs and configuration characteristics is built, enabling the exploration of space mission concepts.

A value centric design architecture for designing and developing a wide range of space systems is established in Chapter 4, based on the general philosophy of value-centric design. This value-centric design architecture enables the quantitative analysis of design configurations, consisting of four key value-centric design processes, i.e., the

elaborate, **analyse**, **evaluate** to **optimize** process. Under this architecture, the characterisation is formulated into a design space to support the formulation of space system design configurations.

The function of the formation of the characterisation is to integrate subsystem cost or intrinsic properties, e.g., mass, reliability, and orbit, into system cost or properties, through system design variables, e.g., the degree of duplication, fractionation, and derivation. Applying appropriate value models, these properties can be measured in the singular monetary dimension.

Through the establishment of the required cost models, Chapter 5 formulates different lifecycle phase costs respectively, i.e., development, launch, replenishment, and disposal costs. Each of the four lifecycle costs corresponds to a lifecycle phase, i.e., development, launch, operation, and retirement phases, as illustrated in Fig. 1.2. The lifecycle phase costs are further adopted as the objective function to develop the analysis and optimization methodology of system design configurations.

To be specific, in Section 5.1, the development cost quantifies the costs in the development phase, with subsystem cost models as inputs. Using the system characteristic space, subsystem costs are integrated into an overall system cost, which measures the efforts made in the development phase. The learning curve factor is considered, when a subsystem is reproduced.

In Section 5.2, the launch cost formulates the costs of the launch phase, with subsystem cost and mass properties as inputs. Real data of various launch vehicle

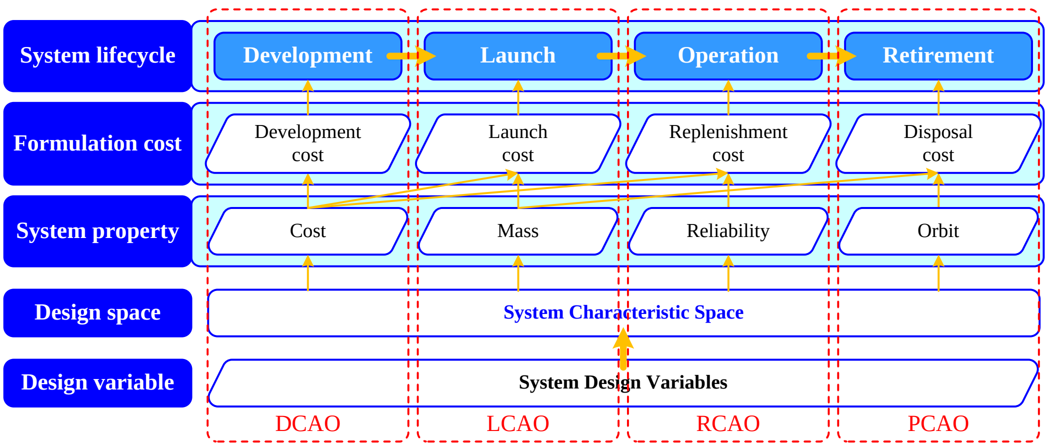


Fig. 1.2: Framework of lifecycle cost decomposition and integration (The abbreviation used are respectively DCAO, LCAO, RCAO, PCAO for development, launch, replenishment, and disposal cost analysis and optimization.)

families, such as cost, total attempts, and success rate, are initially collected to develop a launch vehicle database. Based on this database, we estimate the reliability of each launch vehicle family, using a modified two-level Bayesian analysis. The factors of both launch cost and reliability are subsequently merged into the expected launch cost, acting as the metric to evaluate the cost of launch activities.

In Section 5.3, the replenishment cost estimates the running costs of a space system in the operation phase, with subsystem cost and reliability properties as inputs. The modelling of subsystem reliability associated with the cost models in the development phase, are merged into the expected replenishment costs, which are the metric to evaluate the replenishment activities.

In Section 5.4, the disposal cost assesses the costs of the retirement of a space system. For the space systems with EOL de-orbit or orbital lifetime reduction devices, it is estimated by the development costs of the devices in the development phase. For the systems without these devices, it is estimated by calculating the costs of building and launching the spacecraft to send the de-orbit packages to implement active removal missions. To minimise the costs, the change in velocity is optimized for single orbital transfer between any two targets, while the optimal combination and permutation strategy is developed for the missions of multiple targets.

Overall, in Chapter 6, the analysis and optimization methodologies developed for each lifecycle phase are implemented in Section 6.1 to illustrate how to minimise different lifecycle phase costs respectively. Based on the knowledge of these lifecycle phases, the methodology of analysing and optimizing the entire lifecycle costs of a space system is integrated in Section 6.2, by adopting the sum of the above four costs as the objective function. The mission cases of the Galileo and RapidEye constellation are studied to exemplify the concrete solution process. Thus, different system properties or design requirements are converted into a standardised, comparable, and comprehensive metric for the design of system configurations to minimise the entire lifecycle costs. This copes with the design selection problems by turning the multi-objective optimization process into the single-objective optimization one.

Finally, in Chapter 7, the conclusions of the development of the proposed design architecture are stated, and the recommendations for the future work are also presented.

1.4 Publications

The contents of some chapters and sections of this thesis have previously been published in conference proceedings or as journal articles.

A new categorisation, characterisation, and value-centric design architecture were presented at the 67th International Astronautical Congress (IAC) [19], and subsequently published in Acta Astronautica [20]. In this paper, the categorisation of different system configurations, presented in Section 3.1, was proposed as a basis to develop the characterisation, presented in Section 3.2. Based on the characterisation, the qualitative and quantitative analysis of different system configurations were implemented, respectively presented in Section 3.3 and Chapter 4.

Based on the launch vehicle database established at the 67th IAC [21], an analysis and optimization of system launch costs was presented at the 31st International Symposium on Space Technology and Science (ISTS) [22], and subsequently submitted to the Transactions of the Japan Society for Aeronautical and Space Sciences (TJSASS) [23]. In this paper, the methodology of analysing and optimizing the launch costs of a space system was illustrated based on the developed value-centric design architecture, as presented in Section 5.2.

As a complement, a review of different life cycles for large and complex systems, seen in Appendix A, was presented at the 31st ISTS [24] and accepted by the TJSASS [25].

In summary, the publications are listed as follows:

[19] Q. Xu, P. Hollingsworth, K. Smith, and W. Zheng, "Space System Concept Design: A Value-Centric Architecture Based on System Characteristic Space," in 67th International Astronautical Congress (IAC), 2016, pp. 1–12.

[20] Q. Xu, P. Hollingsworth, and K. Smith, "Value-Centric Design Architecture Based on Analysis of Space System Characteristics," Acta Astronaut., vol. 144, pp. 69–79, 2018.

[21] Q. Xu, M. Zhang, Z. Hao, P. Hollingsworth, and K. Smith, "Small Satellite Launch Opportunity: Statistical Analysis and Trend Forecast," in 67th International Astronautical Congress (IAC), 2016, pp. 1–10.

[22] Q. Xu, P. Hollingsworth, and K. Smith, "Space System Launch Cost Analysis: A

Value-Centric Architecture Based on System Characteristic Space," in 31st International Symposium on Space Technology and Science, 2017, pp. 1–8.

[23] Q. Xu, P. Hollingsworth, and K. Smith, "Launch Cost Analysis and Optimization Based on Analysis of Space System Characteristics," Trans. Jpn. Soc. Aeronaut. Space Sci., p. 1–9, under review.

[24] Q. Xu, P. Hollingsworth, and K. Smith, "A Value-Centric Design and Certification Architecture for Innovative Space Systems," in 31st International Symposium on Space Technology and Science, 2017, pp. 1–9.

[25] Q. Xu, P. Hollingsworth, and K. Smith, "A Value-Centric Design and Certification Architecture for Space Systems," Trans. Jpn. Soc. Aeronaut. Space Sci., p. 1–10, accepted for publication.

Chapter 2

Review of Space System Design

In the recent years, worldwide launches of small satellites have been witnessed [26–28], stimulating the continuous growth of the space market [29, 30]. This is primarily driven by the rapid development of emerging design concepts such as miniaturisation, modularity, standardisation, and fractionation. Therefore, it is necessary to understand the benefits and characteristics of innovative concepts in the system design. This chapter starts with a review of the current state of space design concepts and technologies.

The design of the smaller satellites promotes the use of cheaper components with shorter design and development cycles, e.g., COTS products. However, the traditional design paradigms do not inherently fit with the design of these "smaller, faster, and cheaper" satellites. This chapter analyses the problems of the traditional design methodologies, and concludes with a potential solution approach for both traditional and innovative space system designs.

The traditional design problems generally appear in the process of turning system formulation into implementation. Thus, the particular constraints of system designs and the significant effects of different lifecycle phases in the system implementation are necessary to be included in the proposed approach. This chapter also addresses this need to summarise the major design constraints and effects for different lifecycle phases that can significantly influence the system designs and cause the typical problems in traditional designs.

2.1 Design Concept and Technology Development

Supported in part by the continuous advancements in micro-technologies that have enabled the miniaturisation of electronic systems, a boom of small satellites has been witnessed in the past two decades [26–28], driving growth in the space market [29, 30].

Following the classification of the International Astronautical Federation (IAF) [31], small satellites are the class of satellites weighing less than 1000kg, while large satellites are more than 1000kg. As shown in Fig. 2.1, the Union of Concerned Scientists (UCS) satellite database [32] reveals that the number of small satellites launched has increased sharply since 2012, while the number of large satellites launched has remained relatively constant. Thus, the rapid development of small satellites has become the enabling factor for the constant increase of the total number of satellites launched. Moreover, the State of the Satellite Industry Report 2016 [29] also pointed out the continued and growing global interest and expenditure in inexpensive platforms.

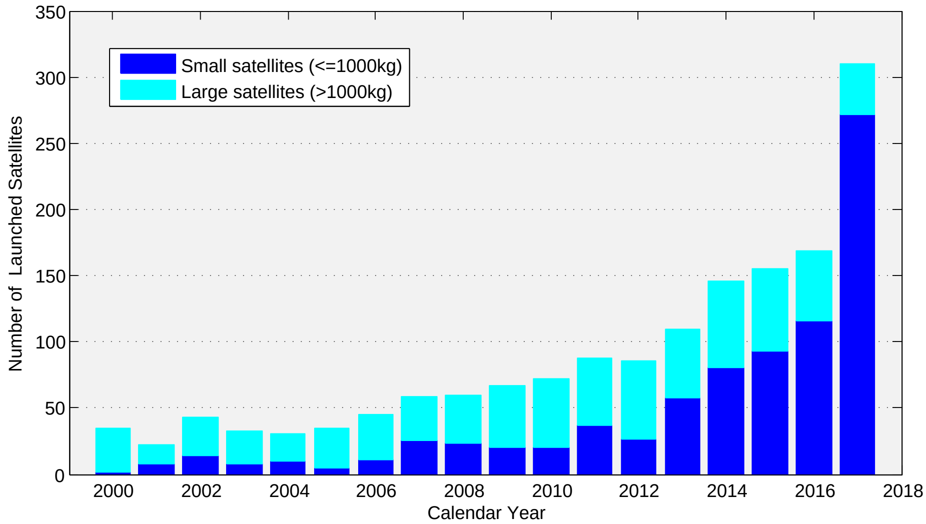


Fig. 2.1: Recent launch history of the satellites recorded in the UCS satellite database [32]

Therefore, space systems are facing an upcoming revolution of reducing size, for equal or increased capabilities and cost competitiveness. This does not mean that the traditional large satellites will be completely discarded. The advantages of traditional large satellites can still be utilised, particularly for long duration and high reliability space missions, while innovative small satellites can offer another way of realising space applications. Such a recognition has been strengthened after the successes of a series of ambitious small satellite missions.

For example, the CanX-2 was the first to deliver GPS navigation fixes and raw measurement [33] on a nanosatellite platform. This is enabled by advanced attitude control and on-board processing systems. The SNAP-1 mission has demonstrated that the critical capabilities of future nanosatellites can be achieved as successfully as their large competitors, e.g., GPS-based orbit determination, 3-axis attitude control, and automated orbital manoeuvres [34].

The QB50 mission to date has launched 36 of intended 50 low-cost CubeSats [35] to address the fundamental scientific questions about the lower thermosphere, controlled re-entry, formation flying, etc [36]. The Flock constellation is the largest constellation of Earth imaging missions, which is performed by a series of nanosatellites capable of 3-5m ground resolution [37].

Simultaneously, the demonstration and development of solar sail propulsion systems [38] for small satellites were fulfilled and realised with the successes of NanoSail-D [39] and NanoSail-D2 [40]. Moreover, the Jet Propulsion Laboratory (JPL) identified six critical technologies integrated into small, low-cost interplanetary missions [41], extending the current capabilities of CubeSats to space applications beyond low Earth orbits.

Apart from these technical breakthroughs, emerging design concepts such as modularity and fractionation [42] have also been proposed, enabling numerous opportunities for the development of small satellites [43].

Modularity, or modular design, is distinguished from the traditional design for its two distinctive characteristics of establishing standardised interfaces among elements and reusing functionally decoupled modules [44]. These characteristics decrease the complexity and increase the reliability of space systems, which lead to the direct reduction of required labour from decreased complexity and the indirect savings from increased reliability. Thus, it is believed to be a key factor for the success of any multi-objective and multi-customer product [45]. In spite of having achieved great successes in some areas such as personal computers, modularity has not drawn much attention from the space industry [44].

Introducing modularity into spacecraft design decouples the design requirements of different subsystems. It can effectively overcome the drawbacks inherent to traditional designs, such as subsystem anomalies, cost variation, and over-complex structures

[10]. Furthermore, the Plug-and-Play (PnP) philosophy of modular small satellites, similar to that of personal computers, has also been promoted to develop a simple but functional network design architecture [46].

Such design philosophy embraces the design concepts of mass production and COTS [47] products. Mass production has completely changed modern lifestyles, through the utilisation of assembly lines to produce large quantities of standardised products [48]. One of the most appealing reasons for introducing mass production into spacecraft engineering is probably its cost saving characteristic. Having successfully brought automobiles, personal computers, and mobile phones into everyone's life, mass production is likely to be one of the enabling factors of reducing space manufacturing cost and time and incorporating the enhanced reliability.

COTS products have already been widely-used in small and inexpensive platforms, particularly in increasing subsystem capabilities and cost-effectiveness [8, 49] in CubeSats such as GeneSat-1 [50] and O/OREOS [51]. The use of COTS products can lower the difficulty and decouple the requirements in designing a satellite. In return, much of the integration and testing time and cost can be saved, shortening the cycle of spacecraft designs. Thus, the common problems associated with traditional design methodologies, such as cost overruns and schedule delays, can be effectively prevented.

If innovative design concepts such as modularity, mass production, and utilisation of COTS products, could be appropriately introduced and fully developed in the space industry, it is likely to realise the dream of making satellites more accessible and affordable to develop.

The concept of distributed systems, namely, a cluster of satellites cooperating as a virtual satellite in the form of spatial separation [11], is a new system configuration that is generally adopted to increase capabilities, reduce costs, or fulfil a mission that cannot be realised by a monolithic system. A good example of this is missions requiring contemporaneous global coverage.

There are two types of distributed systems: homogeneous and heterogeneous systems. Homogeneous systems are generally called satellite constellations, where multiple identical satellites are designed to perform a mission in a collaborative way [52]. Heterogeneous systems are often termed fractionated spacecraft, where different modules are designed with different functions flying close together to implement a mission through shared resources [53].

Distributed systems offer another innovative configuration for small satellites and an ideal carrier for modularity and COTS products, especially fractionated spacecraft. Through decomposing monolithic spacecraft into fractionated modules, each module can be designed with modularity and developed with COTS products. Recent research [54] has also indicated that fractionated spacecraft tend to offer additional survivability compared with monolithic ones. Since the spatial distribution architecture spreads the potential targets, the probability of collision can be well controlled.

Such a configuration also combines the promising benefits of flexibility, robustness, responsiveness, and cost-effectiveness [55, 56] of small platforms with the critical capabilities [57] of monolithic ones. The fractionation of system functionalities enables the design requirements of different subsystems to be decoupled, and therefore the corresponding design process can be independent. This maintains the variability and diversity of system designs, allows the design of some elements to be postponed to later lifecycle phases, and provides fractionated spacecraft more degrees of freedom to unpredictable changes or risks [58].

Throughout the lifetime, it is easier for fractionated spacecraft to scale up or down capabilities by adding or removing individual payloads or resource modules according to changing demands. When any technology becomes obsolete or a certain module fails, only the relative module of fractionated spacecraft needs to be replaced, instead of the complete redesign or reconstruction of monolithic spacecraft. Thus, a single module failure will not likely destroy the entire service, and only degrade system capability to some extent. This shows the excellent fault tolerance capability of fractionated spacecraft. For example, as noted by Brown [42, 53], if one module loses its own processing capability, it can utilize the remaining computation nodes across the entire fractionated network, without the need for any new module.

Meanwhile, the independent design processes of different subsystems shorten the research and development cycle of each subsystem, promote the use of latest technologies, and reduce the time and cost of integrating and testing different subsystems in monolithic systems. For instance, as one of the four key technologies for fractionation of space systems, cluster flying of fractionated spacecraft only requires appropriate relative distance and orientation to ensure inter-satellite communication and collision avoidance [59–61], while the accurate and costly orbit maintenance is necessary in traditional formation flying. This can largely decouple the design requirements between satellites, saving considerable cost and time and introducing greater flexibility into space system designs.

Therefore, fractionated spacecraft offer an increased value during the entire lifecycle. This may stimulate the potential applications of fractionated architectures in the future, leading satellites into the era of "smaller, faster, and cheaper New Space". However, there are many problems inherent in traditional design methodologies that may not fit with the design of these architectures.

2.2 Traditional Design Problems

The rapid development of space system technologies cannot be separated from the motivation of spacecraft design methodologies to reduce lifecycle costs and lead time for a given performance [62]. However, the traditional design methodologies generally focus on the design of monolithic spacecraft with complicated structures and high reliability. This does not naturally fit with the above mentioned innovative concepts and technologies. Moreover, the corresponding design and development process currently remains mainly bespoke and labour-intensive with programmes suffering from cost growth and schedule slippage [63].

Traditional requirement-centric design methodologies generally drive a design process by allocating requirements from overall system level to detailed subsystem or component level. The empirical implementation of the breakdown of design requirements and the trade-offs between subsystems or components increases the complexity and the uncertainty of a system.

The complexity and uncertainty decrease the fault tolerance of traditional designs. For example, the Hubble Space Telescope (HST), which is the largest and most complex on-orbit telescope to date, initially suffered severe performance anomalies in blurred images due to an optical flaw in its primary mirror [64]. This problem was finally fixed at an extra cost of approximately US$ 1.1billion [65].

To mitigate the uncertainty and ensure the reliability, redundancy is incorporated into design in case of failures of any subsystem or component. This in turn increases the costs and complexity of the entire system. Meanwhile, long and customised development, manufacturing, integration and testing cycles are required, due to the system complexity. This makes the potential obsolescence of applied technologies more likely, shortens the effective mission lifetime, and presents challenges for mission scheduling.

As the US Government Accountability Office (GAO) pointed out in 2015 [66], the cost of current defence programs has increased nearly 47% from initial estimates. Considering space programmes alone, the US Congressional Budget Office (CBO) revealed that the average cost growth of the past programmes of the National Aeronautics and Space Administration (NASA) was approximately 50%, wherein 20% of them experienced cost growth above 90% [67].

One of the most convincing cases is the Space Shuttle Program [68], or the Space Transportation System (STS), which was administered by NASA from 1981 to 2011. The total cost was estimated at US$ 43 billion [69, 70], when the demonstration and estimate work started in 1972, however the final programme cost was approximately US$ 196 billion [71] during the 30-year service life. The cost per flight also increased dramatically from US$ 0.4 [72] to US$ 1.5 billion [71] over the life of this programme.

Another representative example is the European Global Navigation Satellite System (GNSS), known as Galileo Programme, which was initially approved at €3.4 billion by European Parliament (EP) in 2007 [73]. Unfortunately, it was not so well planned as initially thought. Thus, another €1.9 billion [74], which was more than 50% of the original budget, had to be invested in 2011, in order to complete the constellation of 30 satellites to offer the full service. Furthermore, since then, a total of €7 billion was proposed in the Commission budget proposal [75].

As for schedule slippage, the average delay of current defence programmes is over 29 months, or more than 36%, also declared by the US GAO in 2015 [66]. In terms of NASA's major space projects, the schedule was delayed 11 months on average [76], compared to schedules which were estimated two or three years ago.

Schedule delays can occur for many reasons, and they are usually intertwined with cost overruns. In the aviation industry, the Boeing 787 Dreamliner suffered over US$ 1.6 billion loss, due to its two-and-a-half-year delay [77]. In the space industry, despite being a medium mission with an initial cost of US$ 650 million, updated costs of Mars Science Laboratory (MSL) were estimated at US$ 1.63 billion [67] in 2006.

Subsequently, its cost soon increased to upwards of US$ 2.5 billion, and its launch was delayed from 2009 to late 2011 [78].

Also taking the Galileo Programme as an example, the deployment phase was initially estimated to be completed by 2013 [79], when the comprehensive navigation service was expected to be fully active [80]. However, it was soon delayed with the deployment and service of the first satellites in 2014 [81]. The programme experienced continuous schedule slippage from the outset, and the full system capability has at the time of writing not yet been achieved [82].

Non-technical factors can also contribute to programme delays. For the Galileo Programme, political pressure was one of the major obstacles at the start-up stage, since the US government was concerned that Galileo would be abused by its enemies [83]. In spite of reaching a compromise, the programme was still delayed due to the wide disagreements within the European Union (EU) [84]. After the suspension, the programme had already exceeded the original budget and had to seek additional funding.

In summary, traditional design methodologies appear to have the possibility to incur the problems such as capability uncertainty, cost growth, and schedule slippage. Capability uncertainty misleads the decision making process and increases the risks for the entire project. Cost growth raises the difficulty in system implementation and decreases the return of investment. Schedule slippage extends the design cycle and prevents the use of the latest technologies. All these problems seem to be in the opposition of the flexibility and responsiveness offered by small platforms and innovative concepts and technologies.

Therefore, in order to meet the needs of the rapid development of small satellite platforms, advances should be made, not only in satellite technologies, but also in space design and development processes. The current spacecraft design is considered as a labour-intensive and customised development and production process [62], with few similar units having ever been produced. Such a tedious design methodology does not naturally fit with the growth of the space market. Therefore, new design methodologies, enabling the use of both traditional and innovative concepts and technologies, e.g., modular, standardised, and distributed design, and the incorporation of COTS products, is required to address this stagnancy.

2.3 Lifecycle Design and Analysis

Throughout a space mission, the process of the entire lifecycle of the system can be decomposed and organised through more manageable pieces called lifecycle phases [85]. The typical breakdowns of the lifecycle of a space system have been summarised in Appendix A, wherein the need, concept, development, launch, operation, and retirement phases are the basic elements, seen in Fig. 2.2.

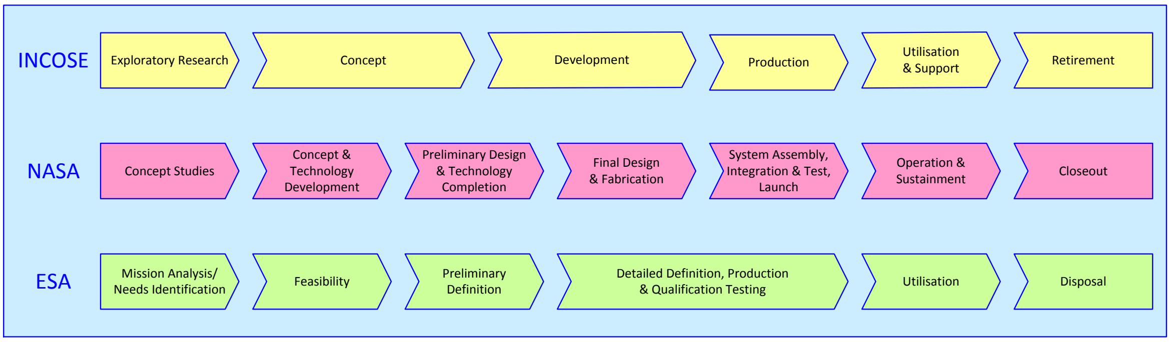


Fig. 2.2: Different system life cycles

Through identification of the needs of users, the concept phase performs the formulation of a system, while the subsequent development, launch, operation, and retirement phases complete the implementation based on the formulation. On the one hand, the conceptual designs have different and long-term effects on the subsequent lifecycle phases, particularly in costs and schedules. On the other hand, the activities in different lifecycle phases introduce different requirements to system designs.

The problems of the traditional design methodologies such as performance uncertainty, cost growth, and schedule slippage, come into being in the process of turning formulation into implementation. Thus, the consideration of the major activities in the implementation lifecycle phases is required in the design process, due to the particular constraints of system designs and the significant effects on capability, cost and schedule.

(1) Development Phase

The traditional spacecraft design generally enlarges the system scale to maximize the value/cost ratio and expands the system lifetime to minimize the annual running cost. This design philosophy makes monolithic spacecraft larger, more complex and more expensive. The cost penalty can be compensated by the enhanced value, yet redundancy and prevention, which come into being from a chain reaction of cost efficiency, always contribute to a rather complicated system. Namely, the cost efficiency of a current paradigm is compensated at the expense of complexity.

The increased complexity makes monolith systems fragile, since any small error can result in entire system failure. Thus, more effort is devoted to the integration, assembly, and testing of such a complex system to ensure the system reliability. This in return increases the cost and time of the entire mission.

Such design philosophy increases the difficulty in establishing a cost model for traditional satellites. Contrary to the growth of space market, there has been little discussion about the cost of developing and producing satellites.

The Unmanned Spacecraft Cost Model (USCM) [86, 87] was established by the US Air Force, Space, and Missile Systems Centre for estimating the development costs of different subsystems of traditional satellites. By analysing the costs of a wide range of space missions, the RAND Corporation developed a spacecraft cost fraction model for different mission types [88]. In addition, in response to the rapid development of small satellites, the Aerospace Corporation developed a spacecraft cost model particularly for small satellites, called the Small Satellite Cost Model (SSCM) [89].

However, what is not yet clear is the impact of system designs on the overall system costs. Similar problems exist in innovative design concepts such as fractionation, despite the continuing concerns surrounding these concepts in the literature.

(2) Launch Phase

Since emerging design concepts and technologies have lowered the threshold of spacecraft design and development, satellite manufacturers are realising significant opportunities for cost reduction. However, the availability, affordability and reliability of access to space seems to be one of the key challenges [90, 91]. Consequently, seeking an appropriate launch opportunity and reducing launch cost may lead to further growth of the space market. This may be more urgent for small satellites, since most of the existing launch vehicles are designed for large satellites.

The current launch opportunities are divided into three categories: dedicated, rideshare, and piggyback launches [92]. Other novel launch concepts such as NanoRacks CubeSat Deployer [93, 94] and Cyclops Deployer System [95], which are devices to deploy small satellites into orbit from the International Space Station (ISS) [96] have recently attracted more and more attention.

a) Dedicated launches

Dedicated launches refer to launching a satellite into orbit by dedicated launch vehicles. The major benefits of using dedicated launches are the accurate orbit insertion and the customised launch schedule. However, the costs of dedicated launches can be high, especially for small satellites. Since most of the existing launch vehicles are initially designed for large satellites, the payload capability of these launch vehicles cannot be fully utilised for most small satellite missions. More specifically, the launch costs are generally associated with payload capacity, total fuel consumption to orbits, installation, maintenance, and operations.

For example, the HST was launched by the Space Shuttle Discovery at a cost of more than US$ 400M [87]. Furthermore, launching a small satellite into Low Earth Orbit (LEO) by the European dedicated launch vehicle Vega costs around US$ 15K per kg [97], while the overall cost of a small satellite can be as low as US$ 10K per kg [98].

b) Rideshare Launches

Rideshare launches refer to the method of launching multiple similar sized payloads into orbit by sharing a single vehicle [99]. One of the advantages of these missions is the decreased launch cost for each individual payload, since the payload capability of the launch vehicle can be fully utilised [100]. On the other hand, rideshare launches may suffer schedule delays and non-optimal orbits, due to the multiple manifestations of the payloads.

The Dnepr is one of the widely-used vehicles for rideshare launches, costing approximately US$ 10 K per kg [101]. In 2013, it successfully delivered 32 small satellites into the orbit [102]. In 2015, the Long March 6 made its first flight to send 20 small satellites into space [103]. In 2017, the PSLV-XL launched a record 104 satellites in a single launch attempt [104], consisting of 88 CubeSats of the Flock 3p constellation [105].

c) Piggyback Launches

Through using the excess mass and volume of commissioned launch vehicles, satellite launches can share a small portion of the cost or receive a complimentary opportunity as piggyback or secondary payloads. This can lead to the significant reduction in launch costs. However, since the launch mission aims to deliver primary payloads into the specific orbits, the accuracy of the orbit, mission time and even the space allocation of the piggyback or secondary payloads are unlike to be guaranteed. Furthermore, there may be significant wait times for such opportunities.

The current potential piggyback launch providers suggested by Heidt [106], are Dnepr (twice a year), Minotaur (every 18 months) and the second stage of Delta.

Despite the above multiple launch options, difficulties still exist in reaching a suitable compromise among user operational intent, industry technical options, and project financial budget. To address this need, the costs and of different launch vehicles were quantitatively assessed by Weigel [107], while the Bayesian method was applied by Guikema [108] to analyse the success rates of different launch vehicles. In addition, an aggregated preference value method was presented by Zhang [109, 110] for users to evaluate different launch opportunities.

However, it is still unclear how the comprehensive effects of all these factors including launch cost, reliability, and preference influence the selection process of the available and applicable launch vehicles for a space launch mission.

(3) Operation Phase

In the operation phase, maintenance and support are implemented to keep the system offering continuous and nominal service in ever-changing conditions. Other activities may be carried out in order to support specific operations, cope with emergencies, reduce running cost, or extend mission lifetime.

As the system reliability decreases over time, system maintenance may be required to maintain the service. However, on-orbit maintenance has traditionally been ignored in the space industry, since it is generally complex and extremely expensive. Both spacecraft and well-trained astronaut crews or robotic systems are a necessity, and often performing on-orbit maintenance can be even more costly than launching a replacement. This is the reason that satellites are generally designed to be highly reliable. So far, the HST is the most successful example of on-orbit servicing. Over its operational life, the HST has undergone repeated on-orbit maintenance and updates including the Servicing Mission 1, which corrected its mirror problem, at a cost of approximately US$ 1.1 billion [65].

In recent years, there has been an increasing interest in fractionated spacecraft

[53, 111], which provides a new insight in realising space system maintenance. Through the use of fractionated configurations, the maintenance of a space system can be fulfilled by the replenishment of the failed or degraded modules, instead of sending a well-trained astronaut or a robotic system to undertake complex repair operations. In terms of Earth imaging missions, O'Neill [112] showed that the replenishment costs of fractionated spacecraft were much lower than monolithic spacecraft.

(4) Retirement Phase

Since the dawn of the space age in 1957, the number of satellites launched has been steadily increasing, so too the quantity of space debris. It is estimated by the European Space Agency (ESA) [113] that more than 170 million space debris objects are currently in Earth orbit. More specifically, a total of 18835 artificial objects in orbit have been tracked by NASA [114], of which only approximately 9% are operating satellites [32].

The two main space debris fields are the ring of objects in Geostationary Earth Orbit (GEO) [115] and the cloud of objects in LEO [116]. These orbit types have greater utility for Earth observation, communication, and navigation missions, and therefore have a higher premium. As revealed by the UCS satellite database, 31% of the 1738 active satellites missions are in GEO, while 62% are in LEO [32]. These two orbit types account for 93% of all the active space missions. However, few satellites are equipped with End of Life (EOL) de-orbit or orbital lifetime reduction devices. This has the potential to increase the risk of collisions between objects and occupy the limited space in particular orbits.

Therefore, the consideration of system disposal should be included into the design of current and future space systems. It is recommended by the Inter-Agency Space Debris Coordination Committee (IADC) Space Debris Mitigation Guidelines [117] for the satellites in the LEO cloud to re-enter Earth's atmosphere within 25 years of mission completion or near the GEO ring to re-orbit to a designed graveyard orbit.

For space systems without EOL de-orbit or orbital lifetime reduction capability, Active Debris Removal (ADR) missions might be the solution. The existing ADR capturing and removal approaches were reviewed and compared by Shan [118]. Despite the conceptual advancements, no single space debris has been removed yet, due to the major concerns about the difficulties in techniques and the high costs [119]. Theoretically, Bonnal [120] analysed the detailed cost trade-offs of ADR between chaser function, including rendezvous, robotic arms, etc., and launch. However, there has been little quantitative analysis of the costs of ADR missions, except the scheme proposed by Yamamoto [121] for the quantitative trade-offs on ADR costs.

Furthermore, far too little attention has been paid to the design of ADR missions to minimise the overall costs, especially for multiple object missions. On the other hand, significant uncertainty still exists about the relation between system designs and ADR mission designs. This may lead to further cost reductions for ADR missions through preliminary conceptual designs of a space system.

2.4 Research Questions and Hypotheses

Based on the review of emerging space design concepts and technologies and the analysis of the problems inherent in traditional design methodologies, the overall aim of this research is decomposed into a series of concrete research questions. The solutions to these research questions form the key steps to develop the proposed design architecture.

(1) Research Questions

To achieve the overall research aim, it is useful to break it down into a set of research questions. Each research question addresses a specific research area of interest that has been identified in the literature review. Furthermore, the solutions to these research questions form the key procedures of the development of the proposed design architecture.

Research Question I: How can a formulation of system design configurations be developed for the overall design process of space systems?

As discussed in the literature review, the traditional requirement-centric design is generally a manual and customised process of selecting an appropriate design to meet requirements. This does not involve the adequate quantitative analysis of system design configurations. One of the keys of this problem is the lack of an appropriate mathematical formulation of system design configurations, namely, a methodology of mathematically defining and expressing the design of space system configurations.

This research question addresses the need for the development of the mathematical formulation of space system design configurations. On the one hand, this mathematical formulation methodology is required to effectively define and express different designs of system configurations. However, on the other hand, it is required to be integrated into the overall design process of space systems, serving for the ongoing design process.

Research Question II: How can intrinsic system properties or value be analysed through system design configurations?

In order to analyse intrinsic system properties or value based on the design configurations, it is necessary to establish a connection between system design configurations and intrinsic system properties or value. The intrinsic system properties or value generally represent the overall characteristics of a system configuration design, and conversely can be used as the requirements or objectives to drive the design process of system configurations. The establishment of the connection can enable the integration of the mathematical formulation methodology of system design configurations into the overall design process.

The primary requirement of this mathematical connection is to analyse the effects of the design of system configurations on different system properties or value. Furthermore, this effects analysis can increase the knowledge and understanding of the trade-offs between different design variables and therefore help the design team improve the design of system configurations with better system properties or higher system value.

Research Question III: How can the analysis of intrinsic system properties or value be used to generate the improved system design configurations?

In order to identify better design configurations, a method to enable the exploration of the design space of system configurations is required. During this exploration, intrinsic system properties or value can be used to influence the preference of the design of system configurations. This process can increase the knowledge and understanding of the design team and provide the supporting information for the decision making process.

In the proposed design architecture, the improvements of system configurations are driven by the selected objectives, instead of the traditional manual modifications. The selection of the optimal design solution is fulfilled through intelligent searching or optimization algorithms, rather than empiricism. Moreover, in terms of multi-objective optimization problems of system configuration design, the proposed architecture also requires the capability to integrate the multiple objectives into a comprehensive metric for the design selection process, in addition to the enablement of the multi-objective optimization process.

(2) Research Hypotheses

The following hypotheses are presented in response to the three research questions, which forms the basis of the proposed design architecture for the analysis of system configurations.

Research Hypothesis I: The formulation of system design configurations can be developed as a set of design variables or a design space for the overall design process of space systems.

In order to overturn the labour-intensive and customised processes of traditional design methodologies, the introduction of methods to enable the quantitative analysis of system design configurations is required. More specifically, the formulation of system design configurations is developed to mathematically define and express the design of system configurations. Such a formulation methodology can enable system configurations to be designed through mathematical operations, instead of traditional labours.

However, to enable the selection of improved system configurations, the mathematical formulation of system design configurations needs to be integrated into the overall design process. To address this need, the mathematical formulation can be used as the design variables throughout the overall design process, forming a design space for system configuration design.

Therefore, it is necessary to implement the other aspects of system design process, e.g., system requirements and objectives, in this design space. In other words, the connections between these aspects and the mathematical formulation should be established, through which the mathematical formulation can form a design space for the overall design process. This benefits the different levels of design parameters and objectives to be traded off in the overall system level, and builds a bridge for them to be designed at a global level.

Based on the analysis of the characteristics of the different space system configurations, this research hypothesis will be accomplished in Chapter 3 and 4, where Chapter 3 provides the definition and description, and Chapter 4 formulates it as a design space.

Research Hypothesis II: Intrinsic system properties or value can be analysed through the establishment of a mapping relationship between system design configurations and system properties or value.

As the system characteristics and design parameters, intrinsic system properties or value play a significant role in the overall design process. Having identified the formulation of system design configurations as the design space, it is useful to establish the mapping relationship between system design configurations and intrinsic system properties or value.

This enables the design of system configurations to be integrated into the overall design process. The functions of this integration are dual: on the one hand, different designs of system configurations correspond to different intrinsic properties or value. Thus, the effects of design configurations on intrinsic properties or value can be analysed, deepening the knowledge and understanding of system configuration designs. On the other hand, the selected intrinsic properties or value can be used as the feedback to the design of system configurations. In other words, design a system configuration with respect to the selected intrinsic properties or value as the objectives.

The establishment of the mapping relationship can be performed through the integration of the intrinsic properties or value from the subsystem or more detailed level into the overall system level. The integration operations can be implemented through the mathematical operations in the design space of system configurations. As the input parameters, the modelling of the intrinsic properties or value at the subsystem or more detailed level is the enabling factor in this analysis process.

Applying the proposed formulation of system design configurations and appropriate value models, this research hypothesis will be accomplished in Chapter 5, establishing an integration process of system intrinsic properties or value.

Research Hypothesis III: The design architecture with the analysis of intrinsic system properties or value can be applied through numerical optimization algorithms to explore the design space of system configurations.

To enable the improvement process of system design configurations, the exploration of the design space with respect to the selected design objectives of system properties or value is implemented. This process can increase the knowledge and understanding of the design team and provide the supporting information for the decision making process.

Throughout the exploration, intrinsic system properties or value can be adopted as the system requirements to identify the feasible domain of the design variables or the design objectives. The domain or design objectives can be used to influence the preference of the design of system configurations. This exploration process meets the conditions of generating the feasible domain and evaluating each design, which are required by a perfect optimization process [122]. Consequently, the optimization process of choosing the design with the highest value can be effectively implemented.

Through applying appropriate numerical optimization algorithms, an improved design configuration can be achieved with the selected property or value as the objective function. In terms of multi-objective optimization problems, multi-objective optimization algorithms can be applied, while the proposed design architecture can also enable the integration of the multiple objectives into a comprehensive metric for the ongoing design selection process. Furthermore, the monetary dimension is suggested in this research as one of the applicable dimensions to fulfil this multi-objective integration process.

Based upon the integration process of system intrinsic properties or value established in Chapter 5, this research hypothesis will be accomplished in Chapter 6 to explore improved design configurations.

Chapter 3

System Configuration Characteristics

Based upon the categorisation of different spacecraft configurations, the system characteristic space consisting of degree of duplication, fractionation, and derivation is proposed [20], each representing a characteristic of system design configurations. Conversely, these three dimensions together construct a design space to explore potential design configurations. In this chapter, their definitions and the characteristics are described to support the qualitative analysis of mission characteristics. Consequently, the connection between system designs and configuration characteristics is established, enabling a comprehensive exploration of space mission concepts.

3.1 Categorisation of Space Systems

It is worth clarifying the terms satellite, spacecraft, and space system, since they are used somewhat ambiguously and can have different meanings depending on the context. For the purpose of this research, these terms are prescribed and described in details in this section.

A **Satellite** is an artificial body revolving in an equilibrium orbit around a celestial body [123], e.g., the Earth, the Sun or any other of its planets and moons. A satellite can be used for a great variety of scientific and technological purposes, to support communication, navigation, observation, etc. A satellite in space is similar to an atom in chemistry, which is the smallest constituent unit that maintains the properties of chemical elements [124]. It is designed to perform at least one function, or the entire mission, depending on the configuration of the space system.

A **Spacecraft** is a vehicle designed for travelling through space [48], which typically has, as a minimum, a propulsion subsystem, a power supply and a payload [125]. According to its definition, a spacecraft might comprise of one or more satellites, if some of the above subsystems are spatially distributed. If satellites are imagined as atoms, spacecraft can be naturally analogised as molecules. A molecule may be homonuclear, which consists of atoms of a single chemical element, such as oxygen (O2); or it may be heteronulear, a chemical compound composed of more than one element, such as water (H2O). Similarly, spacecraft may also be homogeneous, that is, all the satellites are nearly identical; or it may be heterogeneous, also known as fractionated spacecraft.

A **Space system** is the part of a satellite-based system that resides in space [125], which may comprise one or more spacecraft to implement different missions. A space system may be as large as a cluster of hundreds or thousands of satellites, or as small as only one satellite, depending upon specific mission requirements. Using the previous analogies, space systems can be considered as mixtures in chemistry, which are material systems made up of two or more different substances or molecules [124]. In the same way, different molecules are mixed together for different products, different spacecraft are designed in a space system to realise different mission requirements.

According to different configurations, space systems are classified into four categories [20]: monolithic spacecraft, constellations of identical spacecraft, fractionated spacecraft, and hybrid spacecraft, with the definition and characteristics of each category clarified as follows.

Monolithic spacecraft, also known as traditional spacecraft or singular spacecraft, are generally space systems comprising of only one satellite that has a monolithic configuration, where functionally independent subsystems, such as data handling, communication, and power, are not architecturally separated but integrated on a single physical platform [20]. In essence, a monolithic system refers to a massive, unchangeable structure without any individual variation, so that it may be powerful but slow to change [48].

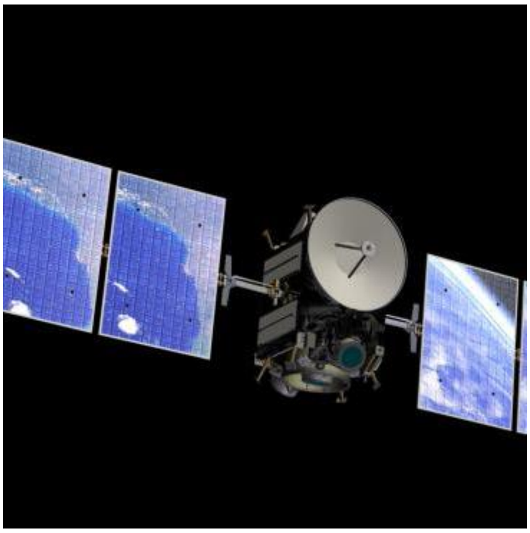


Fig. 3.1: Dawn [129]

Fig. 3.2: HST [131]

For monolithic spacecraft, all the mission objectives are realised by a single spacecraft. Any component failure on board may weaken the integrity of the entire system, and endangers overall system capability and lifecycle value. Despite cautious selection of components, redundancy design and extensive ground testing, fragility remains in such a single and tightly coupled system [126], which is an intrinsic attribute of traditional design.

Monolithic spacecraft can accomplish a variety of space missions, ranging from earth observation to interplanetary exploration, almost all the known missions. Some notable examples are the first artificial satellite Sputnik-1 [127] , the first interstellar spacecraft Voyager-1 [128], NASA's asteroid probe Dawn [129] seen in Fig. 3.1, and the current largest and most complex on-orbit telescope HST [130] shown in Fig. 3.2.

Constellations of identical spacecraft transform monolithic spacecraft into homogeneous clusters composed of a group of identical or near identical spacecraft flying in a certain formation or constellation, which can function independently from each other [59]. They are commonly-used in near-Earth space, for missions that require a continuous and global service. Additionally, a series of identical spacecraft are likely to be manufactured in production lines [132] and can therefore allow economics of scale to give cost reductions.

Unlike monolithic spacecraft, which operate alone throughout the entire mission lifetime, constellations of identical spacecraft tend to work in-group with necessary maintenance or upgradation. Apart from the initial launched space system, a set of

Fig. 3.3: GPS Block IIF [135]

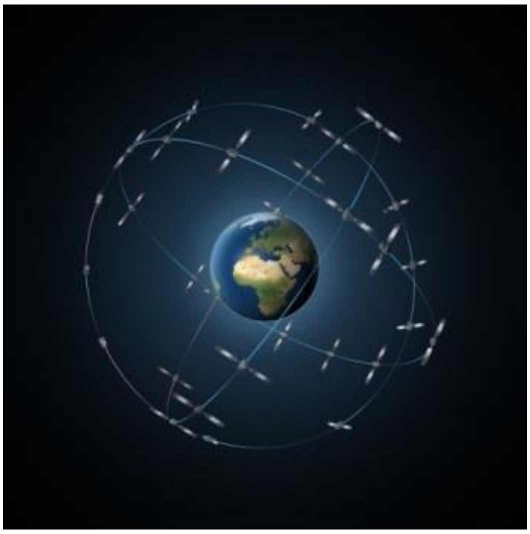


Fig. 3.4: Galileo [136]

replenishment spacecraft are often developed in case of any orbit failure or system upgrade. As the project progresses, a series may be replaced by a more competent or economical series, for the purpose of improving system capabilities or extending mission operations. Moreover, losing any satellite should not destroy the entire service for well-designed constellations, at most the mission should suffer some functionality degradation.

Examples here consist of the famous navigation constellation the Global Positioning System (GPS) shown in Fig. 3.3, which achieved its Full Operational Capability (FOC) in 1995 and has been undertaking replenishment and modernized processes since 2005 [133], the European GNSS Galileo seen in Fig. 3.4, Iridium the unsuccessful commercial communication constellations in LEO, and the Tacking and Data Relay Satellites (TDRS) [134] in GEO.

Fractionated spacecraft decompose monolithic spacecraft into heterogeneous clusters of wirelessly-interconnected modules, each capable of sharing and utilising the resources throughout the entire network [16, 137, 138]. Unlike constellations of identical spacecraft, where each single spacecraft in a cluster is identical or near identical, almost every module of fractionated spacecraft has at least one distinctive functionality [139], corresponding to various subsystems of a monolithic system.

As Defense Advanced Research Projects Agency (DARPA) describes [140], such architecture enhances the adaptability and survivability of space systems, while shortening development timelines and reducing the barrier-to-entry for participation in the

Fig. 3.5: System F6[140]

security space industry. Flexibility comes from the fractionation of functionalities. This makes it possible for the entire system to survive from some unforeseen damage or failures. Meanwhile, modularity and COTS products can cut down the cost and development cycle. Thus, fractionated spacecraft are considered as an emerging concept with promising applications.

To date there exist no true operational fractionated spacecraft. The systems currently only exist as conceptual and technological examples such as Future Fast, Flexible, Fractionated, Free-Flying Spacecraft (System F6) proposed by DARPA [111] seen in Fig. 3.5.

The remainder of space systems are considered as **hybrid spacecraft**, namely combinations of constellations of identical spacecraft and fractionated spacecraft. Strict to the definition, each member of a fractionated spacecraft is different, while backups are quite common in spacecraft design. Therefore, pure fractionated spacecraft only exist theoretically, and for practical purposes hybrid spacecraft can be merged with fractionated spacecraft.

3.2 System Characteristic Space

The system characteristic space is constructed by three different dimensions, which are the degree of duplication, fractionation, and derivation [20], shown in Fig. 3.6.

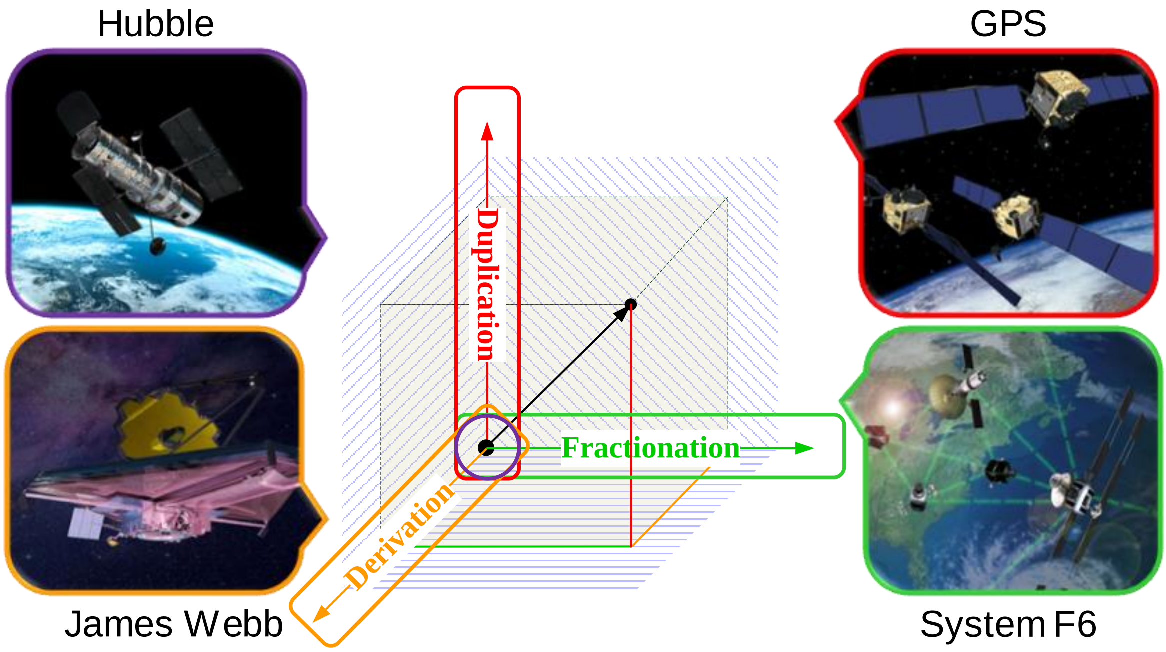


Fig. 3.6: System characteristic space (The colour purple, red, green, and yellow match the monolith, duplication, fractionation, and derivation respectively.) [20]

Duplication refers to components, subsystems, or satellites in a space system, technically duplicated to realise higher performance, more powerful functions, or in case of on-orbit failures [20]. Within the domain of duplication, units performing the same function can be exchanged by one another. Duplication, also known as redundancy, is widely-used in system engineering to increase system reliability. This duplication is generally integrated in a monolithic system, while the proposed duplication is enabled to be spatially distributed in different units of a space system, instead of being limited to monolithic configurations.

For space missions such as GPS and TDRS, duplication is necessary since individual satellites cannot implement the full capability. Currently, 31 of the 32 satellites in the GPS constellation are active, with on average 9 satellites visible from any point on the ground at one time [141], ensuring the minimum 4 satellites for any position calculation [142]. Of the 9 satellites, 4 are duplicated to meet the FOC, while the other 5 are used for improving the calculation accuracy or for redundancy.

Fractionation refers to the spatial distribution of the essential functionalities in a space system [20]. The primary difference between fractionation and duplication is that fractionation increases the heterogeneous degree of a space system, while duplication increases its homogeneous degree. Unlike duplication, each member in a fractionated system performs a different function to fulfil the collaborative mission.

The fractionation of a spacecraft decouples the design requirements of different subsystems, and enables the entire system to be developed in parallel. Meanwhile, the spatial separation of different subsystems offers more options for launching these modules, namely, by one or several launches. This characteristic also increases the flexibility for system lifecycle maintenance.

Previous studies [59–61] have identified that networking, wireless communication, cluster flight, and distributed computing are the four critical technologies to realise the fractionation of a space system. As one of the biggest characteristics of current space technologies, the incremental use of on-board processing [86] offers a great opportunity for the application of fractionation. The fractionation of subsystems such as Attitude and Orbit Control Subsystem (AOCS) may contribute to the decrease of its own scale, since only certain subsystems need to be controlled instead of the entire system. The cluster flight of fractionated spacecraft only requires appropriate relative distances and orientations to ensure communication links and collision avoidance [42], which differs significantly from the accurate orbit maintenance required for traditional constellations.

Derivation refers to components, subsystems, or satellites in a space system, whose technologies and capabilities are derivative from previous missions [20]. On the one hand one could say that, the more derivative elements in a system, the more mature a system is, which may imply higher reliability and cost-effectiveness of the entire system. However on the other hand, the innovative or less derivative products designed by advanced concepts and technologies may improve system performance or extend system capability.

In summary, the three primary characteristics represent different system properties and capabilities. Duplication is a common approach to directly increase the reliability of a system, which in turn effectively manages mission uncertainties and risks. Fractionation is an emerging design concept proposed to shorten the design and development cycle of coupled subsystems, lessen individual launch mass and capacity requirements, and reduce lifecycle maintenance and upgrade costs. Derivation draws on the experience of previous design paradigms to cut down the time and risk of the design and development process of a system, enabling a responsive and inexpensive design.

3.3 Mission Characteristic Analysis

The proposed system characteristic space can be used as a valuable tool for both qualitative and quantitative analysis of space system design. In this section, the qualitative analysis of space system characteristics is implemented, while the remainder of this research will focus on the quantitative analysis of space system conceptual design.

Complying with the definitions of the system characteristic space identified earlier, the system designs of different space missions can be evaluated and compared. Adopting a series of classical space missions as examples for different design configurations, e.g., Mariner and Voyager (M&V), Hubble Space Telescope (HST), Global Positioning System (GPS), International Space Station (ISS), and RapidEye (RE), an exemplary characteristic assessment of these configurations was performed and illustrated in Fig. 3.7, where the three dimensions denote the degree of duplication/ fractionation/ derivation. The levels from the lowest to the highest are achieved respectively from none (level 1), component (level 2), subsystem (level 3), satellite (level 4) to system level (level 5). In Fig. 3.7, it is noteworthy that the bigger the area, the more the space mission encompasses the characteristic space, namely, the greater the mission potential.

Mariner was a space program consisting of a series of interplanetary probes designed for investigating various planets in the solar system, with Voyager program as its extension. The Voyager 1 and 2 spacecraft were nearly identical, and the Mariner series also had an architecture of redundancy, e.g., Mariner 3 and 4 were identical teammates in the Mars flyby mission. Furthermore, during the evolution of system designs, the Mariner series was derivative. As spacecraft design and development technology became more and more advanced, space missions ventured further afield. For instance, Mariner 1 and 2 were just Venus flyby missions, while Mariner 8 and 9 were designed to map the Martian surface.

All the spacecraft in the Mariner mission were monolithic. Currently, the monolithic configuration is believed to be the most reliable and applicable solutions for outer space missions, while duplication and fractionation may incur unpredictable risks [20].

Examples have proved the advantages of monolithic configurations, such as voyager-1, which is the first interstellar spacecraft in the history of space exploration [128]. Despite the additional survivability offered by fractionation, technical difficulties such as interplanetary formation flight and inter-satellite communication have to be resolved beforehand. Moreover, the fractionated modules for interplanetary missions are required to operate synchronously and fly nearby, spontaneously raising the difficulties in the launch and deployment.

HST is a space telescope launched into LEO for deep space observation, which has experienced on-orbit servicing such as repairing, updating and replacement throughout four Space Shuttle missions [20]. It has cost at least US$ 5 billion in Fiscal Year (FY) 2010 over the last 25 years [65]. This indicates that Hubble exhibits some degree of derivation, despite being a monolithic spacecraft. However, such a huge maintenance cost may imply that monolithic configurations are potentially not the most economic solution at least where derivation is concerned.

Assuming HST was designed as a distributed configuration, it is likely that the lifecycle servicing cost would decrease as the degree of fractionation increases. For

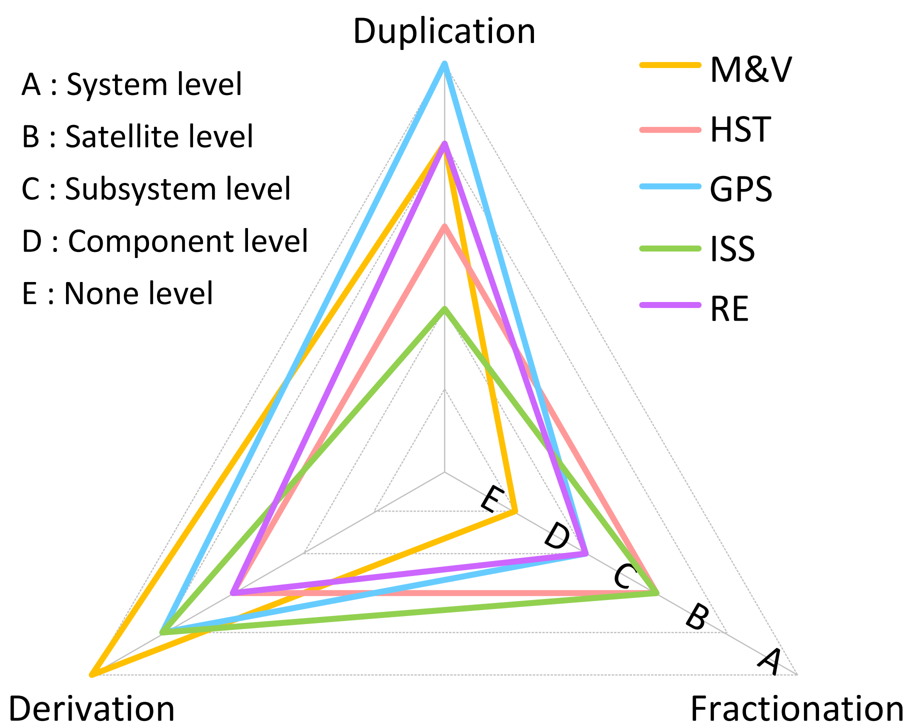


Fig. 3.7: Comparisons of different mission characteristics in the system characteristic space [20]

instance, the servicing mission 3A was to replace six gyroscopes, a fine guidance sensor and the main computer [143, 144]. If subsystems such as command and data handling were spatially separated from the payload, only a few new subsystems would have been required to be launched. Huge maintenance costs could be saved, particularly in developing and testing on-orbit servicing technology. This reveals the outstanding flexibility and robustness of distributed configurations in long lifecycle space missions. In other words, increasing the degree of fractionation and duplication of a space system is beneficial for reducing lifecycle costs, considering on-orbit servicing such as repairing, replacement, and updating.

The well-known GPS constellation is a space-based navigation system providing location and time information in various weather conditions. Overall, it is a good example of constellations of identical spacecraft, since all the satellites performing the same functionality, and can be simply replaced by each other.

Due to the replenishment and modernisation in the evolution of the GPS constellation, satellites in different blocks manifest different behaviours although performing the same function. The difference between new and old blocks can be recognised as the derivation in a satellite constellation, i.e., new blocks retain the same function as earlier ones, but may have some updates and reinforcements. For example, the GPS block IIIA aimed to circumvent the technical issues still threatening GPS block IIF, in addition to achieving more ambitious goals [145, 146]. Last but not the least, the homogeneous configuration indeed makes the endurance of GPS significant in comparison to any monolithic navigation satellite.

The characteristics of the ISS are perhaps more complex. In terms of functionality, it has a modular configuration with system functionalities fractionated to some extent. The basic ISS was like an infrastructure module to provide services such as power and communication, while the payload modules were later launched and docked with the infrastructure one to enable the functionality of space applications. However, all the modules are docked, meaning the entire system is essentially a monolithic spacecraft. Thus, the ISS can be considered as one of the original fractionated spacecraft, albeit a monolithic one.

The monolithic configuration of the ISS has its unique advantages and disadvantages. On the one hand, since all the capsules or modules are docked together, wireless communication devices and other mass penalties caused by fractionation can be naturally avoided. On the other hand, capsules or modules to be launched must possess the ability to rendezvous and dock with the existing system, which inevitably increases the complexity and cost of the development and maintenance of the space station. In addition, both scalability and updatability are largely limited by the initial system design, which restricts the architecture adjustment and technical advancement.

If the ISS was designed in a distributed architecture, this mission could be implemented in a simpler and more cooperative way, since the ISS was built through the cooperation of a number of space agencies, e.g., NASA, ESA, and the Japan Aerospace Exploration Agency (JAXA). Most capsules or modules could be designed separately without docking ports, saving considerable time and cost among different space agencies. Thus, reconfiguration and upgrades could be more easily realised, even the network framework is possible to be changed to some extent.

The final mission considered in Fig. 3.7, RapidEye is a constellation of five identical observation satellites, operated by BlackBridge AG, providing geospatial information for decision making [147]. The configuration of RapidEye is identical, since all the satellites are equally deployed containing the same sensors [148, 149]. RapidEye differs from previously discussed constellations in that it adopts small and inexpensive platforms and COTS components. This implies that bespoke and complicated designs can be replaced by modular and standardised designs, which introduces additional derivation from industries outsides the traditional space sector.

3.4 Summary

Having categorised different spacecraft, three primary configuration characteristics were generalised to construct the system characteristic space. Based on such a design space, a connection was built between system conceptual designs and configuration characteristics, enabling the qualitative exploration of the feasibility and applicability of design configurations for various mission scopes. In other words, mission concepts can be analysed and designed in the system characteristic space, where each dimension reflects different characteristics of design configurations. Additionally, such a system characteristic perspective might provide designers better understanding of the limits and potentials of a space mission, beneficial for further mission modification or extension.

Chapter 4

Value-Centric Design Architecture

Value-centric design has recently received significant attention from the space industry, in order to solve problems such as cost overruns and schedule delays inherent in traditional requirement-centric design methodologies. In accordance with the previously qualitative understanding of the design space of system configurations presented in Chapter 3, the general philosophy of value-centric design will be illustrated in this chapter. Based on this description, a configuration design architecture will be established. Overall, the proposed value-centric design architecture acts as a systematic guideline for the quantitative design and analysis of system configurations, with the specific mathematical approaches and techniques described to formulate each step.

4.1 Value-Centric Design Philosophy

Traditional Requirement-Centric Design (RCD) is generally a process of selecting an appropriate design to meet requirements, as illustrated by Fig. 4.1. The major features of this process are that it is labour-intensive and customised. Thus, this requirement allocation process incurs many problems, most significantly cost overruns and schedule slippage. The US GAO [66] recently found a cost growth of nearly 47% for major defence programmes, while the average delay was over 29 months, or more than 36%. In terms of space programmes, NASA's large programmes have typically experienced cost overruns of 50% [67], and schedule delays of 11 months [76].

Value-Centric Design (VCD) has been widely proposed as an effective solution to the problems arising from traditional design methodologies. Value-centric design, also known as Value-Driven Design (VDD), is a perspective of making full use of engineering optimization methods in the design process of large and complex systems, such as aircrafts and spacecraft. As defined and illustrated by Collopy [150], shown in Fig. 4.2, value-centric design displays three outstanding characteristics:

1) **Enables the use of both traditional and innovative engineering optimization methods**.

Value-centric design searches for the optimal solution based on value models, which estimate the system value and its variance and probabilistic distribution of unanticipated events. The output of value models, namely, system value, offers a comparable metric for enabling system design optimization. Ideally, it is believed to be better than any traditional process that designs a system to meet a set of defined requirements. However, in practical, it is nearly impossible to establish value models to precisely describe space system designs. Therefore, it is the inaccuracy of the value models that distort the truth or the results of value-centric design methodologies.

2) **Prevents trade conflicts in the conceptual design process across multiple subsystems and components**.

Another merit of VCD is that it keeps multiple trade parameters consistent across all the subsystems and components. Value-centric design estimates the system value, apart from parameterizing design variables, elaborating detailed definition, and analysing system properties in the traditional design. The estimate of system value provides better understanding of the designed system than just partitioning customers' requirements into various subsystems and components, which may cause many trade conflicts. Moreover, it is important for certain modules to be designed independent of the rest of the system, due to the schedule and cost constraints.

3) **Involves risk management into the process of system trade design**.

Value-centric design is a new method to deal with risk management in the process of system development, making risk management no longer independent of system design. While traditional design focuses on the pursuit of outcomes to meet mission requirements, value-centric design acknowledges the uncertainties and then seeks for opportunities and avoids undesirable results. This increases the probability of resolving the problems of cost growth and schedule delays in the traditional design process of requirement allocation.

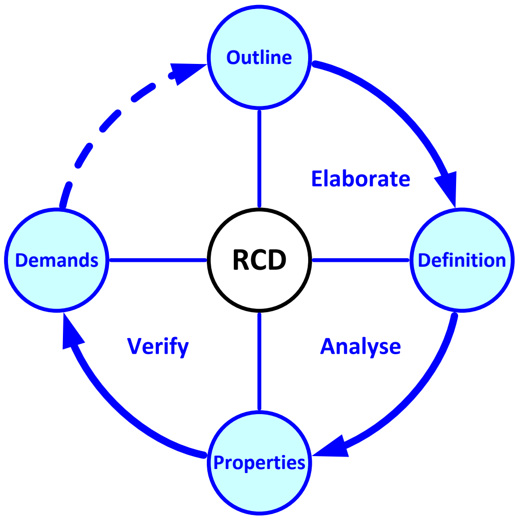


Fig. 4.1: Requirement-centric design

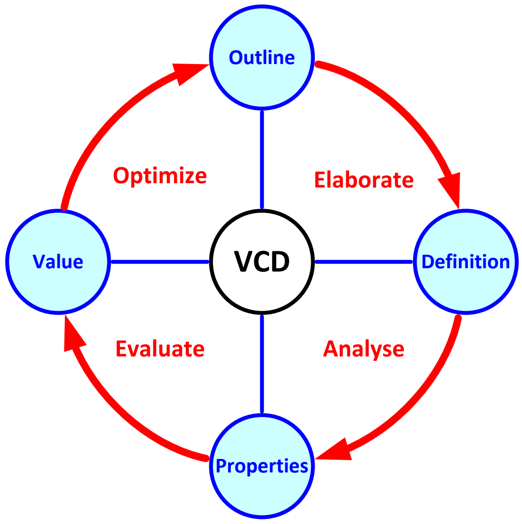


Fig. 4.2: Value-centric design [20]

In conclusion, value-centric design has three outstanding characteristics, compared with traditional requirement-centric design: enabling the application of various engineering optimization methods, preventing trade conflicts in the design of multiple subsystems and components, and realising better management of and even seeking opportunities from risks.

4.2 System Configuration Design Architecture

To enable the use of optimization methods and improve system capabilities and performance with traditional or innovative design concepts and technologies, a new value-centric design architecture is proposed based on the system characteristic space introduced in chapter 3.

The basic value flow process of the architecture is illustrated in Fig. 4.3. Overall, system value is accumulated from the bottom to the top. System value reflects certain design characteristics of space systems such as cost. At the subsystem level (system level 4), the value within each subsystem is generated from subsystem properties, such as mass, size, and reliability. The subsystem value is subsequently integrated at the spacecraft level (system level 3). Additionally, the integrated spacecraft value also influences the value characteristic of the following launch (system level 2) and operation (system level 1) activities. Finally, the value of the four system levels is synthesized into the overall system value.

For instance, assume that the concerned system property is mass and the corresponding value is the expected launch cost. Through the exploration of the feasible domain of design variables, namely, different system designs, the analysis and optimization of the launch cost of a space system can be achieved.

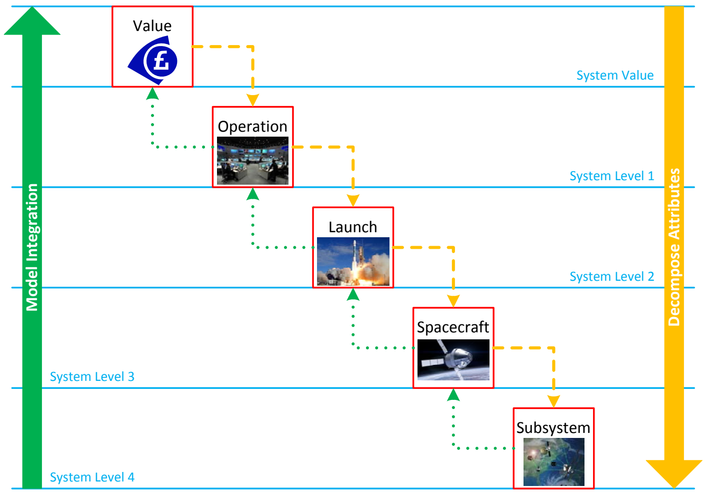


Fig. 4.3: Value flow diagram of the proposed design architecture

Under the philosophy of value-centric design illustrated in Fig. 4.2, the concrete approaches and specific techniques applied in the proposed space system design methodology are described as follows [20].

1) **Elaborate**. Initially, the rough outline of the system to be designed is parameterized by the design variables of system configurations, i.e., the degree of duplication, fractionation, and derivation. In the **elaborate** process, the design variables are transformed into the complete definition of overall system configurations at the subsystem level. Therefore, the key technique of this step is the description or formulation of the proposed system characteristic space and the system configuration definition, and the mutual transformation relations.

2) **Analyse**. In the **analyse** process, the relations between the internal system configurations and the external system properties are established. The integration of subsystem properties into overall properties are divided into two steps: the first is to build the subsystem property models; the second is to discover the integration philosophy by exploring the configuration characteristics. As a generic design architecture, this system configuration definition method should enable the application of monolithic, duplicated, fractionated configurations or any of their feasible combinations.

3) **Evaluate**. In the **evaluate** process, the overall value is quantified by appropriate value models to determine whether the requirements are met, based on different properties. As the major element distinguishing value-centric designs from traditional ones, the value model is the enabling factor of optimizing system designs, while the technique difficulties reside in how to measure different properties in a singular dimension. One of the possible solutions is to convert them into the dimension of cost, however the cost characteristics of some properties are still unclear.

4) **Optimize**. In the **optimize** process, system value is adopted as the objective function or the output of a single design loop, which is returned to design variables as feedback. Such a data flow loop distinguishes the value-centric design from the human-in-the-loop aspect of the requirement-centric design. Under the proposed architecture, the optimization process is realised by optimization algorithms rather than the manual modifications of design parameters. Consequently, the theoretical optimal set of design configurations can be achieved, instead of reaching an empirically feasible solution that simply meets mission requirements.

In conclusion, the proposed system configuration design architecture has been established based upon the value-centric philosophy. The aim of enabling the use of both traditional and innovative design concepts and technologies has been theoretically accomplished, with the concrete approaches and specific techniques described in the following chapters. More specifically, the mathematical definition and formulation of system configuration designs are established in the remainder of this chapter, while the other three steps are exemplified by analysing and optimizing the costs of different lifecycle phases such as development, launch, operation, and retirement phases in the subsequent chapters.

4.3 Formulation of System Configuration Designs

Under the proposed value-centric design architecture based on the system characteristic space, this section will focus on the formulation of the quantitative analysis and design of system configurations, serving as the mathematical basis of the architecture.

Assume m satellites in a space system, and n types of subsystems in each satellite. In this case, the elements of the System Design Matrix D_k, given in Eq. (4.1), are the particular property, k, of every subsystem on every satellite, where each element d_{ij} (i=1,2,$\cdots$,m, j=1,2,$\cdots$,n) denotes the value of that type of subsystem property on that particular satellite. For example, for the property "mass", the value of element $d_{1,2}$ is the total mass of all subsystems of type 2 on satellite 1. There is a separate D matrix for each relevant property, e.g., mass, cost, reliability, etc.

$$D = \begin{bmatrix} d_{11} & d_{12} & \dots & d_{1n} \\ d_{21} & d_{22} & \dots & d_{2n} \\ \vdots & \vdots & \ddots & \vdots \\ d_{m1} & d_{m2} & \dots & d_{mn} \end{bmatrix} \tag{4.1}$$

This system design matrix can be understood in two dimensions: on one hand, each row defines the different designs of these n subsystems in a certain satellite; on the other hand, each column displays the distribution of certain subsystem in the system or across the satellites. For additive system properties such as mass and cost, each element d_{ij} is simply the product of the unit property of that subsystem times the degree of duplication, as given by Eq. (4.2). For non-additive properties such as reliability, a more sophisticated function of degree of duplication is used.

$$d_{ij} = n_{ij} u_{ij} \tag{4.2}$$

Where, n_{ij} and u_{ij} are the degree of duplication and the subsystem unit design property of subsystem j in satellite i respectively. Let the System Configuration Matrix N and the Unit Design Property Matrix U be given by Eq. (4.3) and Eq. (4.4) respectively.

$$N = \begin{bmatrix} n_{11} & n_{12} & \dots & n_{1n} \\ n_{21} & n_{22} & \dots & n_{2n} \\ \vdots & \vdots & \ddots & \vdots \\ n_{m1} & n_{m2} & \dots & n_{mn} \end{bmatrix} \tag{4.3}$$

$$U = \begin{bmatrix} u_{11} & u_{12} & \dots & u_{1n} \\ u_{21} & u_{22} & \dots & u_{2n} \\ \vdots & \vdots & \ddots & \vdots \\ u_{m1} & u_{m2} & \dots & u_{mn} \end{bmatrix} \tag{4.4}$$

The System Configuration Matrix N defines the number of identical or near identical subsystems in each satellite, or the degree of duplication of each subsystem in each satellite. Once the System Configuration Matrix N is determined, the system configuration can thereby be determined. The Unit Design Property Matrix U can be used for system property analysis to represent any kind of property of each subsystem conceptual design. Therefore, the System Design Matrix D in terms of N and U is given by Eq. (4.5).

$$D = N \circ U \tag{4.5}$$

Where, "$\circ$" is Hadamard product, or element-wise multiplication.

4.4 Formulation of System Characteristic Space

This section formulates the system characteristic space, through the three dimensions of duplication, fractionation, and derivation.

4.4.1 Duplication, Fractionation, and Sequence

Mathematically, degree of duplication can be defined as the number of identical or near identical subsystems contained in the overall space system. Thus, the System Duplication Vector P, given by Eq. (4.6), can be calculated from the System Configuration Matrix N.

$$P = \begin{bmatrix} p_1 & p_2 & \dots & p_n \end{bmatrix}^\intercal \tag{4.6}$$

Where, each element p_j is the sum of elements in each column of N.

$$p_j = \sum_{i=1}^{m} n_{ij} \tag{4.7}$$

Similarly, the System Fractionation Vector F, given by Eq. (4.8), can be mathematically determined by the number of satellites with the same subsystem.

$$F = \begin{bmatrix} f_1 & f_2 & \dots & f_n \end{bmatrix}^\intercal \tag{4.8}$$

Where, each element f_j is the number of nonzero elements in each column of N.

The System Duplication Vector P and the System Fractionation Vector F can be derived from the System Configuration Matrix N, which is called forward transformation. However, the System Configuration Matrix N cannot be derived directly from the System Duplication Vector P and the System Fractionation Vector F, hence it is not reversible. To make this transformation reversible, the System Sequence Vector Q, given by Eq. (4.9), is introduced to complement the information loss from the forward transformation.

$$Q = \begin{bmatrix} q_1 & q_2 & \dots & q_n \end{bmatrix}^\intercal \tag{4.9}$$

Where, each element q_j is the sequence number of each subsystem permutation in the overall system, determined by its ranking in the Subsystem Arranging Matrix A_j.

$$A_j = \begin{bmatrix} a_{j,11} & a_{j,12} & \dots & a_{j,1n_j} \\ a_{j,21} & a_{j,22} & \dots & a_{j,2n_j} \\ \vdots & \vdots & \ddots & \vdots \\ a_{j,m1} & a_{j,m2} & \dots & a_{j,mn_j} \end{bmatrix} \tag{4.10}$$

Where, A_j is the Subsystem Arranging Matrix of subsystem j, and n_j is the corresponding total number of possible arrangements. Each column vector $A_j(k)$ of the Subsystem Arranging Matrix A_j presents one possible permutation of subsystem j across m satellites, as shown in Eq. (4.11).

$$A_j(k) = \begin{bmatrix} a_{j,1k} & a_{j,2k} & \dots & a_{j,mk} \end{bmatrix}^\intercal \tag{4.11}$$

Where, each element $a_{j,ik}$ is the element of the k^{th} column of the Subsystem Arranging Matrix A_j. Thus, the sum of each column vector $A_j(k)$ has the same value, which is equal to the total number p_j of subsystem j, as presented by Eq. (4.12).

$$p_j = \sum_{i=1}^{m} a_{j,ik} (k = 1, 2, \cdots, n_j) \tag{4.12}$$

The calculation process of the number of all the possible permutations n_j can be considered as one of the classical problems in combinatorics, i.e., p_j unlabelled balls in f_j labelled buckets.

$$n_j = \binom{f_j}{m} \binom{f_j - 1}{p_j - 1} \tag{4.13}$$

In order to locate each of all the n_j permutations in the matrix A_j, these permutations can be sorted from the first to the last element, in rising or falling value. As a set of permutations, A_j is a matrix consisting of all the possible permutations of subsystem j across m satellites. Thus, different sorting approaches are equivalent. They do not change any of these permutations, but the orders of these column vectors in the matrix A_j. Moreover, all the Subsystem Arranging Matrices A_j together constitute the System Arranging Matrix A.

$$A = \begin{bmatrix} A_1 & A_2 & \dots & A_n \end{bmatrix} \tag{4.14}$$

The System Arranging Matrix A is derived from the System Duplication Vector P and the System Fractionation Vector F. Once given P and F, all the possible arrangements of the configuration design are determined, and the System Arranging Matrix A is the assembly of these permutations. Meanwhile, these arrangements can be sorted in a specific way, so that each one can be tracked effectively and efficiently.

Therefore, the System Duplication Vector P and the System Fractionation Vector F, associated with the System Sequence Vector Q, have established a one-to-one correspondence with the System Configuration Matrix N, and the transformation from P, F, and Q to N is a backward transformation, namely, the reverse transformation of the forward one.

The following provides an example to illustrate this process. If we assume that we have a space system consisting of 6 satellite each with 5 subsystems, i.e., m=6, n=5, and the corresponding System Configuration Matrix N is presented by Eq. (4.15).

$$N = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 \\ 0 & 2 & 0 & 0 & 0 \\ 1 & 2 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 1 & 1 \\ 0 & 0 & 0 & 0 & 2 \end{bmatrix} \tag{4.15}$$

Where, the element n_{ij} presents the degree of duplication of subsystem j in satellite i. For instance, if the satellites are labelled from 1 to 6 and subsystems are labelled from A to E, the element $n_{3,2}$=2 means that satellite 3 has 2 subsystems B.

According to the definitions, the elements of the System Duplication Vector P and the System Fractionation Vector F can be calculated by the matrix N. Take j=2 as an example, namely, subsystem B, we can acquire the results shown in Eq. (4.16) and Eq. (4.17).

$$p_2 = 2 + 2 = 4 \tag{4.16}$$

$$f_2 = 2 \tag{4.17}$$

Through the forward transformation, the complete System Duplication Vector P and the System Fractionation Vector F can be obtained, given by Eq. (4.18) and Eq. (4.19) respectively.

$$P = \begin{bmatrix} 2 & 4 & 2 & 1 & 3 \end{bmatrix}^{\intercal} \tag{4.18}$$

$$F = \begin{bmatrix} 2 & 2 & 2 & 1 & 2 \end{bmatrix}^{\intercal} \tag{4.19}$$

Meanwhile, the sequence vector can also be acquired during this transformation. Also take j=2 as an example. Since p_2=4 and f_2=2, the Subsystem Arranging Matrix A_2 can be derived with the values in Eq. (4.20).

$$A_2 = \begin{bmatrix} 3 & 3 & \dots & 0 & \dots & 0 \\ 1 & 0 & \dots & 2 & \dots & 0 \\ 0 & 1 & \dots & 2 & \dots & 0 \\ 0 & 0 & \dots & 0 & \dots & 0 \\ 0 & 0 & \dots & 0 & \dots & 1 \\ 0 & 0 & \dots & 0 & \dots & 3 \end{bmatrix} \tag{4.20}$$

In this case, the column vectors of the Subsystem Arranging Matrix A_2 are sorted in falling value, while other sorting approaches are equivalent. Each column vector of A_2 presents one possible permutation for subsystem B. All these permutations have two commons, which are determined by the degree of duplication and fractionation of subsystem B. Firstly, the sum of each column vector is 4, which is equal to p_2. Secondly, the number of non-zero elements of these column vectors is 2, which is equal to f_2.

Furthermore, the total number of permutations is calculated by Eq. (4.13), and more specifically in this case, Eq. (4.21) is used.

$$n_2 = \binom{2}{6}\binom{2-1}{4-1} = 45 \tag{4.21}$$

It is worth noting that the 20$^{\text{th}}$ column vector of A_2 matches the configuration of subsystem j=2, that is, the 2$^{\text{nd}}$ column vector of N, namely

$$A_2(20) = N(2) \tag{4.22}$$

Where, $A_j(k)$ and $N(k)$ are the k^{th} column vectors of the matrices A_j and N respectively. Therefore, we can conclude that q_2=20. Since each column vector is different from the others in the Subsystem Arranging Matrix A_2, this matching process is unique. Similarly, the value of the other elements of the System Sequence Vector Q can also be calculated as shown in Eq. (4.23).

$$Q = \begin{bmatrix} 2 & 20 & 10 & 5 & 30 \end{bmatrix}^{\mathsf{T}} \tag{4.23}$$

The above exemplifies the forward transformation, while the following identifies the process of implementing the backward transformation. Since matrices P, F, and Q are defined, as the derivative of P and F, the System Arranging Matrix A is also available. Therefore, the System Configuration Matrix N can be deduced by column vectors of matrix A, shown in Eq. (4.24).

$$N = \begin{bmatrix} A_1(2) & A_2(20) & A_3(10) & A_4(5) & A_5(30) \end{bmatrix}^{\mathsf{T}} \tag{4.24}$$

4.4.2 Derivation

To achieve the mathematical definition of derivation, the relationships of subsystems in different satellites need to be investigated first, that is, the relationships of the elements in the same column of matrix U.

Start from one of the simplest conditions that each subsystem adopts the same design paradigm or COTS component, so that all the elements of certain subsystem design option, which are in the same column of matrix U, can be considered as identical, as shown in Eq. (4.25) or Eq. (4.26).

$$u_{ij} = u_j (i = 1, 2, \cdots, m) \tag{4.25}$$

Or

$$U_c = \begin{bmatrix} u_1 & u_2 & \ldots & u_n \\ u_1 & u_2 & \ldots & u_n \\ \vdots & \vdots & \ddots & \vdots \\ u_1 & u_2 & \ldots & u_n \end{bmatrix} \tag{4.26}$$

For the condition that subsystems differ from each other in different satellites, the System Derivation Matrix V is introduced by Eq. (4.27), which defines the degree of derivation of each subsystem in each satellite.

$$V = \begin{bmatrix} v_{11} & v_{12} & \dots & v_{1n} \\ v_{21} & v_{22} & \dots & v_{2n} \\ \vdots & \vdots & \ddots & \vdots \\ v_{m1} & v_{m2} & \dots & v_{mn} \end{bmatrix} \tag{4.27}$$

Where, each v_{ij} presents the degree of derivation, in percentage, for each subsystem in each satellite. The degree of derivation is similar with the Heritage Factor that is discussed in *Space Mission Analysis and Design (SMAD)* [86], and defined as the percentage of a subsystem that is identical to previous spacecraft by mass. Therefore, the System Derivation Matrix V denotes the recurring characteristic of subsystem design from previous ones, which may have significant impacts on system properties, such as reliability.

$$U = V \circ U_c \tag{4.28}$$

For real and complex cases like GPS and Galileo, a series of the System Derivation Matrices can be applied to exhibit the evolution of the design of satellites, as presented by Eq. (4.29).

$$U = \sum_{k=1}^{n_g} V_k \circ U_k \tag{4.29}$$

Where, n_g is the generation number of the satellite design, U_k is the Unit Design Property Matrix of generation k, and V_k is its corresponding System Derivation Matrix.

4.5 Summary

Complying with the definitions of the system characteristic space, the mathematical modelling of the design space of the proposed design architecture has been developed. This has established a one-to-one correspondence between the system configuration matrix and the three dimensions of configuration characteristics, i.e., duplication, fractionation, and derivation. Such a design space provides the basis for improving overall value or property such as reliability by altering design configurations, beneficial for the quantitative analysis, design, and optimization of system configurations.

Based on the architecture and design space established, a methodology of analysing and optimizing the entire lifecycle costs of a space system will be developed in the subsequent chapters to exemplify the application of this design architecture. To be specific, Chapter 5 will model and analyse the lifecycle costs, by decomposing it into development, launch, replenishment, and disposal costs. Each of the four costs corresponds to a lifecycle phase, i.e., development, launch, operation, and retirement phases. Furthermore, the integration and implementation of the optimization of the entire lifecycle costs will be carried out in Chapter 6.

Chapter 5

Lifecycle Cost Decomposition and Development

The function of the developed system characteristic space is to integrate the cost or intrinsic properties, e.g., mass, reliability, and orbit, from subsystem level to system level, based on system design variables, i.e., the degree of duplication, fractionation, and derivation. Through appropriate value models, these properties can be measured in the singular monetary dimension.

This chapter models, analyses, and quantifies the lifecycle cost of a space system, by decomposing it into four lifecycle phase costs, i.e., development, launch, replenishment, and disposal costs. Each lifecycle cost corresponds to the cost modelling of a lifecycle phase, i.e., development, launch, operation, and retirement phases, as illustrated in Fig. 1.2. The sum of the costs of these four lifecycle phases can be further applied as a standardised, comparable, and comprehensive metric to enable the optimization process of design configurations to minimise the entire lifecycle costs.

The development of the cost analysis methodology of each lifecycle phase is subject to the process of system configuration design architecture established in Section 4.2. Therefore, in this chapter, the development of each lifecycle phase cost analysis is implemented from the **elaborate**, **analyse**, **evaluate** to **optimize** process.

More specifically, the **elaborate** defines and formulates the design configurations in the proposed system characteristic space. In the **analyse** process, the system properties are analysed to establish the mapping relations between design configurations and system properties. The value models which transform the intrinsic properties into a standardised metric are established to evaluate the system value in the **evaluate** process. Based on the three processes, the **optimize** process construct an optimization problem of system lifecycle phase costs.

5.1 Development Cost Development

Development costs refer to the costs of developing and producing a space system, which can be used to measure the major efforts made in the development phase. In this section, space system development costs are quantified with subsystem cost models as inputs. Through the system characteristic space, subsystem development costs are integrated into the overall system development costs according to different design configurations. The effects of different design configuration characteristics on the development costs are analysed. By adopting the development costs as the objective function, the optimization of the overall system design configurations is realised.

5.1.1 Development Cost Modelling

Before conducting development cost analysis, the development costs for different subsystems are modelled. This acts as the value model for the proposed value-centric design architecture, which is the key to evaluate the system value and enable the value flow in the architecture. As an essential element of the costs in the development phase, the costs of the integration, assembly, and test are also modelled according to different system complexity. Furthermore, the learning curve factor is considered, when a subsystem is reproduced.

(1) Subsystem Cost Models

For traditional satellites, the Unmanned Spacecraft Cost Model (USCM) provides the Cost Estimating Relationships (CERs) for estimating the subsystem costs of a wide range of unmanned Earth-orbit satellites [86, 87]. Shown in Table 5.1, the USCM is a parametric estimating tool that applies statistical regression techniques on a database comprised of actual satellites. The satellite subsystems recognised are listed as follows:

a) Payload Subsystem (PLS)

b) Electrical Power Subsystem (EPS)

c) Telemetry, Tracking and Command (TTC)

d) Command and Data Handling (CDH)

e) Attitude Control Subsystem (ACS)

f) Orbit Control Subsystem (OCS)

g) Thermal Control Subsystem (TCS)

h) Spacecraft Structure (STR)

Table 5.1: Unmanned Spacecraft Cost Model Cost Estimating Relationships (USCM CERs) [86, 87, 151]

Subsystem	CER (FY2010$K)		Cost Driver	Input Range	Standard Error (%)
	Non-Recurring	Recurring			
PLS	$y = 353.3x$	$y = 140x$	PLS Mass	65-395kg	51
EPS	$y = 62.7x$	$y = 112x^{0.763}$	EPS Mass	31-491kg	57
TTC	$y = 545x^{0.761}$	$y = 635x^{0.568}$	TTC Mass	12-65kg	57
CDH	$y = 545x^{0.761}$	$y = 635x^{0.568}$	CDH Mass	12-65kg	57
ACS	$y = 464x^{0.887}$	$y = 293x^{0.777}$	ACS Dry Mass	20-160kg	48
OCS	$y = 17.8x^{0.75}$	$y = 4.97x^{0.823}$	OCS Dry Mass	81-966kg	20
TCS	$y = 394x^{0.635}$	$y = 50.6x^{0.707}$	TCS Mass	3-48kg	61
STR	$y = 157x^{0.83}$	$y = 13.1x$	STR Mass	54-392kg	39

Since it is developed for large spacecraft with high reliability, huge cost, and long lifetime, the USCM CERs are naturally inapplicable for the cost estimates of small satellites driven by improved performance, tighter budgets, and shorter development schedules [152].

In order to address this need, the Aerospace Corporation developed the Small Satellite Cost Model (SSCM) [89] as one of the mainstream cost models for small satellite missions. The latest public version of the SSCM can be found in SMAD [86, 87], and the corresponding CERs are listed in Table 5.2.

Table 5.2: Small Satellite Cost Model Cost Estimating Relationships (SSCM CERs) [86, 87, 151]

System	SSCM (FY2010$K)	Cost Driver	Input Range	Standard Error
PLS	$y = 0.4x$	Bus Total Cost	2600-69000$K	1478
EPS	$y = 1261 + 539x^{0.72}$	EPS Mass	7-70kg	910
TTC	$y = 486 + 55.5x^{1.35}$	TTC Mass	3-30kg	629
CDH	$y = 658 + 75x^{1.35}$	CDH Mass	3-30kg	854
ACS	$y = 1850 + 11.7x^2$	ACS Dry Mass	1-25kg	1113
OCS	$y = 89 + 3.0x^{1.261}$	Bus Dry Mass	20-400kg	310
TCS	$y = 335 + 5.7x^2$	TCS Mass	5-12kg	119
STR	$y = 407 + 19.3xlnx$	STR Mass	5-100kg	1097

Other research conducted by the RAND Corporation [88] revealed typical spacecraft cost fraction characteristics in terms of different mission types by analysing the costs of a wide range of space missions, seen in Table 5.3. Considering the dimensionless forms, such subsystem cost fraction models can be applied in various concept design cases. The types of space missions investigated in this research are respectively:

a) General mission (Gen)

b) Communication mission (Com)

c) Observation mission (Obs)

d) Navigation mission (Nav)

e) Science mission (Sci)

f) Technology demonstration and development mission (Tec)

g) Hybrid mission (Hyb)

Table 5.3: Subsystem cost fraction of the full mission cost by missions (%) [86–88]

System	Gen	Com	Obs	Nav	Sci	Tec	Hyb
PLS	21.1	16.5	18.6	19.4	16.3	19.0	17.5
EPS	12.3	18.0	13.4	18.3	11.2	10.8	16.6
TTC	6.6	7.5	11.4	8.8	14.0	19.7	8.9
ACS	9.7	7.5	17.0	11.9	10.4	8.6	10.9
OCS	4.4	6.2	3.5	6.7	3.3	7.2	5.4
TCS	1.1	2.2	1.2	2.7	1.7	1.3	1.9
TOT	55.3	57.9	65.2	67.8	57.0	66.6	61.2

(2) Subsystem Cost Cases

Cases of four real small satellites recorded in *Reducing Space Mission Cost* [151], textbook examples such as FireSat II and Supplemental Communications System (SCS) from *Space Mission Engineering: The New SMAD* [87], and the online Surrey Satellite Technology Ltd (SSTL) COTS product pricing list [153], are investigated, with the corresponding data collected in Table 5.4. In terms of mass range, Ørsted, PoSAT-1, and SSTL COTS are microsatellites (10-100kg), while the others are small satellites (100-500kg). Due to the application of modularity and COTS products, the costs of PoSAT-1 and SSTL COTS are much lower than the others.

These mission cases will be used in the subsequent sections and chapters for the cost analysis and comparison regrading different design configurations.

(3) Integration, Assembly, and Test (IAT) Cost Modelling

Table 5.4: Cases of space subsystem cost (FY2010$M) [87, 151, 153]

System	Ørsted	Freja	SAMPEX	PoSAT-1	FireSat II	SCS	SSTL COTS
PLS	3.78	9.08	14.81	0.34	8.52	20.45	0.74
EPS	2.57	1.23	7.43	0.54	6.75	15.52	0.38
TTC	1.21	0.75	1.56	0.24	1.49	4.02	0.51
CDH	3.03	1.16	10.44	0.21	2.39	4.28	0.21
ACS	1.06	0.60	2.75	0.31	4.60	10.15	0.70
OCS	0.00	0.87	0.00	0.00	1.37	4.13	0.15
TCS	0.30	0.19	0.13	0.02	0.70	2.02	0.03
TOT	11.96	13.88	37.12	1.65	25.82	60.57	2.72

The Integration, Assembly, and Test (IAT) are major activities in the development phase. However, IAT cost is not considered in detail in traditional design methodologies. This is one of the key factors that result in the problems such as cost growth and schedule delay. To prevent these traditional problems, the effects of IAT are necessary to be integrated into the design process of the proposed architecture.

Fundamental to this endeavour is to establish the cost model of IAT. Generally, the IAT cost of a monolithic spacecraft is 14% of the total cost of its spacecraft bus [86–88]. To decompose it into the subsystem or module level, the IAT cost of each fractionated module can be assumed to be proportional to the number of all the possible interfaces between different subsystems, given by Eq. (5.1). As the number of interfaces increases, the complexity of the entire system increases, which results in the increment of the IAT efforts and costs.

$$c_{\mathrm{IAT},f,i} = \frac{n_{f,i}}{\sum_{i=1}^{m} n_{f,i}} \cdot c_{\mathrm{IAT},m} \tag{5.1}$$

Where, m is the number of modules or satellites in a space system , $c_{\mathrm{IAT},f,i}$ is the IAT cost of the fractionated module i, $c_{\mathrm{IAT},m}$ is the IAT cost of the monolithic system, and $n_{f,i}$ is the number of all the possible interfaces of the fractionated module i. The sum of all the possible interfaces between any two subsystems in a module can be calculated by Eq. (5.2).

$$n_{f,i} = \binom{2}{\sum_{j=1}^{n} n_{ij}} \tag{5.2}$$

Where, $\sum_{j=1}^{n} n_{ij}$ is the sum of row i of the System Configuration Matrix N, namely, the total number of all the different subsystems in the fractionated module i. For monolithic systems, the sum of all the elements in N is adopted, where only one of the rows is nonzero since there is no fractionation in it.

In this model, the IAT cost is assumed to be proportional to the complexity of a module or system, which is expressed by the number of subsystems. As an example for fractionated systems, wireless sharing subsystems increase the complexity of each fractionated module or system, which is the penalty of fractionation. However, on the other hand, fractionation breaks down a monolithic system into several separated modules, which decreases the overall system complexity to some extent.

(4) Learning Curve Factor

When multiple identical subsystems are required, the effect of learning curve factor needs to be considered. This can be implemented as a mathematical technique to account for productivity improvements as a number of units are produced [86–88]. Assuming the expected learning curve slope is s and the number of units being built at the same time is n, the average unit cost c_a can be calculated by Eq. (5.3) [87].

$$c_a = c_{T1} \cdot n^{lns/ln2} \tag{5.3}$$

Where, c_{T1} is the Theoretical First Unit (TFU) cost. The accumulative average learning is typically 0.95 in the aerospace industry [86, 87], the effects of which are shown in Fig. 5.1.

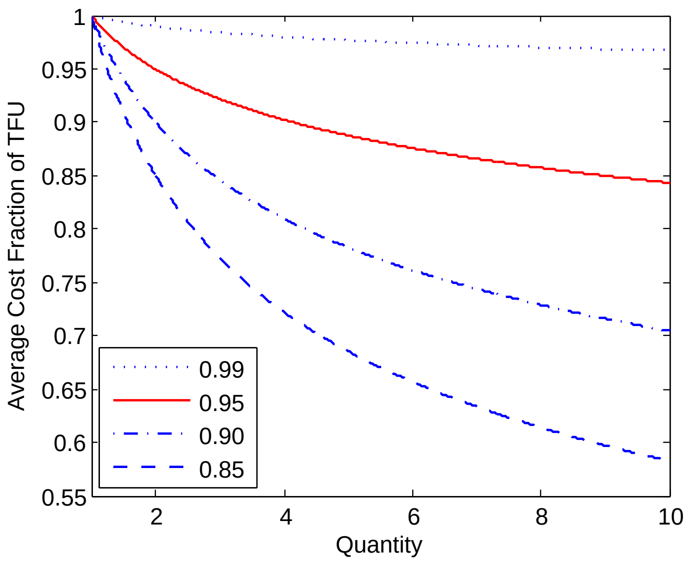


Fig. 5.1: Cost improvements due to learning rate

As can be seen from Fig. 5.1, the average cost decreases asymptotically as the number of units produced increases. This describes the productivity improvement when multiple identical units are manufactured.

5.1.2 Development Cost Analysis

The effects of different design configuration characteristics, i.e., the degree of duplication, fractionation, and derivation on the development costs are analysed in this section. This establishes the mapping relationship between design configurations and development costs, which enables the optimization of design configurations to minimise system development costs. Since the effect of duplication is intuitive, which is proportional, it is not discussed in detail.

(1) Development Cost Analysis on Fractionation

The mass and cost penalty of fractionation is mainly derived from the additional structure for the decomposition of a system and the wireless sharing devices to execute the inter-satellite exchange of information, called Overhead of Fractionation and Wireless Sharing Subsystem (WSS). Functionally, the operation of inter-satellite information exchange is similar to the TTC but for closer range, and thus less powerful systems are required. Since the mass and size of the TTC are inversely proportional to the square of the required transmission distance [87], the cost of inter-satellite information exchange should be far less than that of the TTC. Thus, it is reasonable to make the assumption that the cost of the WSS is lower than that of any subsystem, considering the effect of the fractionation of structure.

As pointed out by Brown [42], it is possible for the potential exchange of information, power, and force between fractionated modules to be realised in the future. This means that the fractionation can be applied to more subsystems, not focused on the TTC and the CDH. Therefore, in the proposed design architecture, the fractionation of any subsystem is enabled if it was realised in the future. However, if not yet, the corresponding subsystem can still be treated as integrated modules with other subsystems such as PLS, during the proposed configuration design process.

Applying the spacecraft cost fraction model shown in Table 5.3, it is possible to compare the development costs of different space system configurations for different mission types. Thus, an example case is provided to analyse the cost characteristics of different design configurations, focused on monolithic and fractionated configurations.

The case is a space system with 5 subsystems labelled from A to E. The subsystem can be all integrated into a monolithic satellite, or fractionated as 5 separate modules connecting by the WSS. For the cases covered, the lowest subsystem cost is adopted as the cost of the WSS. The corresponding input parameters are listed in Table 5.5.

Wherein, the element of the System Duplication Vector respectively represents the number of the subsystem A to E in the entire system. The element of the System Fractionation Vector respectively represents the number of satellites that has the subsystem A to E.

Table 5.5: Inputs of comparisons of system development cost fraction

Parameters	Symbols	Monolithic	Fractionated
Number of satellites	m	1	5
Number of subsystems	n	5	
System Duplication Vector	P	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	
System Fractionation Vector	F	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	

There are three major steps for this simulation approach, and it could be implemented in various program languages such as Matlab. Firstly, in the system characteristic space, the parameters of the System Duplication and Fractionation Vectors are input to derive the System Configuration Matrix. Secondly, through different system cost models, the development costs of monolithic and fractionated systems are calculated. Meanwhile, the costs of the WSS are introduced as the penalties of fractionated configurations. Thirdly, according to the number of satellites and subsystems, the corresponding IAT costs of monolithic and fractionated configurations are estimated and integrated to the above development costs.

As seen in Fig. 5.2, the simulation results indicate that fractionated configurations can slightly reduce the development costs in observation (Obs) and technology demonstration and development (Tec) missions. This cost reduction is a result of using inter-satellite communication, which cuts down numerous efforts of IAT activities. However, such fractionation of different subsystems or modules can also result in cost penalties, which increase the development cost especially in communication (Com) and navigation (Nav) missions. Overall, the cost penalty of fractionation is more than the cost reduction of IAT in most mission types. This favours monolithic configurations as a better design to control development costs.

(2) Development Cost Analysis on Derivation

Assuming that general missions (Gen) are the basic design for the other missions

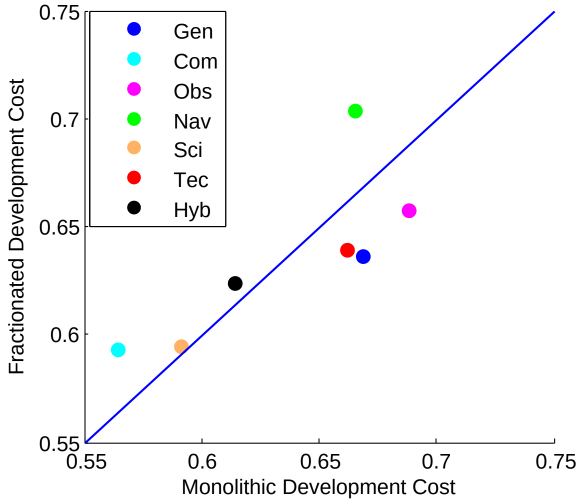
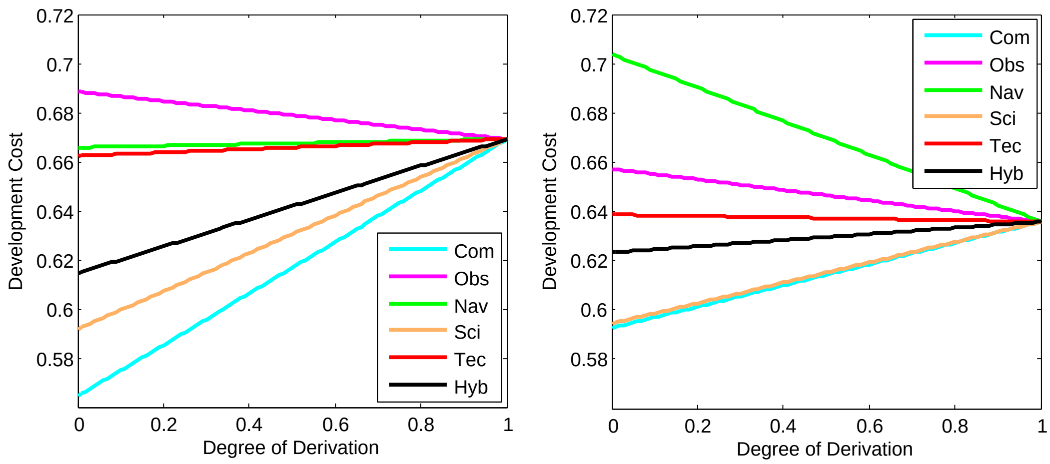


Fig. 5.2: Comparison of development cost fraction of the total mission cost for different systems

(Com, Obs, Nav, Sci, Tec, and Hyb) to be derivative from, development costs are analysed to show this derivative characteristic. The results of these analyses are shown in Fig. 5.3, where it is assumed that the other types of missions are derivative from Gen. Moreover, the simulations are executed by mathematically integrating the cost of derivative and non-derivative modules or subsystems based on the percentage of mass.



(a) Monolithic systems (b) Fractionated systems

Fig. 5.3: Development cost of different degree of derivation

It is apparent from Fig. 5.3 that the effects of derivation on system development costs are linear, and proportional to the degree of derivation. The cost relationship between each specific mission and the general mission is also presented, where 100% degree of derivation denotes Gen missions and 0% degree of derivation denotes specific missions. As can be seen in Fig. 5.3, regardless of configuration designs, Nav and Obs missions are typically more costly than Gen in the development phase, while Sci and Com missions are generally less expensive.

Through comparing the points at 0% degree of derivation in Fig. 5.3, different system designs have their own advantages in different types of space missions. Fractionated systems are more appropriate for communication (Com) and science (Sci) missions to slightly reduce the development costs, while monolithic systems support the reduction in the development costs of most types of space missions. Due to the mass and cost penalties, fractionated systems are typically more costly than monolithic systems in the development phase.

5.1.3 Development Cost Integration

Having analysed the effects of different system characteristics, the integration of the overall development costs is executed in the **evaluate** process, generating an appropriate value for the analysis and optimization of satellite designs. The system value is then integrated in the system characteristic space, which builds a bridge between system design configurations and development costs. The corresponding calculation procedures are illustrated in Fig. 5.4.

1) The system design configuration is defined by the degree of duplication, fractionation, and derivation, which can be equivalently transformed into a more intuitive expression, that is, the system configuration matrix. This step transforms the input parameters into the intermediate parameters for the following calculation process. The three parameters of duplication, fractionation, and derivation before the transformation are easier to be defined, while the system configuration matrix acquired after the transformation can be directly used for the analysis of different properties and costs.

2) The subsystem cost models described in Section 5.1.1 and wireless sharing cost penalty are applied to calculate all the subsystem development costs of each satellite in a space system, through the system configuration matrix. For the same input parameters of duplication and derivation, the satellite development costs for both monolithic and fractionated systems are the same, while the cost penalties of fractionated systems increase as the degree of fractionation increases.

3) The IAT cost model described in Section 5.1.1 is applied to calculate the IAT

costs of each satellite in a space system, through the system configuration matrix. The IAT cost for each satellite is calculated independently, and that of the entire system is the sum of satellite IAT costs.

4) The IAT costs and the subsystem development costs are then summed up to generate the development costs of each satellite. Finally, the overall development costs are the sum of the development costs of all the satellites in this system.

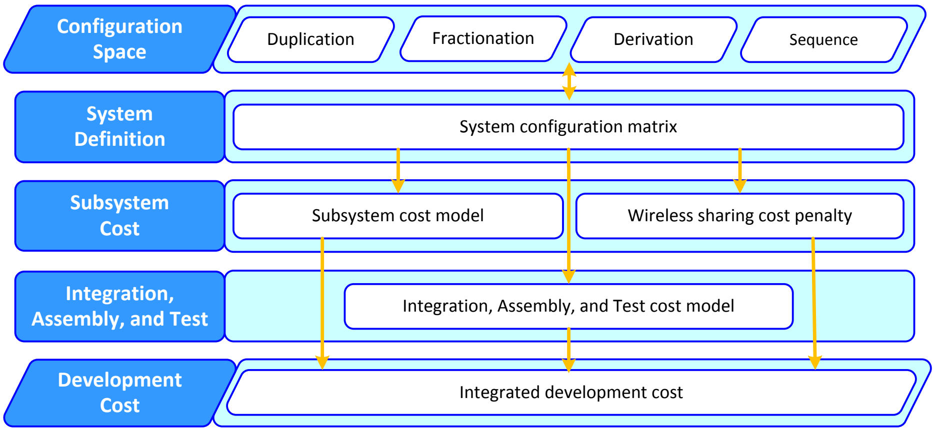


Fig. 5.4: System development cost integration process

The system designs of a hybrid spacecraft are performed as an example. The input values are shown in Table 6.6. In this example, the space system is assumed to have 6 satellites labelled from 1 to 6, and 5 types of subsystems labelled from A to E in each satellite. For a system design, there can be less than 6 satellites, and each can have any combination of 5 types of given subsystems.

The elements of the System Duplication Vector P respectively define the number of the subsystem A to E in the entire system. The elements of the System Fractionation Vector F respectively define the number of satellites with the subsystem A to E equipped. The elements of the System Sequence Vector Q respectively define the sequence number of the permutation of the subsystem A to E in the system.

More specifically, in the example, there are 2 subsystems A, 4 subsystems B, 2 subsystems C, 1 subsystem D, and 3 subsystems E. Meanwhile, all the subsystems A, B, C, and E are allocated in 2 satellites, while all the subsystems D are allocated in 1 satellite. It is worth noting that the allocated 2 satellites for the subsystem A, B, C, and E can be different. Furthermore, the System Sequence Vector Q provides the label of the satellites that have the subsystem A to E.

The corresponding outputs are exhibited in Table 5.6, according to different subsystem cost models and cases established in Section 5.1.1. To support comparisons, all the development costs are normalised by the original monolithic spacecraft costs to give the percentage in Table 5.6.

Table 5.6: Inputs of system development cost integration

Parameters	Symbols	Values
Number of satellites	m	6
Number of subsystems	n	5
System Duplication Vector	P	$\begin{bmatrix} 2 & 4 & 2 & 1 & 3 \end{bmatrix}^\intercal$
System Fractionation Vector	F	$\begin{bmatrix} 2 & 2 & 2 & 1 & 2 \end{bmatrix}^\intercal$
System Sequence Vector	F	$\begin{bmatrix} 2 & 20 & 10 & 5 & 30 \end{bmatrix}^\intercal$

Table 5.7: Outputs of system development cost integration (All the results are normalised to the fraction of the original monolithic spacecraft costs.)

Input Models	Development Cost (%)	Input Models	Development Cost (%)
Ørsted	101.63	Gen	100.69
Freja	98.47	Com	106.24
SAMPEX	97.93	Obs	101.16
PoSAT-1	100.38	Nav	106.69
FireSat II	103.97	Sci	102.19
SCS	105.33	Tec	99.42
SSTL COTS	99.04	Hyb	104.37

As can be seen in Table 5.6, fractionated designs may have less development costs in some space missions, while monolithic designs may have less costs in other missions. However, since the System Sequence Vector Q of fractionated designs is alterable during the simulations, it is feasible to seek the optimal Q, namely, the best design configuration with all available subsystems suitably arranged to reach the minimum development costs.

Furthermore, for the subsystems that cannot be practically fractionated from others, they can be treated as integrated modules with other subsystems such as PLS, while the concrete calculation processes and optimization approaches will be introduced in Chapter 6.

5.1.4 Development Cost Optimization

Having modelled the development costs and analysed the effects of various design configurations on the development costs, this section completes the final connection, namely, the **optimize** process, for the proposed value-centric design loop, enabling the system to be designed through a global optimization process.

This optimization process is similar to the previous integration process, but free of the system design configuration, or mathematically the System Sequence Vector Q. It acts as the variables of the optimization problem. The development costs are selected as the objective function. Furthermore, this optimization process can be executed in the proposed system characteristic space. In summary, the proposed optimization problem regarding system development costs can be mathematically expressed by Eq. (5.4).

$$\begin{aligned} \min \quad & J(x) \\ \text{s. t.} \quad & x_j^l \leq x_j \leq x_j^u : \text{integer} \qquad j = 1, 2, \ldots, n \\ & x = \begin{bmatrix} x_1 & x_2 & \ldots & x_n \end{bmatrix}^\intercal \end{aligned} \tag{5.4}$$

Where, $J(x)$ is the objective function, n is the number of state variables, x_j is the state variable, and x_j^l and x_j^u are the corresponding lower and upper bounds respectively.

The optimization process of development cost is shown in Fig. 5.5. Throughout this process, the development costs of the entire space system is the objective function, and the design configurations of the system are the state variables.

In the design space, the design configurations can be expressed by the duplication, fractionation, derivation, and sequence. Since the former three parameters are fixed, the System Sequence Vector is the only variable in this optimization process. Associated with the degree of duplication, fractionation, and derivation, the System Sequence Vector can be transformed to the System Configuration Matrix. Applying the development cost integration procedures described in Section 5.1.3, the total system development cost can be calculated from this design matrix. This development cost is adopted as the objective function during the entire optimization process. When reaching the best objective function, the corresponding variable is the solution, namely, the System Sequence Vector. Through the proposed transformation, the intuitive design configuration can also be acquired using this system characteristic parameter and

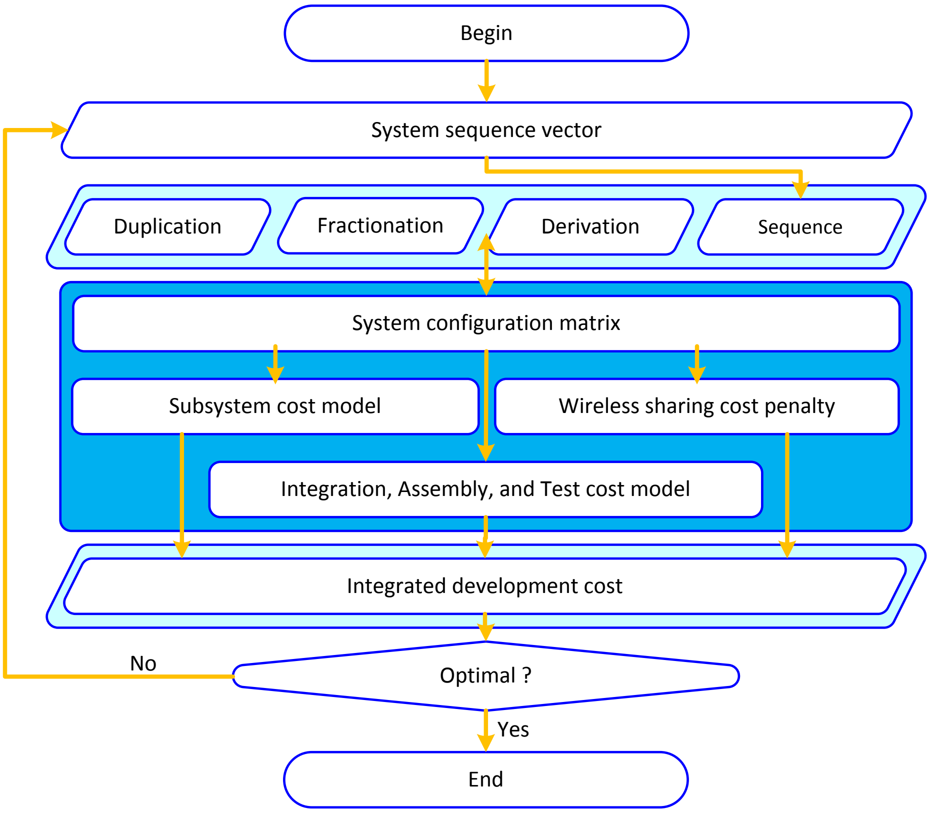


Fig. 5.5: System development cost optimization process

other input parameters.

Furthermore, since the input variables are all integers, the overall optimization can be considered as a Mixed Integer Programming Problem (MIPP), or more specifically, a Mixed Integer Nonlinear Programming (MINLP) problem. The corresponding solution process will be described and exemplified in Section 6.1.1.

5.1.5 Summary

This section models, analyses, and quantifies the costs in the development phase of a space system. The subsystem development cost models are established as the inputs to calculate the system development costs through the system characteristic space. The effects analysis of different design configurations on the development costs are performed to build the mapping relationship between design configurations and development costs. This enables the optimization process of system design configurations to minimise the overall development costs.

5.2 Launch Cost Development

This section formulates the costs in the launch phase, with subsystem cost and mass properties as inputs. Real data of various launch vehicle families, such as cost, total attempts, and success rate, are initially collected to develop a launch vehicle database. Based on this database, an estimate the reliability of each launch vehicle family is made, using a modified two-level Bayesian analysis. The factors of both launch cost and reliability are subsequently merged into the expected launch cost, acting as the comprehensive metric to evaluate the cost of launch activities.

5.2.1 Launch Cost Modelling

Before performing the launch cost analysis, the launch vehicle database is established to provide the data for the modelling of launch reliability and cost. The launch reliability is modelled by the two levels of the Bayesian analysis, while the cost is estimated for a dedicated launch. The estimate of launch reliability also influences the manufacturing cost, since launch failures might result in producing multiple copies of satellites. The reliability and cost information are later integrated into the expected launch cost, which shows the statistical expectation of the cost to ensure inserting a satellite into orbit at given confidence level. The expected launch cost can be used as an objective function for the optimization process, associated with the manufacturing cost of satellites.

(1) Launch Vehicle Database

For this research, the launch vehicle database established in our previous research [21, 22], has been updated to April 2018, based on online space launch reports [154–156]. The launch vehicle data is listed in Table 5.8 and Table 5.9 for active and retired vehicles respectively. It is worth noting that all the launch vehicles are categorized into small ($<$5000lb, $\sim$2268kg), medium ($<$12000lb, $\sim$5443kg), intermedium ($<$25000lb, $\sim$11340kg), and heavy ($\geq$25000lb, $\sim$11340kg) classes, according to their launch capabilities in pounds to LEO[87].

It is noteworthy that piggyback and rideshare launches are not considered in this research, due to significant uncertainty in costs. Moreover, this approach could also be further adapted for piggyback and rideshare launches if the corresponding model

Table 5.8: Launch data for active launch vehicle families*

Launch vehicle family	Country	Mass class	Capability to LEO (kg)	Capability to GTO (kg)	Payload length (m)	Payload diameter (m)	Launch cost (FY2010$M)	Number of successes	Number of failures
Ariane 5G & 5E	Europe	Heavy	21000	6800	16.19	4.57	175.99	92	5
Atlas 5 400s	USA	Heavy	15700	7700	10.31	3.75	152.00	51	1
Atlas 5 500s	USA	Heavy	20520	8900	12.92	4.57	172.50	23	0
Delta 2 6000s & 7000s	USA	Medium	5144	1800	6.83	2.74	74.91	152	2
Delta 4M, 4M+, 4H	USA	Heavy	22560	13130	15.71	4.57	215.00	35	1
Dnepr	Russia	Medium	4400	0	4.20	2.70	20.43	21	1
Epsilon	Japan	Small	1200	0	5.39	2.12	38.00	3	0
Falcon 9 V1.0, 1.1, FT	USA	Heavy	22800	8300	11.00	4.60	56.22	45	3
Falcon Heavy	USA	Heavy	63800	26700	13.90	5.20	81.61	1	0
GSLV	India	Medium	5000	2350	7.30	3.05	44.00	6	5
H-2 & 2A	Japan	Heavy	11730	5000	10.23	3.70	101.25	41	3
H-2B	Japan	Heavy	16500	8000	9.12	4.60	142.42	6	0
Kosmos 3M	Russia	Small	1500	0	4.72	2.40	18.39	424	22
KZ	China	Small	300	0	1.40	1.20	2.61	3	0
Long March 2C & 2D	China	Medium	3500	1000	5.00	3.00	20.49	81	2
Long March 2E & 2F	China	Intermedium	8400	3500	6.54	3.80	68.10	18	2
Long March 3A & 3C	China	Intermedium	9100	3800	5.25	3.00	102.45	40	0
Long March 3B	China	Heavy	13600	5200	6.85	3.85	81.72	41	3
Long March 4A, 4B, 4C	China	Medium	4200	1500	6.50	3.35	47.81	50	2
Long March 5	China	Heavy	23000	13000	10.25	4.50	150.10	1	1
Long March 6	China	Small	1500	0	4.60	2.20	13.05	2	0
Long March 7	China	Heavy	13500	5500	6.85	3.80	87.45	2	0
Long March 11	China	Small	700	0	2.00	1.60	6.09	3	0
Minotaur I & IV	USA	Small	1735	0	5.71	2.04	45.96	17	0
Minotaur V	USA	Small	630	630	4.02	2.04	45.96	1	0
Pegasus & Pegasus XL	USA	Small	443	0	2.12	1.16	18.45	37	6
Proton K & M	Russia	Heavy	21000	5500	9.86	3.86	141.15	367	46
PSLV, PSLV-CA, & XL	India	Medium	3800	1300	5.40	2.90	20.11	39	3
Rockot	Russia	Small	1950	0	6.21	2.38	18.45	27	3
Shavit 1 & 2	Israel	Small	800	0	2.84	1.30	20.49	8	2
Shtil 1	Russia	Small	140	0	1.47	0.63	2.20	2	0

Soyuz 2.1	Russia	Intermedium	8200	3250	9.51	3.80	54.65	65	5
Soyuz U	Russia	Intermedium	7000	1660	9.00	2.85	54.65	764	21
Start-1	Russia	Small	632	0	2.31	1.24	12.29	6	0
Strela	Russia	Small	1560	0	2.95	2.20	14.34	3	0
Taurus & Taurus XL	USA	Small	1590	557	5.71	2.04	45.26	7	3
Vega	Europe	Small	1963	0	6.30	2.30	35.00	11	0
Volna	Russia	Small	140	0	1.25	0.82	1.57	2	3
Zenit 3 SL, SLB, SLBF	Russia	Heavy	15876	6066	8.53	3.75	115.77	42	4

* Small, medium, intermedium, and heavy launch vehicles are identified by payload capabilities in kg to LEO (<2268, <5443, <11340, ≥ 11340).

Table 5.9: Launch data for retired launch vehicle families*

Launch vehicle family	Country	Mass class	Capability to LEO (kg)	Capability to GTO (kg)	Payload length (m)	Payload diameter (m)	Launch cost (FY2010$M)	Number of successes	Number of failures
Athena 1 & 2	USA	Small	2065	0	4.29	2.05	32.69	5	2
Ariane 4	Europe	Intermedium	10200	4790	11.12	3.65	153.23	113	3
Atlas 2, 2A, 2AS	USA	Intermedium	8618	3719	9.39	3.75	132.80	63	0
Atlas 3	USA	Intermedium	10759	4119	10.31	3.75	141.62	6	0
Cyclone 2 & 3	Russia	Medium	4100	0	6.60	2.40	28.28	219	9
Delta 2000s	USA	Small	1860	724	4.93	2.18	57.90	43	1
Delta 3 8000s	USA	Intermedium	8290	3810	9.93	3.75	122.94	1	2
Delta 3000s, 4000s, 5000s	USA	Medium	3848	1405	6.12	2.54	67.85	38	3
Falcon 1	USA	Small	668	0	2.79	1.37	6.95	2	3
Long March 2A	China	Small	2000	0	5.00	3.00	28.80	3	1
Molniya-M	Russia	Medium	3700	0	4.04	2.35	47.81	276	21
M-V	Japan	Small	1900	1280	6.03	2.20	71.65	6	1
Soyuz	Russia	Intermedium	7000	1350	5.86	3.43	51.08	30	2
Space Shuttle	USA	Heavy	28803	5900	18.3	4.57	408.60	133	2
Titan 2	USA	Small	1900	0	6.65	2.83	47.81	13	0
Titan 3A & 3B	USA	Medium	3300	0	6.75	2.83	49.45	66	6
Titan 3C, 3D, 3E, 34D	USA	Heavy	14515	5000	8.27	2.83	250.93	71	9
Titan 4A & 4B	USA	Heavy	21600	5760	18.23	4.57	546.38	35	4
VLS-1	Brazil	Small	380	0	2.05	1.10	9.42	0	3
Zenit 2	Russia	Heavy	13740	0	12.24	3.40	57.89	30	8

* Small, medium, intermedium, and heavy launch vehicles are identified by payload capabilities in kg to LEO (<2268, <5443, <11340, ≥ 11340).

and data were available.

(2) Bayesian Analysis of Launch Reliability

Assuming the probability of successes of a family of launch vehicles remains constant for every experiment, the space launch activities can be modelled as a series of Bernoulli trials [157], which are defined as random experiments with exactly two outcomes: success and failure. Our work is to achieve the best estimate of this probability of success.

One of the biggest challenges in the modelling of space launch activities, distinguishing it from other Bernoulli trials such as coin-toss problems, is the limited sample size. This may increase the inaccuracy of the estimate. The Bayesian method is proposed to overcome this problem [158], as it incorporates the prior knowledge of the known launch vehicles into the record of the observed launches of the launch vehicles in question, shown in Eq. (5.5).

$$f_{\mathrm{A|D}}(a|s,f) = \frac{f_{\mathrm{D|A}}(s,f|a)f_{\mathrm{A}}(a)}{\int f_{\mathrm{D|A}}(s,f|x)f_{\mathrm{A}}(x)dx} \tag{5.5}$$

Where, A is the probability of successful launches, given s successes and f failures in the past t=s+f trials. The probability density function $f_{\mathrm{A}}(a)$ is the prior distribution, representing the state of knowledge about a given launch vehicle prior to any of the t launch trials. The likelihood function $f_{\mathrm{D|A}}(s,f|a)$ denotes the probability of the observed data D, namely, of achieving s successes and f failures in t launch attempts. The result of the calculation $f_{\mathrm{A|D}}(a|s,f)$ is the posterior distribution, which updates the previous knowledge with the realized launch data.

Howard [159] derived the general form of the Bayesian probability and demonstrated that the mean of the posterior distribution is the optimal estimate of the probability of successes. Based on these mathematical fundamentals, Guikema proposed using beta distribution as the conjugate prior distribution for modelling and analysing the reliability of launch vehicles [108]. The corresponding prior distribution is given by Eq. (5.6).

$$f_{\mathrm{A}}(a|s_0,f_0) = \frac{\Gamma(s_0+f_0)}{\Gamma(s_0)\Gamma(f_0)}p^{s_0-1}(1-p)^{f_0-1} \tag{5.6}$$

Where, $\Gamma(n)$ is the gamma function with the parameter n, shown in Eq. (5.7).

$$\Gamma(n) = \int_0^\infty t^{n-1}e^{-t}dt \tag{5.7}$$

The mean and variance of the beta distribution can be presented by the parameter s and f, given by Eq. (5.8) and Eq. (5.9).

$$\overline{p} = \frac{s}{s+f} \tag{5.8}$$

$$\sigma = \frac{sf}{(s+f)^2(s+f+1)} \tag{5.9}$$

If the updated observed data of s successes and f failures can be considered as a Bernoulli process, the posterior distribution is also a beta distribution with the parameters s_0+s and f_0+f.

$$f_{\mathrm{A|D}}(a|s,f) = \frac{\Gamma(s_0+s+f_0+f)}{\Gamma(s_0+s)\Gamma(f_0+f)} p^{s_0+s-1}(1-p)^{f_0+f-1} \tag{5.10}$$

The two levels of the Bayesian analysis proposed by Guikema [108], are adopted in this research for the reliability analysis of different types of launch vehicles. The first level applies the uniform distribution for the prior distribution of the Bayesian method. In this case, we assume that we know nothing about the reliability of these launch vehicles in advance. This level is appropriate for those launch vehicles with sufficient updating data, so that the prior distribution will have little influence on the posterior distribution, since the data dominates the updating process.

We fit the second-level prior distributions by combining the means of the first-level posterior distributions for all the launch vehicles except the one in question. This incorporates all the experience of the known launch vehicles. In this case, we assume that the probability of success for emerging launch vehicles is similar to previous ones, since many of the lessons learned in previous generations have been widely shared. The recorded data of all the other launch vehicles acts as the prior knowledge of launch activities, while the data of the launch vehicle to be investigated is used for the updating process. Therefore, this level of the Bayesian analysis can compensate for the lack of information in new launch vehicles or ones with limited launch trials. In addition, removing the information of the vehicle to be investigated in the prior distribution can solve the redundancy problem existing in the original second-level Bayesian analysis proposed by Guikema [108].

The results of the two levels of the Bayesian analysis for different families of launch vehicles are summarised in Table 5.10, with the corresponding success rates as the references. Overall, the Delta 2 family is the most reliable current launch vehicle, while the reliability estimate of the Soyuz U family exhibits the lowest uncertainty or variance, as a result of the sheer number of launches. Among the launch vehicles with limited trials, the Minotaur and Vega families have the highest expected reliability.

As stated above, different levels of reliability analysis are appropriate for different vehicles. In this research, we assume that the launch vehicles with at least 20 launches are considered to have comparatively sufficient updating data, thus are considered using only the first-level analysis. On the contrary, the launch vehicles with less than 20 launches are classified as relatively novel or infrequently used, and considered using the second-level analysis. Furthermore, the probability to be applied for each launch vehicle family is highlighted in bold in Table 5.10.

Thus, the posterior distributions of the Bayesian probability for different families of launch vehicle are shown in Fig. 5.6 and Fig. 5.7 respectively. In both figures, the x-axis denotes the value of launch reliability, and the y-axis shows the corresponding probability density.

As shown in Fig. 5.7, the absence of the intermedium class indicates that all the active launch vehicles in this class are appropriate for the first level analysis. In other words, most novel launch vehicles are in the smaller classes, while the other classes are mainly mature launch vehicles. Additionally, it is noteworthy that the launch vehicles lacking in capability or cost information, e.g., SS-520, are not included here, as additional information about these systems is necessary in order to apply launch cost analysis and optimization.

(3) Expected Launch Cost

Expected launch cost refers to the statistical expectation of the cost of a successful launch, i.e., successfully insert the payload into an effective orbit, which is the combination of launch cost and launch reliability, given by Eq. (5.11).

$$c_e = n_e \cdot c_l \tag{5.11}$$

Where, c_e denotes the proposed expected launch cost, c_l is the dedicated launch cost of single launch attempt, and the number of the statistical average launch attempts n_e to achieve a successful one is defined by Eq. (5.12).

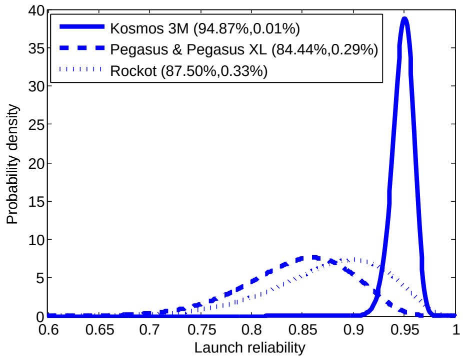
$$1 - (1 - r)^{n_e} = 1 - \alpha \tag{5.12}$$

Where, r is the reliability of the launch vehicle family and α is the significance level

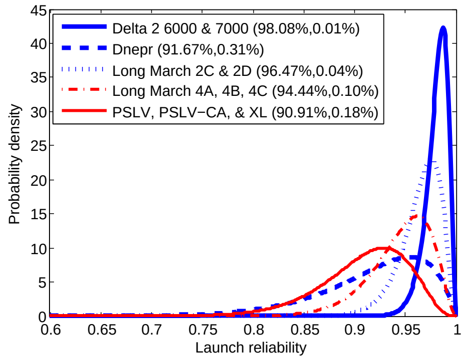
Table 5.10: Means and variances of the posterior probability distribution for active launch vehicle families*

Launch vehicle family	Successes/ attempts	Success rate (%)	First level posterior		Second level posterior	
			Means(%)	Variances(%)	Means(%)	Variances(%)
Ariane 5G & 5E	92/97	94.85	**93.94**	**0.06**	94.35	0.05
Atlas 5 400s	51/52	98.08	**96.30**	**0.06**	96.93	0.05
Atlas 5 500s	23/23	100.00	**96.00**	**0.15**	97.35	0.09
Delta 2 6000s & 7000s	152/154	98.70	**98.08**	**0.01**	98.28	0.01
Delta 4M, 4M+, 4H	35/36	97.22	**94.74**	**0.13**	95.72	0.10
Dnepr	21/22	95.45	**91.67**	**0.31**	93.44	0.23
Epsilon	3/3	100.00	80.00	2.67	**90.12**	**1.12**
Falcon 9 V1.0, 1.1, FT	45/48	93.75	**92.00**	**0.14**	92.87	0.12
Falcon Heavy	1/1	100.00	66.67	5.56	**86.29**	**1.98**
GSLV	6/11	54.55	53.85	1.78	**62.29**	**1.46**
H-2 & 2A	41/44	93.18	**91.30**	**0.17**	92.28	0.15
H-2B	6/6	100.00	87.50	1.22	**93.01**	**0.59**
Kosmos 3M	424/446	95.07	**94.87**	**0.01**	94.95	0.01
KZ	3/3	100.00	80.00	2.67	**90.12**	**1.12**
Long March 2C & 2D	81/83	97.59	**96.47**	**0.04**	96.88	0.03
Long March 2E & 2F	18/20	90.00	**86.36**	**0.51**	88.76	0.40
Long March 3A & 3C	40/40	100.00	**97.62**	**0.05**	98.36	0.04
Long March 3B	41/44	93.18	**91.30**	**0.17**	92.28	0.15
Long March 4A, 4B, 4C	50/52	96.15	**94.44**	**0.10**	95.16	0.08
Long March 5	1/2	50.00	50.00	5.00	**72.37**	**2.79**
Long March 6	2/2	100.00	75.00	3.75	**88.52**	**1.46**
Long March 7	2/2	100.00	75.00	3.75	**88.52**	**1.46**
Long March 11	3/3	100.00	80.00	2.67	**90.12**	**1.12**
Minotaur I & IV	17/17	100.00	94.74	0.25	**96.61**	**0.15**
Minotaur V	1/1	100.00	66.67	5.56	**86.29**	**1.98**
Pegasus & Pegasus XL	37/43	86.05	**84.44**	**0.29**	85.75	0.25
Proton K & M	367/413	88.86	**88.67**	**0.02**	88.80	0.02
PSLV, PSLV-CA, & XL	39/42	92.86	**90.91**	**0.18**	91.95	0.16
Rockot	27/30	90.00	**87.50**	**0.33**	89.12	0.28
Shavit 1 & 2	8/10	80.00	75.00	1.44	**80.76**	**1.04**
Shtil 1	2/2	100.00	75.00	3.75	**88.52**	**1.46**
Soyuz 2.1	65/70	92.86	**91.67**	**0.10**	92.29	0.09
Soyuz U	764/785	97.32	**97.20**	**0.00**	97.25	0.00
Start-1	6/6	100.00	87.50	1.22	**93.01**	**0.59**
Strela	3/3	100.00	80.00	2.67	**90.12**	**1.12**
Taurus & Taurus XL	7/10	70.00	66.67	1.71	**73.65**	**1.30**
Vega	11/11	100.00	92.31	0.51	**95.29**	**0.28**
Volna	2/5	40.00	42.86	3.06	**60.03**	**2.33**
Zenit 3 SL, SLB, SLBF	42/46	91.30	**89.58**	**0.19**	90.60	0.17

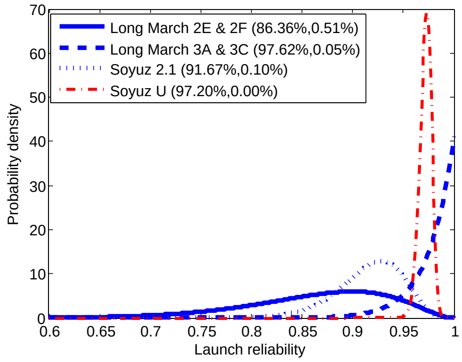
* The means and variances of the posterior probability to be used for each family of launch vehicles are highlighted in bold.



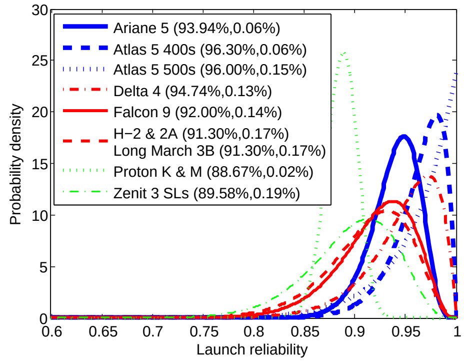
(a) Small launch vehicles



(b) Medium launch vehicles

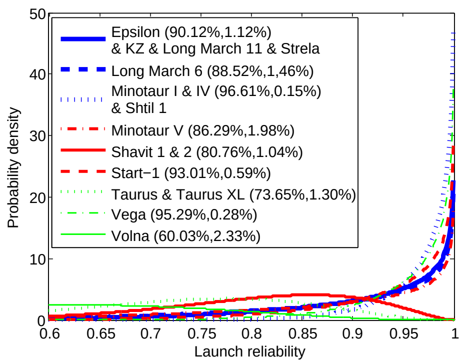


(c) Intermedium launch vehicles

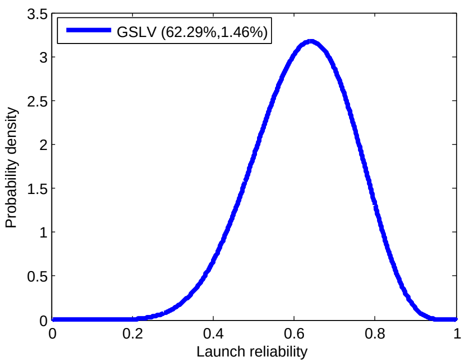


(d) Heavy launch vehicles

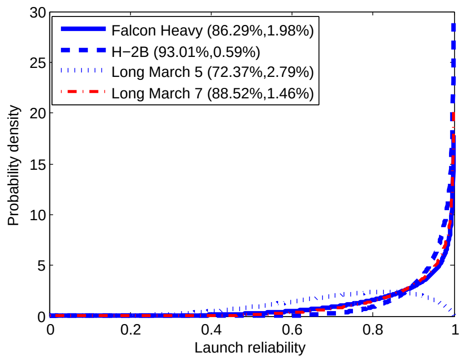
Fig. 5.6: First level posterior probability function with at least 20 launches



(a) Small launch vehicles



(b) Medium launch vehicles



(c) Heavy launch vehicles

Fig. 5.7: Second level posterior probability function with less than 20 launches

or $1-\alpha$ the confidence level (a measure of how confident we want to be). Therefore, the expected launch cost including reliability and confidence is given by Eq. (5.13).

$$c_e = c_l \cdot log_{1-r}\alpha \tag{5.13}$$

With the unit launch cost and reliability of different launch vehicles as the inputs, the expected launch cost can be obtained. Thus, the expected number of launch attempts and the corresponding cost of different launch vehicle families can be calculated by Eq. (5.13). Assuming the confidence level of 0.99, i.e., taking α as 0.01, Table 5.11 gives the corresponding expected launch costs. From Table 5.11, the optimal launch strategy can be considered, once the payload mass is determined. It is also noteworthy that, in the table, the costs of satellites are not included.

In Table 5.11, the number of expected launches expresses the reliability of different families of launch vehicles. The more reliable a launch vehicle is, the less number of expected launches it requires to get a 99% confindence for launch missions. As pointed out that Delta 2 family is the most reliable, it requires the smallest number of expected launches.

The expected launch cost incorporates the cost and reliability of a launch vehicle. Despite being the most reliable launch vehicle, the expected launch cost of Delta 2 family is not the best. Dnepr, Long March 2C & 2D, and PSLV perform better than Delta 2, due to the cost of single launch attempt, within the medium class. However, for the decision making process of real launch cases, the cost of satellites to be launched may also be considered.

5.2.2 Launch Cost Optimization

To enable the optimization process to seek an appropriate launch vehicle, the expected launch cost is used as the objective function, associated with the development cost of the satellites to be launched.

The optimization of space system launch cost can also be concluded as a MINLP problem, given by Eq. (5.4). The corresponding calculation procedures are shown in Fig. 5.8. Throughout the optimization process, the objective function $J(x)$ is the launch cost under risk in this phase, which is described in the following.

Table 5.11: Expected number of launches required to get a 99% confidence in deploying necessary satellites in orbit and the corresponding cost for active launch vehicle families

Launch vehicle family	Expected launches	Single cost (FY2010$M)	Expected cost (FY2010$M)
Ariane 5G & 5E	1.64	175.99	289.10
Atlas 5 400s	1.40	152.00	212.38
Atlas 5 500s	1.43	172.50	246.79
Delta 2 6000s & 7000s	1.17	74.91	87.31
Delta 4M, 4M+, 4H	1.56	215.00	336.26
Dnepr	1.85	20.43	37.86
Epsilon	2.55	38.00	96.82
Falcon 9 V1.0, 1.1, FT	1.82	56.22	102.51
Falcon Heavy	3.38	81.61	275.62
GSLV	5.52	44.00	242.72
H-2 & 2A	1.89	101.25	190.91
H-2B	2.07	142.42	295.15
Kosmos 3M	1.55	18.39	28.52
KZ	2.50	2.61	6.52
Long March 2C & 2D	1.38	20.49	28.22
Long March 2E & 2F	2.31	68.10	157.40
Long March 3A & 3C	1.23	102.45	126.23
Long March 3B	1.89	81.72	154.09
Long March 4A, 4B, 4C	1.59	47.81	76.17
Long March 5	5.22	150.10	783.13
Long March 6	2.84	13.05	37.00
Long March 7	2.87	87.45	251.13
Long March 11	2.50	6.09	15.25
Minotaur I & IV	1.49	45.96	68.58
Minotaur V	3.37	45.96	154.98
Pegasus & Pegasus XL	2.47	18.45	45.66
Proton K & M	2.11	141.15	298.43
PSLV, PSLV-CA, & XL	1.92	20.11	38.62
Rockot	2.21	18.45	40.86
Shavit 1 & 2	3.13	20.49	64.18
Shtil 1	2.80	2.20	6.16
Soyuz 2.1	1.85	54.65	101.28
Soyuz U	1.29	54.65	70.35
Start-1	2.03	12.29	24.95
Strela	2.52	14.34	36.20
Taurus & Taurus XL	3.93	45.26	177.96
Vega	1.69	35.00	59.32
Volna	6.87	1.57	10.78
Zenit 3 SL, SLB, SLBF	2.04	115.77	235.72

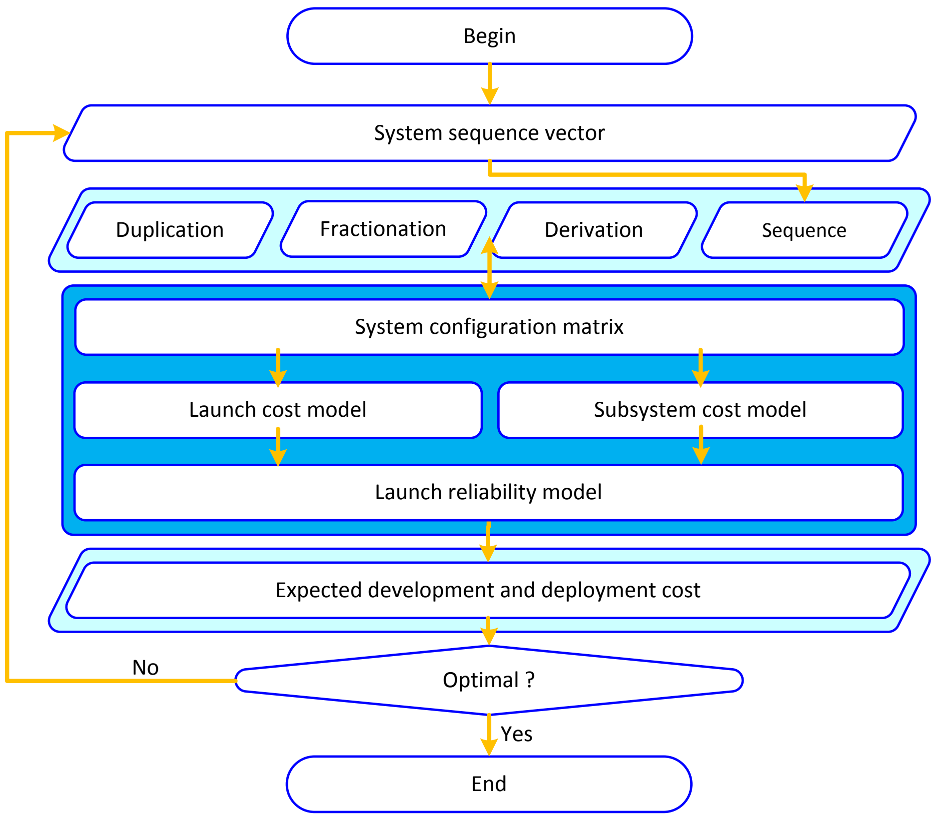


Fig. 5.8: System launch cost optimization process

Regarding the objective function of the launch cost optimization process, the expected launch cost is one of the simplest solutions. However, if the design and development cost of space systems is not considered, the optimizing process is likely to distort the facts. In order to distinguish different risk control requirements, the development and deployment cost is adopted as the objective function of the optimization process, including both the expected launch cost and the cost of the satellites to be launched. Mathematically, the expected development and deployment cost, c_d, is given by Eq. (5.14).

$$c_d = (c_l + c_s) \cdot log_{1-r}\alpha \tag{5.14}$$

Where, c_s is the development cost of the launched satellites. Furthermore, the learning curves factor given by Eq. (5.3) is also used to predict the average unit cost of the launched satellites, when more than one identical or near identical satellites are produced.

Furthermore, the state variables x are the system design configurations, or mathematically the System Sequence Vector Q, which are used to define the combinations and permutations of a space system considering duplication and/or fractionation. As the elements of Q are all integers, this optimization problem is a MINLP problem.

Applying the launch and subsystem cost models, the cost of launching and deploying a space system is estimated using the System Configuration Matrix. This cost is subsequently integrated into the expected development and deployment cost by incorporating the effects of launch reliability. Adopting this expected cost as the objective function, the optimization algorithms can be applied to seek the optimal design solution.

5.2.3 Summary

This section models, analyses, and quantifies the costs in the launch phase of a space system. The development cost models established in the previous chapter are inputs to calculate the system launch costs, associated with mass property. The real data of various launch vehicle families, such as cost, total attempts, and success rate, are initially collected to develop a launch vehicle database. Based on this database, we estimate the reliability of each launch vehicle family, using a modified two-level Bayesian analysis. The factors of both launch cost and reliability are subsequently merged into the expected launch cost, acting as the comprehensive metric to evaluate the cost of launch activities. This enables the optimization process of system design configurations to minimise the overall launch costs.

More specifically, the reliability of a launch vehicle is calculated by a modified two-level Bayesian analysis. The first level is acceptable for the launch vehicles with a large sample size of launch data, where the launch data dominate the estimation process. The second level is appropriate for the launch vehicles with small sample size data, where prior knowledge plays a significant role in the estimation process. The cost of individual dedicated launches for different families of launch vehicles is estimated by the average of the history launches.

5.3 Replenishment Cost Development

This section estimates the running costs of a space system in the operation phase, with subsystem development costs and reliability properties as inputs. Since on-orbit servicing is extremely costly and operationally complex, system maintenance can be executed by replacing the failed or degraded satellite by a new one. Due to the similar impacts on on-orbit space systems, other costs such as the cost of ground supporting are not considered in this approach.

The modelling of subsystem reliability in the operation phase and subsystem cost established in the development phase, are applied to calculate the expected replenishment costs of a space system. Based upon the modelling, the mapping relationship between different design configuration characteristics, namely, the degree of duplication, fractionation, and derivation, and replenishment costs are established. The applications of this mapping relationship are dual: on the one hand, the effects of different design configurations on replenishment costs can be analysed; on the other hand, by adopting replenishment costs as the objective function, the design configurations can be optimized in the system characteristic space.

5.3.1 Replenishment Cost Modelling

Before analysing the replenishment costs, the modelling of subsystem reliability and cost are built. This acts as the value model for the proposed value-centric design architecture, the key to evaluate the system value and enable the value flow in the design process. More specifically, the models are used to generate replenishment costs, namely, the costs of replacing the failed or degraded satellite by a new one in the operation phase. The replenishment costs can be used as an objective function for the optimization process of design configurations.

It is noteworthy that on-orbit servicing is not considered in this research, due to significant uncertainty in costs. However, this approach could also be further adapted for on-orbit servicing if the corresponding model and data were available.

(1) Subsystem Reliability Models

Mathematically, reliability is defined by Thompson [160] and Davidson [161] as the probability of a component, device or system performing its intended function without failure for a specific period of time under nominal conditions. In order to describe how the failure occurrences are distributed over time, different types of probability distributions are generally applied, such as the exponential, Weibull, lognormal, extreme-value distributions. The modelling of subsystem reliability is time-related, and it differs from the previous modelling of launch reliability that is time-independent.

Due to its mathematical simplicity, flexibility, and applicability to the modelling of different failure behaviours, and the demonstrated ability to fit most lifetime data, the Weibull distribution [161] is one of the most widely-used distributions in reliability analysis. The Probability Density Function (PDF) of the Weibull distribution with a continuous random variable can be written by Eq. (5.15).

$$r(t) = \begin{cases} \left(\frac{\beta}{\eta}\right)\left(\frac{t}{\eta}\right)^{(\beta-1)} e^{-\left(\frac{t}{\eta}\right)^{\beta}} & t \geq 0 \\ 0 & t < 0 \end{cases} \tag{5.15}$$

Where, t is the satellite lifetime in orbit before failure, β is the dimensionless shape parameter, η is the scale parameter with units of time. Consequently, the reliability Cumulative Distribution Function (CDF) can be expressed by Eq. (5.16).

$$R(t) = e^{-\left(\frac{t}{\eta}\right)^{\beta}} \tag{5.16}$$

This is the reliability model used in this research for the different subsystems of space systems. For each specific case, the parameters β and η are calculated by parameter estimation methods, among which Maximum Likelihood Estimation (MLE) and Bayesian theory combined with Markov Chain Monte Carlo simulations (MCMC) are two of the most effective and applicable ones. The selection of these two methods is primarily determined by the sample size of a database.

Based on 1584 Earth-orbiting satellites successfully launched from January 1990 to October 2008 recorded in SpaceTrak Database (STD), Castet and Saleh [162, 163] developed and fitted a parametric reliability model for actual satellite subsystems. Meanwhile, Guo [164] conducted reliability modelling particularly for small satellite subsystems, and found that MCMC is suitable for such databases with small sample sizes and censored data problems. The Small Satellite Anomalies Database (SSAD) is a database covering 222 small satellites launched over the last few decades. It is apparent that SSAD is more applicable for the reliability analysis of small satellite subsystems since all the data are collected from small satellites, while STD may appeal to large satellites. The estimated Weibull parameters for STD and SSAD are both listed in Table 5.12 and the corresponding lifetime reliability of different subsystems are exhibited in Fig. 5.9.

As Fig. 5.9 shows, there is a significant difference between the two databases. The reliability of STD with most data from large satellites is clearly higher than that of

Table 5.12: The estimated Weibull parameters per subsystem [162–164]

Subsystem	STD		SSAD	
	β	η (years)	β	η (days)
PLS	0.8874	7983	0.4162	275006
EPS	0.7460	7733	0.3110	58385
TTC	0.3939	400982	0.2588	176896
CDH	0.8874	7983	0.4162	275006
ACS	0.7182	3831	0.4144	85409
OCS	0.3375	6206945	-	-
TCS	0.3560	21308746	0.2655	15068200
SAS	0.4035	1965868	-	-

SSAD with all the data from small satellites, regardless of the type of subsystems. A possible explanation for the sharp drop of system reliability over time might be that small satellites are still in the infant and most are not designed for a long mission lifetime.

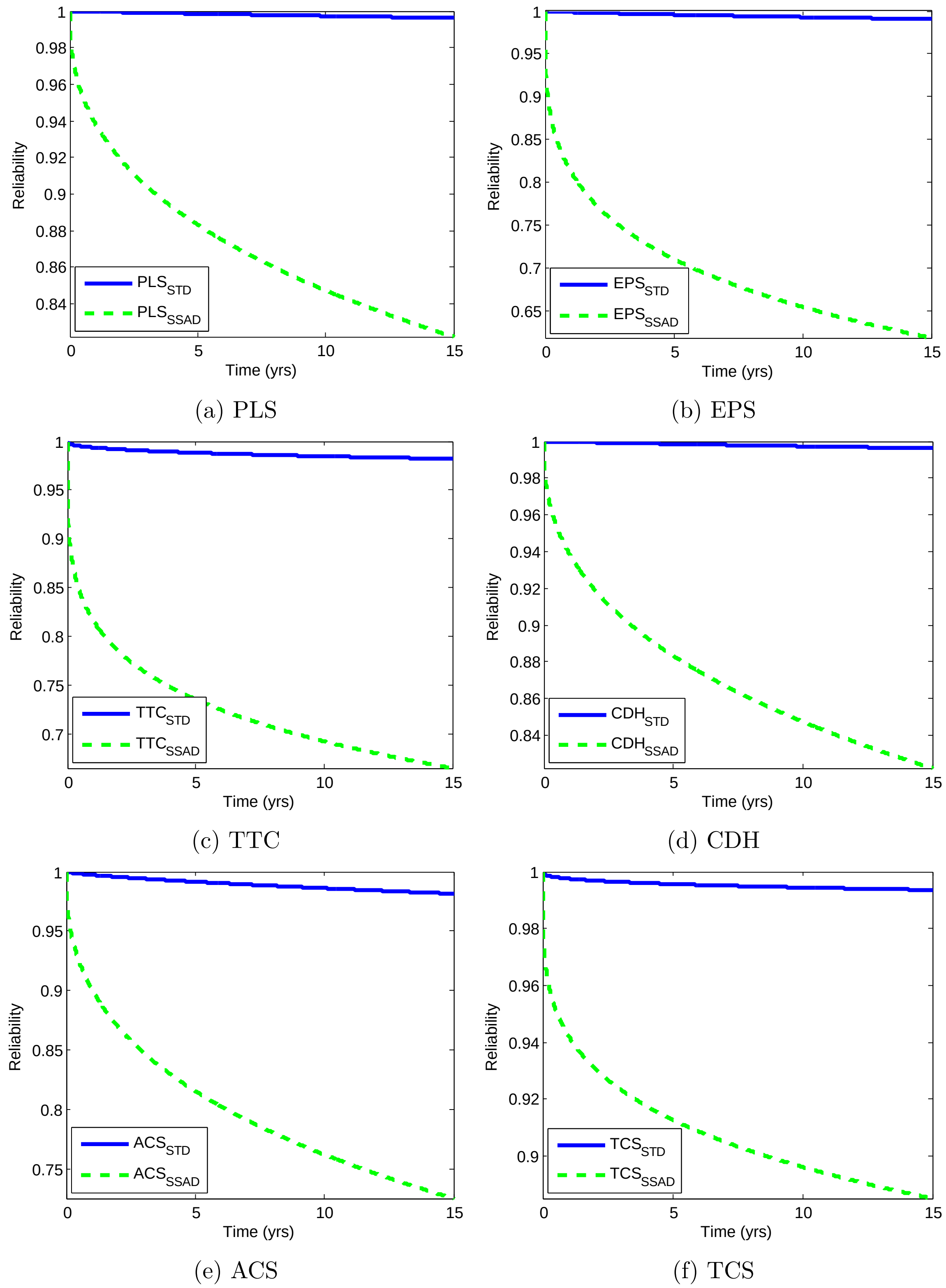
(2) Subsystem Cost Models

The replenishment costs are defined as the expense of replacing subsystems, satellites or systems that have failed or are near the end of life by new ones. More specifically, it focuses on the expected costs of such replacements, considering the lifetime reliability of a space system. The corresponding development costs of the replacement subsystems are calculated by the cost models established in the development phase, shown in Table 5.1, Table 5.2, Table 5.3, and Table 5.4. If multiple copies of any subsystem are manufactured, the learning curve factor expressed by Eq. (5.3) in Section 5.1.1 is considered. Furthermore, the launch costs of the replacement activities can also be evaluated, according to the launch cost models established in the launch phase.

5.3.2 Replenishment Cost Analysis

The effects of different design configuration characteristics, i.e., the degree of duplication, fractionation, and derivation on the replenishment costs are analysed in this section. This establishes the mapping relationship between design configurations and replenishment costs, which enables the optimization of the design configurations to minimise the system replenishment costs.

(1) Reliability Analysis on Duplication and Fractionation



(a) PLS (b) EPS

(c) TTC (d) CDH

(e) ACS (f) TCS

Fig. 5.9: Comparison of subsystem reliability for STD and SSAD

Prior to integrating the system reliability, preliminary effects analyses of the degree of duplication and fractionation on independent subsystems are carried out to help understand their impacts on overall system reliability. Firstly, the mathematical operation of duplication can be represented by Eq. (5.17).

$$R_{p,j}^{k}(t) = 1 - \Big(1 - R_j(t)\Big)^{k} \tag{5.17}$$

Where, $R_j(t)$ is the reliability CDF of individual subsystem j, k is the associated degree of duplication, and $R_{p,j}^{k}(t)$ is the reliability CDF of individual subsystem j at the degree of duplication of k. Since $0<R_j(t)< 1$, the reliability of duplicated systems monotonically increase with increasing degree of duplication, approaching 100% asymptotically. For comparison, this expression can be transformed into its recurrence relation, shown in Eq. (5.18).

$$R_{p,j}^{k+1}(t) = R_j(t) + \Big(1 - R_j(t)\Big)R_{p,j}^{k}(t) \tag{5.18}$$

As for fractionation, it can be viewed as another form of duplication, which is spatially separated. The potential exchange of information, power, and force between fractionated modules is assumed to be applied by WSS [42]. Therefore, the mathematical operation of fractionation can be expressed by Eq. (5.19).

$$R_{f,j}^{k+1}(t) = R_j(t)R_{w,j}(t) + \Big(1 - R_j(t)R_{w,j}(t)\Big)R_{f,j}^{k}(t) \tag{5.19}$$

Where, $R_{w,j}(t)$ is the reliability CDF of the WSS used for subsystem j, and $R_{f,j}^{k}(t)$ is the reliability CDF of individual subsystem j for the degree of fractionation of k. Mathematically, the only difference between these two operations is replacing $R_j(t)$ with $R_j(t)R_{w,j}(t)$, which leads to totally different behaviours in performance. When the reliability of the WSS reaches 100%, fractionated systems can theoretically act as duplicated systems.

Assume that the reliability of the WSS also obeys the Weibull distribution, and that its EOL reliability is 88.83%, which is in between all the subsystem reliability of STD and SSAD. This means that the WSS is considered to be more reliable than small satellite subsystems but less than large satellite ones. The simulations are carried out using the parameters in Table 5.12 to present the relationships between the reliability of independent subsystems, and the degree of duplication and fractionation, displayed in Fig. 5.10.

As shown in Fig. 5.10, duplicated subsystems enjoy a continuous reliability growth with increasing degree of duplication, while fractionated subsystems experience a trough and recovery period, yet still cannot perform as reliably as duplication at the same degree. The high reliability of duplication benefits from its parallel connections, while the extra wireless sharing factor limits the reliability of fractionation. Moreover, for the subsystems with reliability higher than that of wireless sharing, the fractionated reliability cannot recover to the individual subsystem reliability with increasing degree of fractionation.

In conclusion, the extra WSS, which is the major difference between duplication and fractionation, is also the main constraint of the upper bound of system reliability. On the one hand, if the reliability of the WSS could reach as high as possible, fractionation would become as reliable as duplication. On the other hand, the actual reliability of the WSS that is less than 100% decreases the reliability of the entire system.

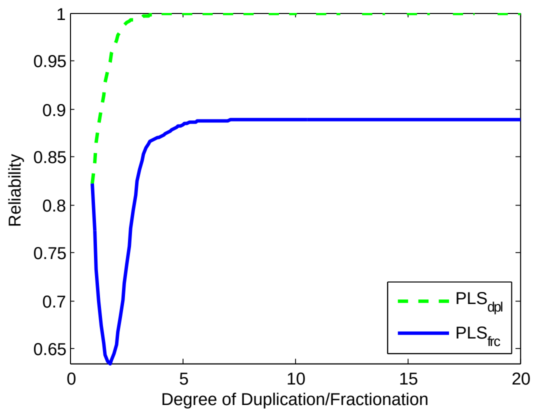
(2) Reliability Analysis on Derivation

Derivation exhibits a different impact on system reliability, by influencing each individual subsystem unit, instead of the entire system configuration. In other words, it assigns different values for each individual element, rather than their combinations.

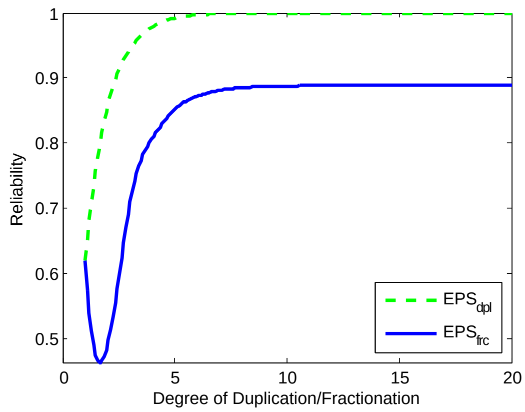
To display the impacts of derivation, the derivation simulations are conducted by altering the degree of derivation from 0.0 to 1.0 with the interval of 0.1, respectively representing by varying the colour from the lightest to the darkest in Fig. 5.11. Throughout the simulations for Fig. 5.11, the parameter and data of STD is used for mature design, and that of SSAD is applied for innovative design. This is because most data of STD are from large satellites, namely, mature design, while all the data of SSAD are from small satellites, namely, innovative design. Since the subsystem reliability CDFs are exponential or non-additive, the relationship between subsystem reliability and the degree of derivation can be mathematically expressed by Eq. (5.20).

$$R_{v,j}(t) = R_{\mathrm{STD},j}(t)^{v_j} \cdot R_{\mathrm{SSAD},j}(t)^{1-v_j} \tag{5.20}$$

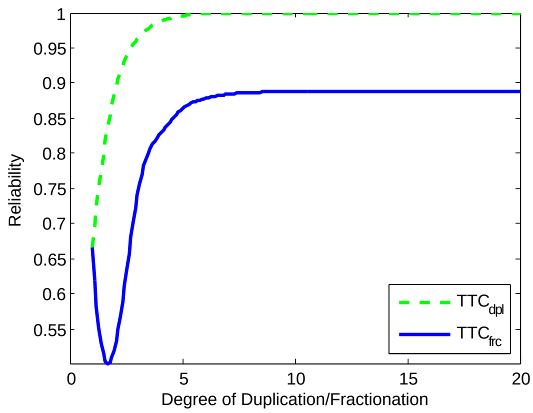
Where, $R_{v,j}(t)$ is the reliability of subsystem j at the degree of derivation of v_j, and $R_{\mathrm{STD},j}(t)$ and $R_{\mathrm{SSAD},j}(t)$ are the reliability CDFs of subsystem j in STD and SSAD respectively.



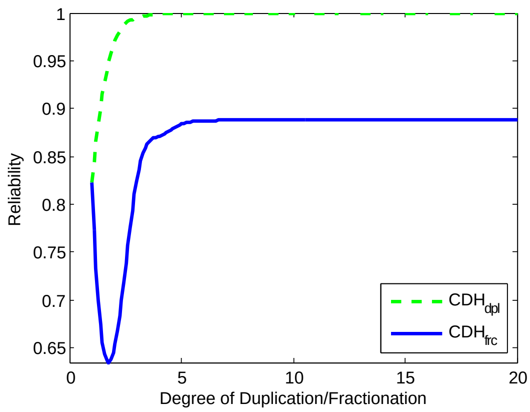
(a) PLS



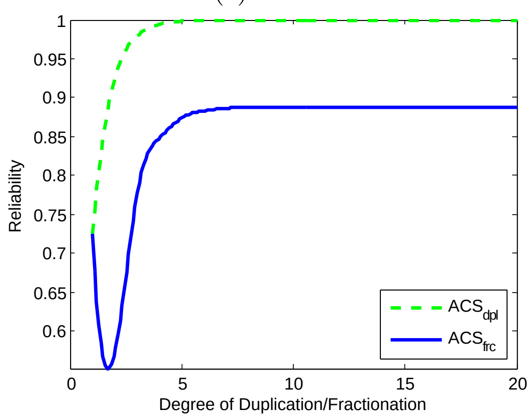
(b) EPS



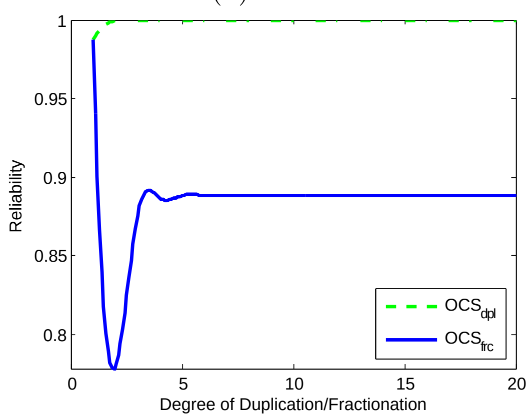
(c) TTC



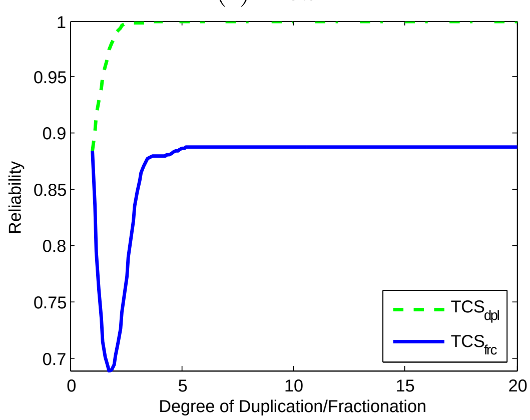
(d) CDH



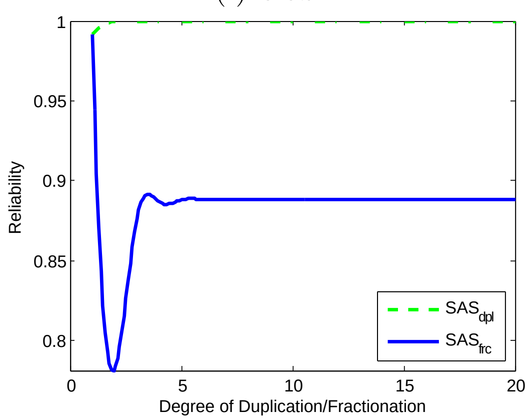
(e) ACS



(f) OCS

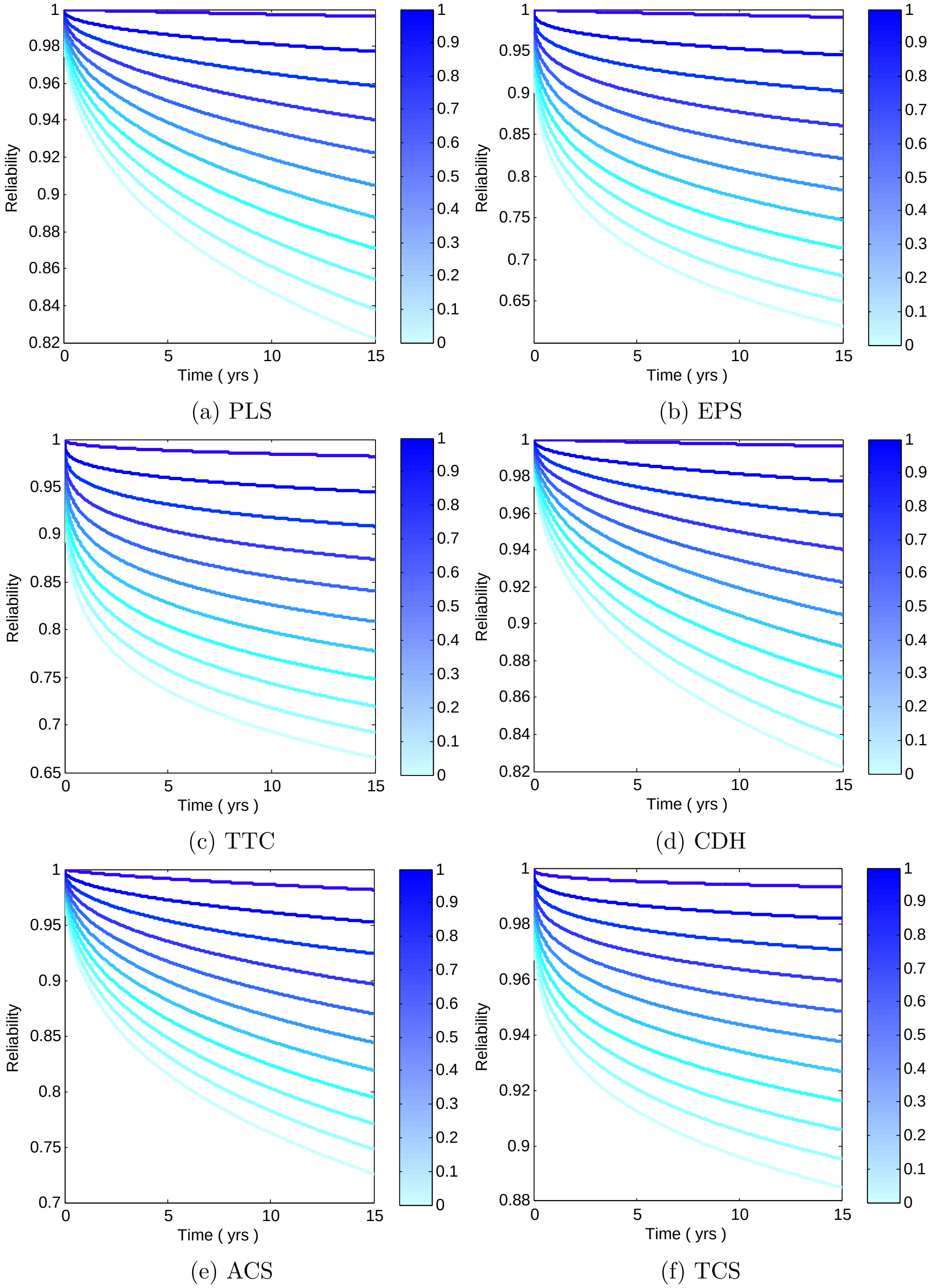


(g) TCS



(h) SAS

Fig. 5.10: Duplication and fractionation characteristics per subsystem [20]



(a) PLS (b) EPS (c) TTC (d) CDH (e) ACS (f) TCS

Fig. 5.11: Derivation characteristic per subsystem (The colourbar indicates the degree of derivation) [20]

Taking the natural logarithm of both sides, we obtain Eq. (5.21).

$$\ln R_{v,j}(t) = v_j \ln R_{\mathrm{STD},j}(t) + (1 - v_j) \ln R_{\mathrm{SSAD},j}(t) \tag{5.21}$$

This is the equation used for derivation analyses, whose variable $\ln R_{v,j}(t)$ is an additive property.

As can be seen in Fig. 5.11, the high degree of derivation of mature designs ensures system reliability, while innovative design inevitably brings risks for a system. It is interesting to note that the reliability increment between the two adjacent curves increases with the increasing degree of derivation. In summary, derivation provides another insight for system reliability behaviours. On the one hand, the more derivative to mature designs, the more reliable a system is. On the other hand, adopting novel designs might bring in capability advancement, performance improvement, or cost reduction.

(3) Replenishment Cost Analysis on Fractionation

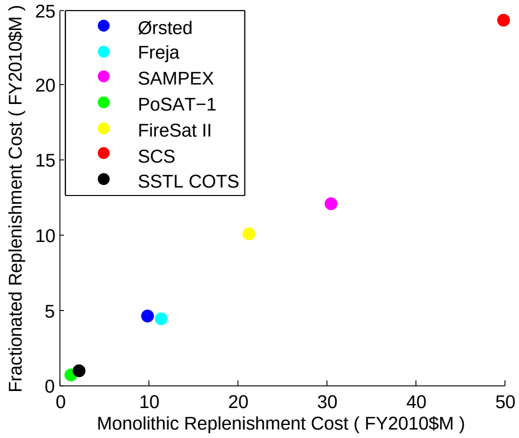
Based on the cost and reliability models established, case studies are carried out to compare the replenishment costs between two extreme design configurations: monolithic and fractionated.

In accordance with the effects analysis in the development phase, the lowest cost of subsystems is adopted as the cost of the WSS for the case studies, and the input parameters are same with the ones in Table 5.5. Moreover, the parameters of SSAD are applied for the EOL reliability, as shown in Table 5.13.

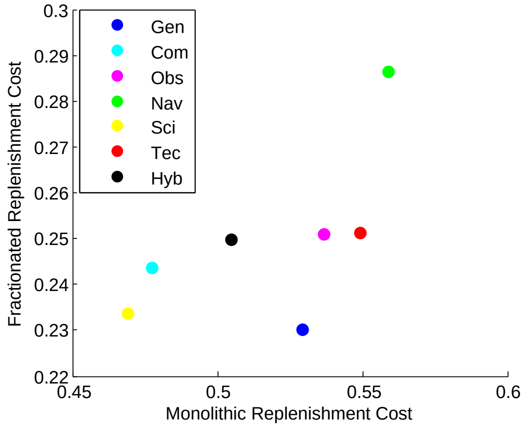
Table 5.13: Subsystem EOL reliability from SSAD [20, 164]

Subsystem	EOL Reliability (%)
PLS	82.21
EPS	61.94
TTC	66.58
CDH	82.21
ACS	72.59
OCS	98.74
TCS	88.50
SAS	99.14

The simulation results shown in Fig. 5.12 indicate that fractionated configurations cost approximately half of their monolithic counterparts throughout the lifetime maintenance for both spacecraft cost fraction models and real satellite cases. In other words, fractionation is likely to cut down the replenishment costs by almost 50% for the investigated small satellites.



(a) Comparison by Cases



(b) Comparison by Missions

Fig. 5.12: Comparison of replenishment cost for different system configurations (The maintenance costs in Fig. 5.12(b) are normalized into the fraction of the total space mission cost.)

Having analysed the replenishment costs of typical cases, it might be useful to theoretically derive the relationship of the replenishment costs between monolithic and fractionated configurations. Assume that R and C are the System Reliability and Cost Vectors respectively, as given by Eq. (5.22) and Eq. (5.23).

$$R = \begin{bmatrix} r_1 & r_2 & \dots & r_n \end{bmatrix} \tag{5.22}$$

$$C = \begin{bmatrix} c_1 & c_2 & \dots & c_n \end{bmatrix}^{\mathsf{T}} \tag{5.23}$$

Where, r_j and c_j are the EOL reliability and the development cost of subsystem j. In this condition, monolithic $c_{r,m}$ and fractionated $c_{r,f}$ replenishment costs can be presented by Eq. (5.24) and Eq. (5.25) respectively.

$$c_{r,m} = (1 - \prod_{j=1}^{n} r_j) \cdot \sum_{j=1}^{n} c_j \tag{5.24}$$

$$c_{r,f} = \sum_{j=1}^{n} (1 - r_j \cdot r_w)(c_j + c_w) \tag{5.25}$$

Where, r_w and c_w are the reliability and the cost of the WSS. Thus, the difference between monolithic and fractionated replenishment cost is expressed by Eq. (5.26).

$$c_{r,m} - c_{r,f} = r_w \cdot \sum_{j=1}^{n} r_j c_j - \prod_{j=1}^{n} r_j \cdot \sum_{j=1}^{n} c_j + c_w \cdot \sum_{j=1}^{n} (r_j r_w - 1) \tag{5.26}$$

It can also be transferred into matrix form as shown by Eq. (5.27).

$$c_{r,m} - c_{r,f} = \begin{bmatrix} r_1 & r_2 & \dots & r_n & r_w \end{bmatrix} \begin{bmatrix} r_w - \frac{r_\Pi}{r_1} & & & & \\ & r_w - \frac{r_\Pi}{r_2} & & & \\ & & \ddots & & \\ & & & r_w - \frac{r_\Pi}{r_n} & \\ & & & & r_\Sigma - \frac{n}{r_w} \end{bmatrix} \begin{bmatrix} c_1 \\ c_2 \\ \vdots \\ c_n \\ c_w \end{bmatrix} \tag{5.27}$$

Where, $r_\Pi = \prod_{j=1}^{n} r_j$ and $r_\Sigma = \sum_{j=1}^{n} r_j$. Assuming that $c_k = \max_{j \epsilon n} c_j$, the above matrix expression can be further transformed into Eq. (5.28).

$$c_{r,m} - c_{r,f} = \begin{bmatrix} r_1 & \dots & r_k & \dots & r_n \end{bmatrix} \begin{bmatrix} r_w - \frac{r_\Pi}{r_1} & & & & \\ & \ddots & & & \\ & & r_w - \frac{r_\Pi}{r_k} + \frac{c_w}{c_k}\frac{r_\Sigma r_w - n}{r_k} & & \\ & & & \ddots & \\ & & & & r_w - \frac{r_\Pi}{r_n} \end{bmatrix} \begin{bmatrix} c_1 \\ \vdots \\ c_k \\ \vdots \\ c_n \end{bmatrix} \tag{5.28}$$

Or, it can be written as shown by Eq. (5.29).

$$c_{r,m} - c_{r,f} = R_{1\times n} H_{n\times n} C_{n\times 1} \tag{5.29}$$

Where,

$$H = \begin{bmatrix} r_w - \frac{r_\Pi}{r_1} & & & & \\ & \ddots & & & \\ & & r_w - \frac{r_\Pi}{r_k} + \frac{c_w}{c_k}\frac{r_\Sigma r_w - n}{r_k} & & \\ & & & \ddots & \\ & & & & r_w - \frac{r_\Pi}{r_n} \end{bmatrix} \tag{5.30}$$

Since it is obvious that $R \geq 0$ and $C \geq 0$, the expression is positive semidefinite, when every eigenvalue of H is non-negative.

$$\begin{cases} r_w \geq \max_{j \epsilon n} \frac{r_\Pi}{r_j} \\ r_w \geq \frac{r_\Pi}{r_k} - \frac{c_w}{c_k}\frac{r_\Sigma r_w - n}{r_k} \end{cases} \tag{5.31}$$

Adopting the parameters of SSAD for the EOL reliability as shown in Table 5.13, the remainder of this section will demonstrate the positive semi-definiteness of Eq. (5.31), namely, the cost-efficiency of fractionated configurations.

For the small satellite subsystems in Table 5.13, $r_w \geq \max_{j\epsilon n} \frac{r_\Pi}{r_j}$, when $r_w \geq$ 28.30%, this condition can easily be reached for wireless sharing devices as one of core technologies of fractionation. Therefore, H is a positive semidefinite matrix, equivalent to Eq. (5.32).

$$\frac{c_w}{c_k} \leq \frac{r_k}{r_\Sigma r_w - n}\left(\frac{r_\Pi}{r_k} - r_w\right) \tag{5.32}$$

Assuming k as the PLS (since the payload is generally considered as the most costly subsystem in a space system, seen in Table 5.3) and r_w is only 50%, we can obtain Eq. (5.33).

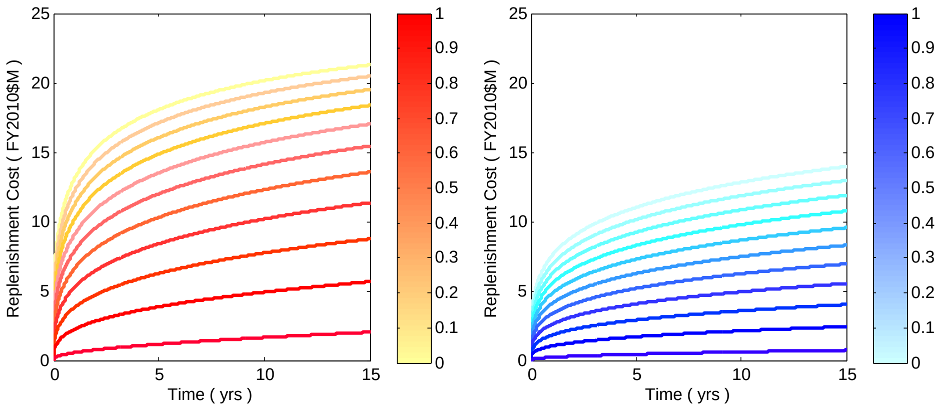
$$\frac{c_w}{c_k} \leq 0.05 \tag{5.33}$$

Namely, for small satellites, the cost of a single WSS has to be no more than 5% of the cost of the PLS, to ensure fractionated systems have a lower cost than monolithic systems during the lifecycle maintenance. In terms of general small satellite missions, seen in the Table 5.3, that is almost as much as the cost of the TCS, which does not seem to be a very challenging or strict requirement for close range sharing subsystems. In conclusion, the maintenance cost of fractionated systems could be reasonably believed to be more cost-efficient than that of monolithic systems, when using small satellite COTS products.

(4) Replenishment Cost Analysis on Derivation

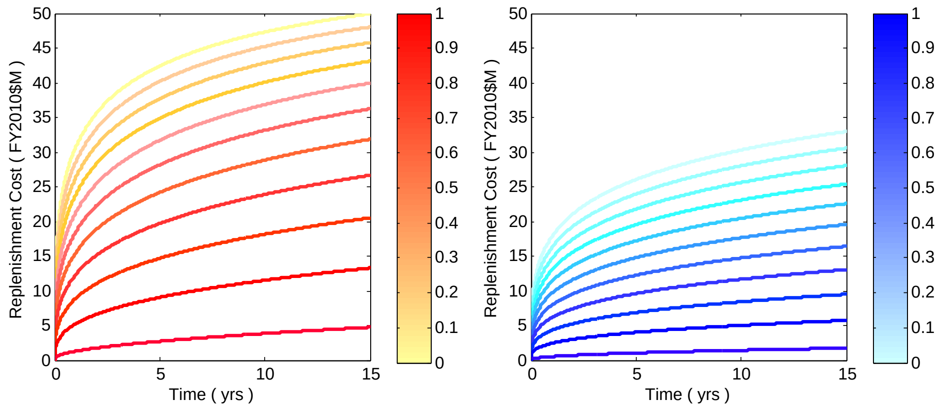
The effects of derivation on maintenance costs are twofold: on one hand, derivative modules tend to cost less in the fabrication process due to the learning curve factor; on the other hand, innovative modules tend to be less reliable, regardless of their advancement and competitiveness. Both effects influence the value of individual modules, instead of the integration process of value. Simulations are executed by mathematically integrating the cost and the reliability of derivative and non-derivative components based upon the percentage of mass. The subsystem data of FireSat II and SCS satellites are utilised for monolithic and fractionated configurations. As shown in Fig. 5.13 and Fig. 5.14, the colour variation from the darkest to the lightest represents the variation of the degree of derivation from the highest to the lowest.

Throughout the lifetime, maintenance costs for a system can be reduced by increasing the degree of derivation. The influence of the degree of derivation on maintenance cost reduction is much greater for monolithic systems than fractionated systems. The conclusions are dual: on one hand, increasing the degree of derivation, or applying



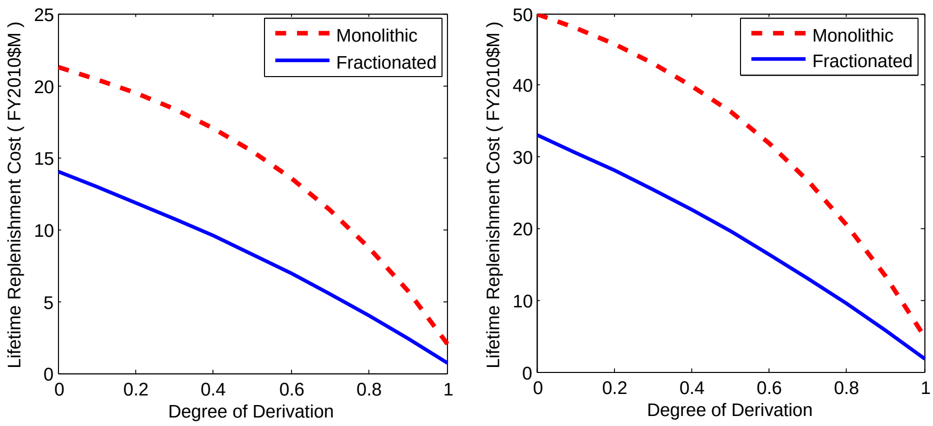
(a) Monolithic Systems (b) Fractionated Systems

Fig. 5.13: Effects of derivation on lifetime replenishment cost for FireSat II (The colour variation from the darkest to the lightest represents the variation of the degree of derivation from the highest to the lowest.)



(a) Monolithic Systems (b) Fractionated Systems

Fig. 5.14: Effects of derivation on lifetime replenishment cost for SCS (The colour variation from the darkest to the lightest represents the variation of the degree of derivation from the highest to the lowest.)



(a) FireSat II (b) SCS

Fig. 5.15: Lifetime replenishment cost of different degree of derivation

mature technologies, can effectively control the running cost of monolithic systems; on the other hand, fractionated configurations can be a good paradigm for testing or initially applying of latest concepts and technologies. More intuitive relationship between lifetime maintenance costs and the degree of derivation can be found in Fig. 5.15.

5.3.3 Replenishment Cost Integration

Based upon the analysis of the effects of the degree of duplication, fractionation, and derivation on the replenishment costs, the integration of system replenishment costs in the system characteristic space is implemented to establish the mapping relationships between the internal configuration designs and the external replenishment costs, generating an appropriate value for the value flow process in the operation phase.

In accordance with the definition, replenishment costs can be considered as an integrated metric covering both system reliability and cost characteristics. Therefore, the key to this process is the establishment of integrated value models, that is, the objective function that outputs the on-orbit replenishment costs with the inputs of subsystem cost and reliability models, and an integration mechanism. Specific calculation procedures are illustrated in Fig. 5.16, with the system characteristic space consisting of duplication, fractionation, derivation, and sequence as variables.

1) The system design configuration is defined by the degree of duplication, fractionation, and derivation, which can be equivalently transformed into a more intuitive expression, that is, the system configuration matrix.

2) The subsystem cost models established in Section 5.1.1 and wireless sharing reliability model are applied to calculate the reliability of all the subsystems of each satellite in a space system.

3) The subsystem cost models described in Section 5.1.1 and wireless sharing cost penalty are applied to calculate the development costs of all the subsystem of each satellite in a space system.

4) The development costs and reliability of all the subsystems are then integrated to generate the replenishment costs of each satellite, through the system configuration matrix. Finally, the overall replenishment costs are the sum of those of all the satellites in this system.

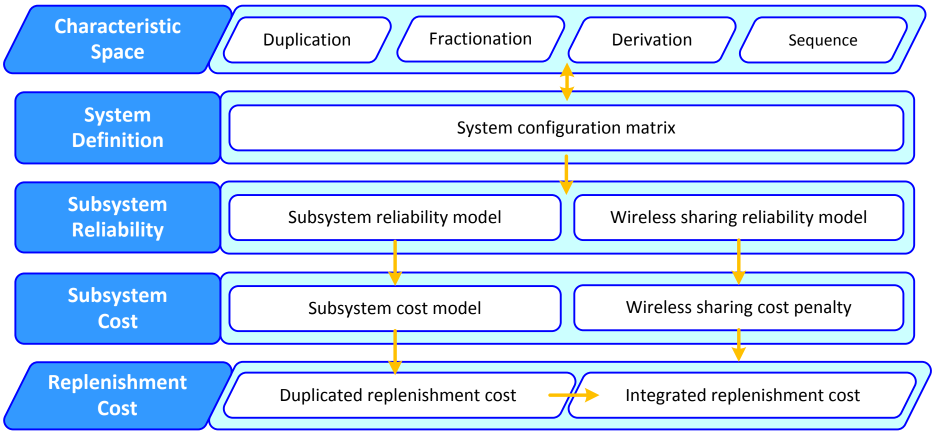


Fig. 5.16: System replenishment cost integration process

Table 5.14: Inputs of system replenishment cost integration

Parameter	Symbols	Values
Number of satellites	m	6
Number of subsystems	n	5
System Duplication Vector	P	$\begin{bmatrix} 2 & 4 & 2 & 1 & 3 \end{bmatrix}^T$
System Fractionation Vector	F	$\begin{bmatrix} 2 & 2 & 2 & 1 & 2 \end{bmatrix}^T$
System Derivation Matrix 1	V_1	$\begin{bmatrix} 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \end{bmatrix}$
System Derivation Matrix 2	V_2	$\begin{bmatrix} 0.1 & 0.1 & 0.1 & 0.1 & 0.1 \\ 0.2 & 0.2 & 0.2 & 0.2 & 0.2 \\ 0.3 & 0.3 & 0.3 & 0.3 & 0.3 \\ 0.4 & 0.4 & 0.4 & 0.4 & 0.4 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.6 & 0.6 & 0.6 & 0.6 & 0.6 \end{bmatrix}$
System Sequence Vector	Q	$\begin{bmatrix} 2 & 20 & 10 & 5 & 30 \end{bmatrix}^T$

Table 5.15: Outputs of system replenishment cost integration (All the results are normalised to the fraction of spacecraft costs.)

Input Model	Replenishment Cost (%)	Input Model	Replenishment Cost (%)
Ørsted	6.57	Gen	6.18
Freja	8.20	Com	5.42
SAMPEX	7.47	Obs	5.56
PoSAT-1	8.40	Nav	7.06
FireSat II	6.85	Sci	7.88
SCS	5.86	Tec	8.07
SSTL COTS	7.94	Hyb	6.56

The input values are shown in Table 5.14, same as the simulations in the development phase, and the corresponding results are listed in Table 5.15, according to different subsystem cost models. It is noteworthy that two different System Derivation Matrices are given in Table 5.14, and they are used as the inputs for different case studies in Section 6.1.3. To be specific, V_1 represents the identical satellites in a system, while V_2 represents the satellites with the increasing degree of derivation in a system.

Although the expectations of lifetime replenishment costs are controlled to some extent, system designers might further wonder whether there is an overall best configuration with all the available subsystem resources suitably arranged to reach the minimum replenishment costs when the degree of duplication, fractionation, and derivation is fixed. Same with Section 5.1.3, the System Sequence Matrix Q is alterable during the simulations, the remainder of this chapter will focus on optimizing Q under the conditions of the fixed P, F, and V, namely, making full use of the available subsystems.

5.3.4 Replenishment Cost Optimization

Having derived the mathematical relationships between system configuration designs and lifetime replenishment costs, this section completes the final step for the proposed value-centric design loop, namely, the **optimize** process, encouraging the system to be designed through a global optimization process rather than a complex and discrete requirement allocation process.

The proposed configuration optimization problem regarding system replenishment costs can also be classified into MINLP problems, given by Eq. (5.4). The corresponding optimization process is shown in Fig. 5.17. During this optimization process, the value used as the objective function $J(x)$ is replenishment costs. The state variable x, namely, the System Sequence Vector Q, is used to define the combinations and permutations of a space system considering duplication and/or fractionation.

As the design space of the system configurations, the system characteristic space can be used to enable this process. The proposed space transforms the input parameters of duplication, fractionation, derivation, and sequence into the System Configuration Matrix, which provides the complete definition of the entire system. Through

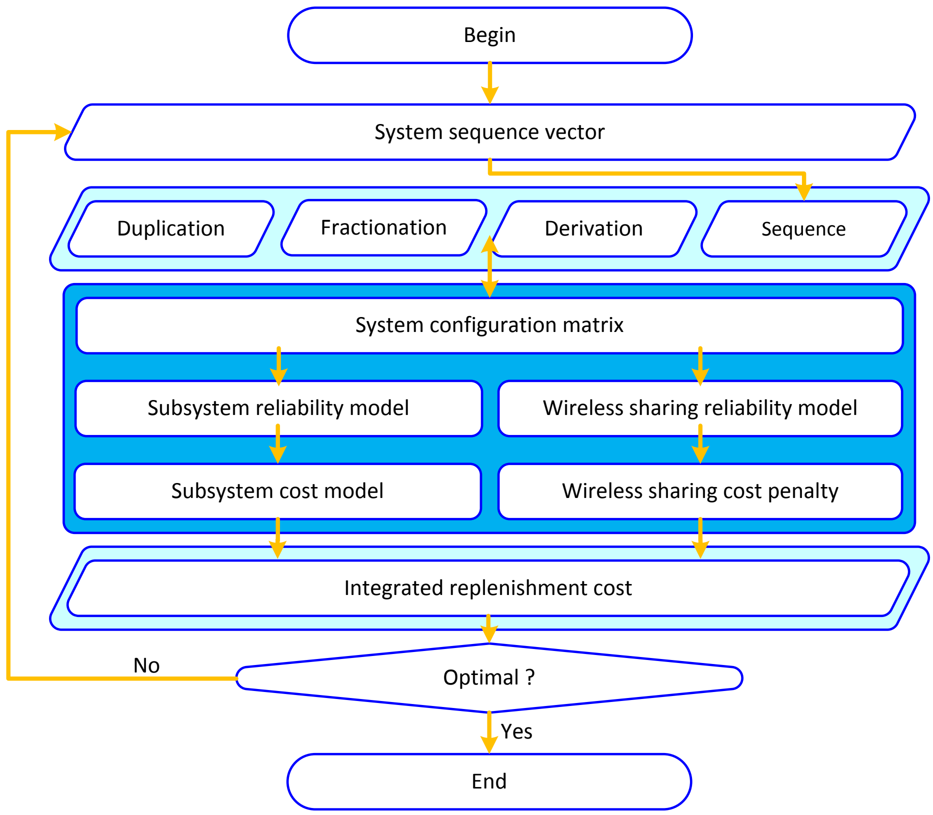


Fig. 5.17: System replenishment cost optimization process

this matrix, the reliability and cost properties can be integrated into the replenishment costs, from the subsystem level to the system level. Based on this integration process, the optimization algorithms can therefore be applied to find the optimal set of system designs with the minimum replenishment costs.

5.3.5 Summary

This section models, analyses, and optimizes the costs in the operation phase of a space system. Under the value-centric design architecture, the exploration of design configurations is realised in the system characteristic space, with replenishment cost as the objective function. Based upon the subsystem reliability and development cost models, the effects analysis is implemented on different configuration characteristics. The value integration process was conducted to establish the mapping relationships between system design configurations and replenishment costs. This enables the optimization of design configurations to achieve the minimum replenishment costs in the operation phase.

5.4 Disposal Cost Development

Disposal costs assess the costs to implement the retirement of a space system. There are two circumstances: if a space system is equipped with EOL de-orbit or orbital lifetime reduction devices, the disposal costs can be attributed to the development costs of these devices, which has been discussed in the development phase; if not, the active debris removal (ADR) activities are applied, and the corresponding costs are considered as the disposal costs in the retirement phase. Furthermore, the assessment of the costs in the event of the failure of EOL de-orbit or orbital lifetime reduction devices is considered as the latter.

The disposal costs are estimated by calculating the costs of building and launching the spacecraft to send the de-orbit packages to implement disposal activities. To minimise the costs, the change in velocity is optimized for the orbital transfer between any two targets, while the optimal grouping strategy is developed for missions of multiple targets.

5.4.1 Disposal Cost Modelling

To be specific, disposal costs refer to the costs of a series of missions designed to remove all the critical debris after a space system is retired. Each mission is a multiple-rendezvous trajectory where a subset of the total target debris is removed by a spacecraft delivering and activating the de-orbit packages. Thus, the cost function can be presented by Eq. (5.34).

$$J_p = \sum_{i=1}^{m} C_{p,i} (i = 1, 2, \cdots, m) \tag{5.34}$$

Where, J_p is the objective function, namely, the total costs for m missions, and $C_{p,i}$ is the cost for the i^{th} mission. Each mission cost $C_{p,i}$ consists of two major aspects: the spacecraft launch cost $C_{l,i}$ and the on-orbit removal cost $C_{r,i}$, given by Eq. (5.35).

$$C_{p,i} = C_{l,i} + C_{r,i} \tag{5.35}$$

The on-orbit removal cost refers to the cost for developing and delivering the de-orbit packages to remove the on-orbit target debris. Mathematically, it can be further broken down into two parts, given by Eq. (5.36), the cost $C_{rd,i}$ of developing the de-orbit packages and the cost $C_{rs,i}$ of developing a spacecraft to deliver them to the target debris or inactive satellites by multiple orbital manoeuvres.

$$C_{r,i} = C_{rd,i} + C_{rs,i} \tag{5.36}$$

Thus, the removal cost of each mission can be written as Eq. (5.37). All these costs will be discussed in detail in the following sections of this chapter.

$$C_{p,i} = C_{l,i} + C_{rd,i} + C_{rs,i} \tag{5.37}$$

(1) Spacecraft Launch Cost C_l

The spacecraft launch costs refer to the costs charged by launcher suppliers for launching the entire spacecraft into the required orbit. The expected launch costs have been modelled in the previous launch phase as a measurement of the costs of launching a spacecraft. Estimate of the total launch costs involves the integration of both the reliability and the cost information of a launch vehicle to define the expectation of the launch costs under risk, given by Eq. (5.11) and Eq. (5.13) in Section 5.2.1.

(2) Spacecraft Development Cost C_{rs}

The development costs of a spacecraft C_{rs} can be distributed into the development costs of each subsystem, which are estimated by the subsystem cost models established as shown in Table 5.1 and Table 5.2.

To achieve the total system costs, it is essential to input the mass of each subsystem into the cost models. The subsystem masses required can be calculated by the overall system dry mass, associated with the mass distribution model. The dry mass of the total spacecraft can be calculated according to the change in velocity ΔV, which can be estimated for the orbital manoeuvre between any two target debris. To simplify this calculation process, the following assumptions can be made:

1) The two-body model is adopted for estimating the change in velocity of orbital manoeuvres, without any orbital perturbation, e.g., gravity anomaly, atmosphere drag, third body interaction, etc.

2) The instantaneous manoeuvres, instead of continuous low thrust manoeuvres, are applied to the spacecraft velocity.

Based upon the above assumptions, the dry mass of the total spacecraft can be derived through the Tsiolkovsky equation, given by Eq. (5.38).

$$m_f = m_i \exp(-\frac{\Delta V}{I_{sp} g_0}) \tag{5.38}$$

Where, m_i is the initial mass or launch mass of a spacecraft, m_f is the final mass or dry mass of a spacecraft, I_{sp} is the specific impulse, and g_0 is the gravitational constant at the Earth's surface.

A mass distribution model particular for debris removal spacecraft [121] is adopted. This breaks down the total mass of the spacecraft particularly for debris removal missions into subsystems, listed in Table 5.16.

Table 5.16: Subsystem mass fraction of debris removal spacecraft [121]

Subsystem	Mass Distribution
EPS	0.25
TTC	0.08
ACS	0.12
OCS	0.10
TCS	0.05
STR	0.40

So far, the modelling of the spacecraft development cost $C_{rs,i}$ has been completed. In summary, the corresponding calculation procedures are shown in Table 5.17, with each step illustrated as follows.

Table 5.17: Calculation procedures of ADR spacecraft development cost

Procedures	Key Parameters
Input	Change in velocity ΔV
Step 1	Propellant mass m_{fuel}
Step 2	Spacecraft dry mass m_{dry}
Step 3	Subsystem mass m_j
Step 4	Subsystem cost c_j
Output	Spacecraft cost c_{rs}

1) Given the velocity change ΔV, the corresponding propellant mass m_{fuel} required to generate ΔV can be calculated by Eq. (5.38).

2) The spacecraft dry mass m_{dry} can be derived from the obtained propellant mass m_{fuel} through the calculation procedures shown in Table 5.18.

Where, m_{tank} is the mass of spacecraft tank, m_{kit} is the mass of single de-orbit kit that carries out the de-orbit function, and m_{plat} is the spacecraft dry mass excluding tank and kits.

3) Adopting the subsystem mass fraction relationships shown in Table 5.16, the mass of each subsystem of the mission spacecraft can be derived based on the spacecraft

Table 5.18: Calculation procedures of ADR spacecraft dry mass [121]

Parameters	Equations	
m_{fuel}	$m_{\text{fuel}} = m_i(1 - \exp(-\frac{\Delta V}{I_{sp} g_0}))$	
m_{tank}	$m_{\text{tank}} = 0.1 m_{\text{fuel}}$	
m_{kit}	Depending on target debris mass	
	80kg	Micro platform
m_{plat}	250kg	Small platform
	2000kg	Large platform
m_{bus}	$m_{\text{bus}} = m_{\text{plat}} + m_{\text{tank}}$	
m_{dry}	$m_{\text{dry}} = m_{\text{bus}} + n_{\text{kit}} \cdot m_{\text{kit}}$	

dry mass m_{dry}.

4) The mass of each subsystem is used as the input to calculate the development costs of each subsystem, using the USCM or SSCM CERs established in Section 5.1.1.

Since the mission spacecraft have different sizes for various target debris, the corresponding CERs are also different. During the cost estimation, the micro and small platforms apply the SSCM CERs, while the large platform applies the USCM CERs.

(3) De-orbit Package Development Cost C_{rd}

From the functional perspective, the de-orbit packages can be viewed as an individual propulsion system. Under this assumption, the corresponding costs can be roughly estimated using the USCM or SSCM CERs established in Section 5.1.1. Therefore, the calculation procedures of the de-orbit costs can be summarised as follows:

1) Calculate the change in velocity, according to the orbit parameters of the target debris.

2) Derive the mass of the de-orbit package or kit using Eq. (5.38), with the input parameters of the required change in velocity and the mass of the debris.

3) Estimate the development costs of the de-orbit package or kit using USCM or SSCM CERs, with the input of its mass.

The process by which the spacecraft removes the debris is also one of the major parameters to estimate the mission costs. There are four types of processes designed for debris removal missions, as defined by Yamamoto [121].

1) **Single**. In the **single** process, each mission spacecraft captures and removes one debris. In this case, each mission spacecraft launches only one de-orbit package or kit, while the entire mission requires a series of spacecraft. The de-orbit costs of each debris can be estimated using the above calculation procedures of the de-orbit costs,

with the mass of that debris. The total de-orbit costs are the sum of the de-orbit costs of each debris.

2) **Multiple**. In the **multiple** process, each mission spacecraft captures multiple debris one by one, and removes them after gathering them all together. Similar to the **single** process, each spacecraft launches only one de-orbit package or kit. Therefore, the total de-orbit costs can be estimated through similar procedures, but with the input of the sum of the mass of all the debris.

3) **Mothership**. In the **mothership** process, each mission spacecraft equips with multiple de-orbits packages to remove multiple debris one by one. In this case, each debris is removed by launching a separate de-orbit package or kit, after the mission spacecraft carries them to the target orbit. The total costs can be estimated by the sum of the costs of all the de-orbit packages or kits. Although the total de-orbit costs of the **mothership** process is same with that of the **single** process, the spacecraft development costs of these two processes are completely different.

4) **Shuttle**. In the **shuttle** process, each mission spacecraft captures a debris, removes it, then ascents to the original altitude, and heads to the next target debris. It differs from the **multiple** process in that the spacecraft removes the debris immediately after reaching it, instead of storing them until the last target to carry out the de-orbit activities all together. Moreover, in this process, de-orbit activities are performed by spacecraft itself without de-orbit packages or kits. The de-orbit costs of this process can be considered as zero, while the corresponding spacecraft development costs may increase sharply.

In conclusion, the calculation procedures of the de-orbit costs can be summarised as follows:

1) Identify the type of the process by which the spacecraft removes the debris.

2) Calculate the change in velocity of removing each target debris.

3) Estimate the corresponding costs using the calculation procedures of the de-orbit costs, with the input parameters of the required change in velocity and the debris mass. There are three different cases: for the **shuttle** process, the total de-orbit costs are zero; for the **multiple** process, the spacecraft conducts the de-orbit activity after capturing all the target debris, so that the total de-orbit costs are estimated by the sum of the mass of all the debris; for the **single** and **mothership** processes, each

debris is removed immediately, so that the total de-orbit costs are the sum of the de-orbit costs of each debris. It is also noteworthy that the spacecraft development costs for the three cases are completely different.

(4) Change in Velocity

Throughout the calculation procedures of the disposal costs, the change in velocity is the necessary parameter as one of the inputs. In this section, the calculation of the velocity change is provided.

Generally, the change in velocity between any two target debris can be considered as a two-point boundary value problem, which can be calculated by the Lambert solutions [165]. In order to seek the optimal two-impulse solution with the minimum velocity change, the transfer time is optimized by applying the optimization algorithms.

To simplify the calculation process and save the calculation time, the Hohmann transfer can be used to estimate the two-impulse manoeuvre minimum velocity change between two coplanar circular orbits [165]. Given the radius of the initial orbit r_i and the final orbit r_f, the change in velocity of the Hohmann transfer trajectory is presented by Eq. (5.39).

$$\Delta V = \left|\sqrt{\frac{2\mu}{r_i} - \frac{\mu}{a_t}} - \sqrt{\frac{\mu}{r_i}}\right| + \left|\sqrt{\frac{\mu}{r_f}} - \sqrt{\frac{2\mu}{r_f} - \frac{\mu}{a_t}}\right| \tag{5.39}$$

Where, the semi-major axis of the transfer orbit can be calculated by the two mission orbits, given by Eq. (5.40).

$$a_t = \frac{r_i + r_f}{2} \tag{5.40}$$

Furthermore, the Hohmann transfer can be applied into two coaxially aligned elliptical orbits, while the transfer must start at either apogee or perigee to be the lowest energy.

5.4.2 Combination and Permutation

For the removal missions of multiple debris, multiple spacecraft each targeting a set of the debris may be required. In this condition, the combination and permutation of all the debris are two key research problems. Combination refers to the grouping of multiple debris, and each group of debris are removed by a separate spacecraft. Each spacecraft trajectory is defined as a removal mission, while the missions are independent from each other. Permutation refers to the removal sequence of the debris within a group, which determines a mission trajectory. In this section, a specific method for the combination and permutation of multiple debris is developed to explore the optimal set of mission designs to minimise the total costs. The calculation procedures can be illustrated as follows:

1) Calculate the total number of all the possible combinations and permutations.

To calculate the total number of all the possible combinations and permutations of the target debris, Stirling numbers of the second kind [166] are introduced. These are the number of ways to partition a set of n labelled objects, namely, the target debris, into k non-empty unlabelled subsets, namely, a series of removal missions, denoted by $S(n,k)$ or $\left\{ {n \atop k} \right\}$. Therefore, given the number of groups, k, the total number of all the possible combinations and permutations of n debris, can be calculated using Eq. (5.41).

$$S(n,k) = \frac{1}{k!}\sum_{j=0}^{k}(-1)^{k-j}\binom{k}{j}j^n \tag{5.41}$$

Obviously, $S(n,n) = 1$, and for $n \geq 1$, $S(n,1) = 1$.

2) List all the possible combinations and permutations.

In the calculation process of the total number, every one number counted represents one possible permutation under one possible combination. Combination determines the grouping of all the target debris, while permutation determines the removal sequence of the debris within a group. Therefore, all the possible combinations can be firstly listed. Based on it, all the possible permutations under every combination can be further listed.

3) Label every possible combination and permutation in a certain rule.

There are many ways to label the combinations and permutations of orbital transfers, where one of the simplest ones is provided. Firstly, all the n target debris are labelled from 1 to n in any way, which does not affect the following calculation procedures. Under this condition, the n numbers can construct a base-n system, so that each series of k mission designs are considered as a set of k numbers in this system. Subsequently, this set of k numbers can be converted from the base-n system into the base-10 system for comparison. When compared, each set of k numbers can be ordered in rising or falling value. Consequently, every possible series of k mission designs is labelled in a unique way, which acts as an index to enable the optimization process.

4) Establish the transformation relationship from the index to the total change in

velocity.

Using the calculation procedures of the velocity change between any two orbits, the total velocity change of a series of transfers can thereby be viewed as the sum of a series of single transfers. The value of each element of the sum is alterable according to their combinations and permutations, due to the variation of the starting and ending points of an orbital transfer. Therefore, the labelling of the combination and permutations of a series of transfer missions can be used to determine the sum of the series of velocity changes, namely, the total change in velocity. Altering the combination and permutation of the transfers can help minimise the total velocity change or the energy required, enabled by appropriate optimization algorithms.

5.4.3 Disposal Cost Optimization

Having established the disposal cost models and the transformation relationship from the index of an overall mission design to the total change in velocity, the total disposal costs corresponding to an overall mission design can thereby be calculated.

An optimization process can be enabled based on this mathematical mapping relationship, as shown in Fig. 5.18. Given the design configuration of a space system, each design of the overall removal mission can be labelled with an index, which acts as the variable of the optimization problem. The disposal cost modelling can be used as the objective function. By applying appropriate optimization algorithms, the optimal design solution can be reached with the minimum disposal costs.

Therefore, the proposed optimization problem has been constructed with the index of an overall mission design as the variable and the total disposal costs as the objective function. It is noteworthy that the variables are integers, and this type of optimization problems can be modelled as a MINLP problem, expressed by Eq. (5.4).

More specifically in this case, the variable is the index number of orbital transfer combinations and permutations, while the objective function is the disposal cost model established. Using the index, the sequence of all the orbital transfers can be acquired, and the corresponding total change in velocity can be calculated as the sum of the individual velocity changes required for single orbital transfers. With the input parameter of the total change in velocity, the disposal cost model outputs the total disposal costs as the objective function for the optimization process. Similar to the

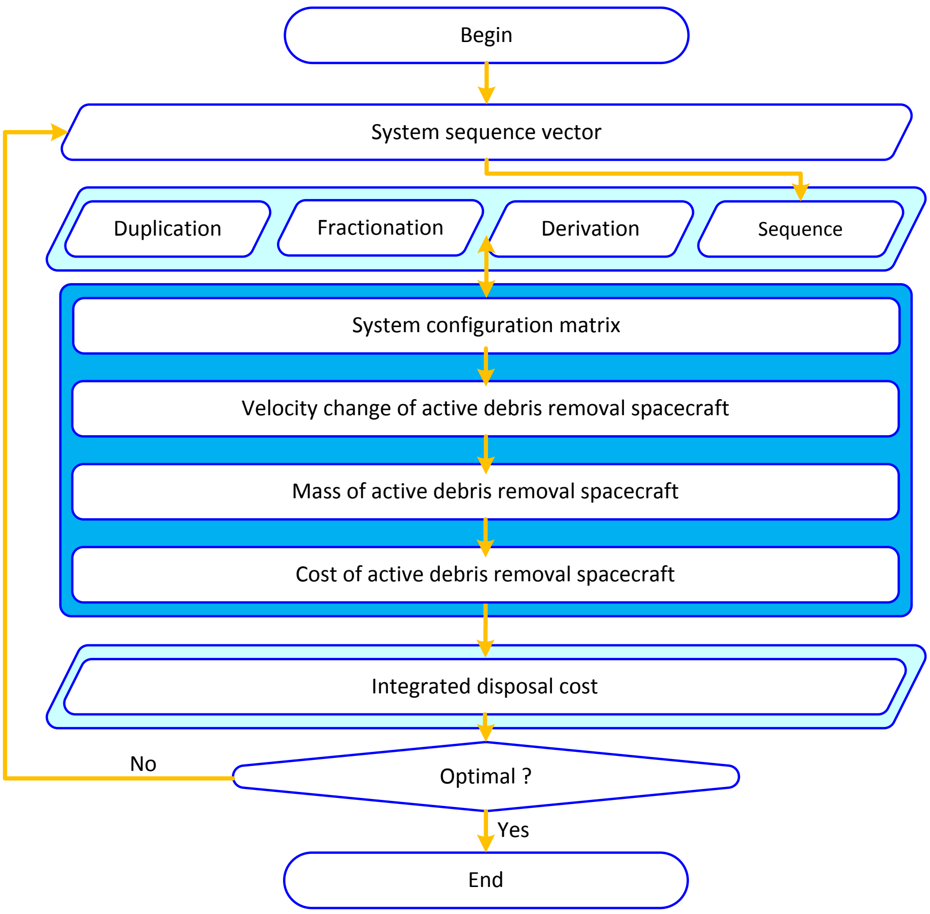


Fig. 5.18: System disposal cost optimization process

previous optimization processes, this is also a MINLP problem.

5.4.4 Summary

The approach of analysing and optimizing the cost of designing a series of missions to remove all the satellites or modules of a space system after failure or retirement has been developed under the proposed value-centric architecture.

All the possible groupings and sequencings of the target debris are listed and labelled through a specific combination and permutation method. This helps to improve the overall mission design by altering the grouping and the sequencing of all the target debris, which results in the reduction of the total velocity change and the corresponding disposal costs.

The costs of designing a series of missions to remove all the target debris has been modelled based on the total velocity change. This includes the development costs of multiple spacecraft each flying a multiple-rendezvous trajectory to deliver the de-orbit packages or kits and the expected costs of launching the entire spacecraft and the de-orbit packages or kits.

Having labelled all the possible groupings and sequencings of the target debris and established the value model, the optimization problem of the disposal mission design is constructed.

Chapter 6

Lifecycle Cost Integration and Implementation

Based on the analysis and optimization approaches developed for different lifecycle phases in Chapter 5, i.e., development, launch, operation, and retirement phases, a number of case studies are considered in this chapter to illustrate how to minimise different lifecycle phase costs respectively, i.e., the development, launch, replenishment, and disposal costs.

Through the integration of the available knowledge of different lifecycle phases, the methodology of analysing and optimizing the entire lifecycle costs of a space system is established in this chapter. The modelling of the costs of different lifecycle phases is merged into a standardised, comparable, and comprehensive metric for the design of system configurations to minimise the total costs in the system implementation from development to retirement. Provided that the space systems could achieve the same level of capability, this cost metric would be further viewed as the system value.

Having established the mathematical model of the proposed lifecycle cost optimization problem and identified the appropriate optimization algorithm, two representative mission cases are studied to show the concrete solution process. The Galileo FOC mission is a case of the system configuration design of constellations of identical spacecraft distributed in different orbits, while the RapideEye mission is a case of the system configuration design of cluster formation.

6.1 Lifecycle Phase Cost Implementation

In this section, the analysis and optimization processes of different lifecycle phases established in the previous chapter are implemented. To be specific, a series of typical space missions are reviewed and redesigned to illustrate how to minimise different lifecycle phase costs respectively, namely, the development, launch, replenishment, and disposal costs.

Having identified the proposed optimization problems as MINLP problems in the previous chapter, an appropriate optimization algorithm is required to fulfil the optimization processes. The general mathematical expression of these optimization problems is given by Eq. (5.4).

This is a so called Mixed Integer Programming Problem (MIPP), or more specifically, mixed integer nonlinear programming (MINLP) problem. Since the objective function is nonlinear, classical Mixed Integer Linear Programming (MILP) techniques such as branch and bound technique, cutting planes technique, etc., are obviously not applicable.

However, many stochastic algorithms that may be suitable have been developed or updated for MIPP in the latest decades. The Simulated Annealing (SA) technique, proposed by Kirkpatrick [167], has been proved to be a valuable tool in solving the MINLP problems [168]. Other techniques such as the Differential Evolution (DE) [169] and the Particle Swarm Optimization (PSO) [170] can also be used for the MINLP problems. However, most of these approaches cannot ensure the quality of the obtained solution or the efficiency of the optimization process.

Based upon one of the most widely-used evolution algorithms Genetic Algorithm (GA) , Deep [171] developed the MI-LXPM algorithm for the MINLP problems. This algorithm incorporates a special truncation procedure to handle integer restrictions on decision variables associated with a parameter free penalty approach for handling constraints. The algorithm also applies Laplace Crossover (LX) [172], Power Mutation (PM) [173], and tournament selection reproduction operator [171] to effectively increase the possibility of obtaining the global optimum.

Therefore, the MI-LXPM algorithm is adopted as the optimization algorithm in this research for the optimization processes of different lifecycle phase costs. The mission cases are studied to exhibit the effectiveness and applicability of the applied algorithm and the overall design architecture for the analysis of lifecycle costs.

To be specific, the multiple satellite cases and different space mission models are applied for the implementation of optimization of the development and replenishment costs, based on the corresponding subsystem cost models established in Section 5.1.1. In terms of the implementation of the launch and disposal costs, the case of the Galileo FOC constellation is used to illustrate the optimization process of multiple satellites. The Galileo case is further studied for the integration and implementation of the overall lifecycle costs, associated with the mission case of the RapidEye constellation. The former is a case of a constellation of identical satellites distributed in different orbits, while the latter is a case flying in a formation in close range.

6.1.1 Development Cost Implementation

In the development phase, the optimization problem can be constructed by the variables of design configurations and the objective function of the system development costs. Under this value-centric analysis and optimization process, the MI-LXPM algorithm can be applied to minimise the development costs, given different subsystem costs models or cases established in Section 5.1.1.

In this section, the example space system presented in Section 5.1.3 is still adopted. In the example, the space system is assumed to have 6 satellites labelled from 1 to 6, and 5 types of subsystems labelled from A to E in each satellite.

To be specific, there are 2 subsystems A allocated in 2 of all the 6 satellites, 4 subsystems B allocated in 2 satellites, 2 subsystems C allocated in 2 satellites, 1 subsystem D allocated in 1 satellite, and 3 subsystems E allocated in 2 satellites. It is noteworthy that the 2 satellites allocated with different types of subsystems are unnecessary to be the same 2 satellites. Furthermore, the elements of the System Sequence Vector Q respectively provide the label of the satellites that have the subsystem A to E.

All the above input parameters are summarised in Table 5.6, while the corresponding optimization results are presented in Table 6.1.

For comparisons, the optimized solutions of development costs are normalised by the corresponding spacecraft costs to give the cost fraction. For reference, the solutions of the Global Search (GS) are also provided to show that GA can effectively and efficiently find the global optimum. The simulation results also reveal that GA is able to find the optimum with approximately 98% on average of the GS time saved.

Table 6.1: Optimization results of development cost fraction of the original monolithic spacecraft costs (The abbreviation used are respectively Pct for Percentage, Fcn for Function, GA for Genetic Algorithm, and GS for Global Search.)

Inputs	Optimum (%)	GA Fcn Calls	Pct of GS Fcn Calls (%)	System Sequence Vector
Ørsted	98.38	1251	1.75	$\begin{bmatrix} 14 & 40 & 5 & 1 & 3 \end{bmatrix}^\mathsf{T}$
Freja	97.13	1951	2.73	$\begin{bmatrix} 7 & 26 & 3 & 2 & 3 \end{bmatrix}^\mathsf{T}$
SAMPEX	94.51	1551	2.17	$\begin{bmatrix} 6 & 44 & 6 & 2 & 2 \end{bmatrix}^\mathsf{T}$
PoSAT-1	97.98	2251	3.15	$\begin{bmatrix} 10 & 8 & 2 & 4 & 7 \end{bmatrix}^\mathsf{T}$
FireSat II	101.19	1601	2.24	$\begin{bmatrix} 7 & 21 & 1 & 1 & 3 \end{bmatrix}^\mathsf{T}$
SCS	102.32	1901	2.66	$\begin{bmatrix} 13 & 39 & 3 & 1 & 4 \end{bmatrix}^\mathsf{T}$
SSTL COTS	97.31	1951	2.73	$\begin{bmatrix} 7 & 8 & 7 & 1 & 17 \end{bmatrix}^\mathsf{T}$
Gen	98.30	1351	1.89	$\begin{bmatrix} 14 & 8 & 3 & 6 & 10 \end{bmatrix}^\mathsf{T}$
Com	103.11	1701	2.38	$\begin{bmatrix} 11 & 7 & 4 & 3 & 9 \end{bmatrix}^\mathsf{T}$
Obs	99.31	1501	2.10	$\begin{bmatrix} 7 & 6 & 3 & 1 & 19 \end{bmatrix}^\mathsf{T}$
Nav	103.57	2201	3.08	$\begin{bmatrix} 7 & 8 & 3 & 2 & 6 \end{bmatrix}^\mathsf{T}$
Sci	99.89	1901	2.66	$\begin{bmatrix} 13 & 8 & 13 & 1 & 9 \end{bmatrix}^\mathsf{T}$
Tec	97.42	1601	2.24	$\begin{bmatrix} 12 & 7 & 5 & 3 & 29 \end{bmatrix}^\mathsf{T}$
Hyb	101.77	1701	2.38	$\begin{bmatrix} 6 & 6 & 1 & 3 & 7 \end{bmatrix}^\mathsf{T}$

It is interesting to note that some optimal solutions of the cases studied are more than the value of monolithic configurations, which is 100.00% as the reference. This discrepancy could be attributed to the input value of System Fractionation Vector F in Table 5.6, which fixes degree of fractionation of design configurations and excludes the monolithic designs. In other words, the above optimization processes are executed to reach the optimal set of the System Sequence Vectors Q, given the degree of fractionation. Based upon this level of optimization, a higher level of optimization can be performed to seek the optimal System Fractionation Vector F, namely, conducting a nested optimization with the optimization of Q as inner problem and the optimization of F as outer problem.

Nested optimization helps find the best set of design configurations with the minimum development costs. The optimization solutions and the corresponding System Sequence Vector Q and System Fractionation Vector F are shown in Table 6.2. Through the nested optimizations, the development costs of all the cases are cut down, compare with the previous results in Table 6.1. It is worth noting that all the optimal solutions are no more than 100.00%, which indicates the optimal fractionated designs can be found with less development cost than monolithic designs in these cases.

Table 6.2: Nested optimization results of development cost fraction of the original monolithic spacecraft costs (The abbreviation used are respectively Pct for Percentage, Fcn for Function, GA for Genetic Algorithm, and GS for Global Search.)

Inputs	Optimum (%)	System Fractionation Vector	System Sequence Vector
Ørsted	95.94	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 3 & 5 & 3 & 1 & 1 \end{bmatrix}^\intercal$
Freja	94.11	$\begin{bmatrix} 1 & 2 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 5 & 21 & 5 & 1 & 1 \end{bmatrix}^\intercal$
SAMPEX	92.24	$\begin{bmatrix} 1 & 2 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 3 & 23 & 3 & 1 & 1 \end{bmatrix}^\intercal$
PoSAT-1	95.26	$\begin{bmatrix} 2 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 11 & 1 & 5 & 5 & 3 \end{bmatrix}^\intercal$
FireSat II	97.70	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 3 & 5 & 3 & 1 & 1 \end{bmatrix}^\intercal$
SCS	98.79	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 2 & 5 & 2 & 1 & 1 \end{bmatrix}^\intercal$
SSTL COTS	95.65	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 2 & 1 & 2 & 2 & 5 \end{bmatrix}^\intercal$
Gen	95.94	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 3 & 2 & 3 & 1 & 1 \end{bmatrix}^\intercal$
Com	98.89	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 4 & 1 & 4 & 1 & 5 \end{bmatrix}^\intercal$
Obs	96.43	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 4 & 1 & 4 & 4 & 3 \end{bmatrix}^\intercal$
Nav	99.58	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 3 & 1 & 3 & 1 & 2 \end{bmatrix}^\intercal$
Sci	96.97	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 4 & 1 & 4 & 4 & 5 \end{bmatrix}^\intercal$
Tec	95.05	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 4 & 1 & 4 & 4 & 2 \end{bmatrix}^\intercal$
Hyb	98.01	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$	$\begin{bmatrix} 3 & 1 & 3 & 3 & 2 \end{bmatrix}^\intercal$

It is worth noting that, for the System Fractionation Vector F of $\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$, it means that each type of the subsystems A to E is respectively integrated into a satellite, however, different types of the subsystems may not be integrated into the same satellite. In other words, all the subsystem A are allocated in a satellite, while subsystem A and subsystem B may be in different two satellites.

6.1.2 Launch Cost Implementation

In the launch phase, the design configurations and the launch costs are respectively used as the variables and the objective function of the optimization process. With the detailed problem description and solution approach previously provided, a few typical launch cases of space systems are reviewed and redesigned.

More specifically, the cases consist of a case of monolithic spacecraft HST and two cases of the constellations of identical spacecraft Galileo and Flock 3p. The Galileo constellation is a case of large satellites, while the Flock 3p is a case of small satellites. The optimal solutions of these cases are shown in Table 6.3, and the corresponding launch strategies with the minimum costs are exhibited in Table 6.4.

It is noteworthy that the expected cost of the actual launch vehicle is also included for comparison in Table 6.3. The actual costs are usually less than these expected costs, since the actual launches were successful and the expected ones also considered the risk of launch failures. Moreover, the optimized solutions generally provide another launch plan, which is expected to be better than the actual ones. However, due to the facts of launch successes or non-dedicated launches such as rideshare and piggyback launches, the actual costs may be less than the optimized ones, while the expected

Table 6.3: Cases of dedicated launch cost optimization

Mission	Actual cost (FY2010$M)	Expected cost of actual launch vehicle (FY2010$M)	Optimized expected cost (FY2010$M)
HST	408.6	492.4	336.3
Galileo	569.0	854.4	408.4
Flock 3p	~10.0	35.4*	28.22

* This cost is estimated based on the dedicated launches with associated risks, while the cost of the piggyback launch that was actually used is given in the actual cost column

Table 6.4: Optimal dedicated launch strategies for each case

Mission	Expected launch vehicles	Launch attempts	Number of satellites per launch
HST	Delta 4M, 4M+, 4H	1	1
Galileo	Long March 2C & 2D	10	1
	Long March 3A &3C	1	4
Flock 3p	Long March 2C & 2D	1	88

costs of the actual launches are all more than the optimized ones.

The first case is a monolithic spacecraft, the largest and most powerful on-orbit telescope HST [130]. HST had a mass of 11866kg, and its development cost was over $1500M in the FY 2010 [65]. It was deployed by the Space Shuttle Discovery on April 24, 1990, with the launch cost estimated more than $400M [87].

In this case, any launch failure is intolerable, since the development cost of the satellite is far more expensive than the launch cost. The risk level of this launch mission needs to be controlled to be as low as possible. Therefore, more reliable launch vehicles with adequate launch capability are more appealing. If this mission was implemented by currently available launch vehicles, Delta 4 would probably be the best solution, since only a few launch vehicles meet the mass and size requirement and the space shuttle has been retired.

The second case exhibits the launch strategy optimization of a constellation such as the European GNSS Galileo. The Galileo FOC consists of 14 identical navigation satellites each of mass 733kg [174], which were launched by 6 independent launch missions (the former 5 missions launched 2 satellites and the last one launched 4 satellites). Due to different orbit requirements and manufacturing time, the launch sequence is retained in this case, thus only the launch strategy for each mission is optimized. The results of this case reveal that the launch missions of Galileo FOC are well-organized, and the optimal strategy only saves 28.2% of the original cost.

The last is a small satellite constellation case. Flock 3p is a constellation with 88 cubesats each weighting 4.7kg [105]. The mission was performed by PSLV-XL on February 15, 2017 [104], which launched a record 104 satellites in a single launch attempt.

The reason for the reduction of the launch cost of Flock 3p is that the constellation is the secondary payload of this launch mission. However, the objective of the proposed approach is to find a better dedicated launch strategy. The result shows that the optimal solution saves 20% of the expected launch cost, but still costs more than the actual mission. This proves the advisability and advantage of rideshare or piggyback launches for small satellite launches.

This result has two potential implications for the space launch market. On the one hand, current dedicated launches are not economically comparable with rideshare or piggyback launches in the small satellite market. On the other hand, innovative launch vehicles need to be developed, especially for the small satellite market.

6.1.3 Replenishment Cost Implementation

Similar to the previous two lifecycle phases, the design configurations are also used as the variables of the optimization process in the operation phase, while the objective function is substituted by the replenishment costs. Using the proposed value-centric design approach, a few mission cases are studied to exhibit how different design configurations influence the overall system replenishment costs in this section.

Different subsystem cost cases are studied for replenishment cost analysis and optimization, associated with different System Derivation Vectors V given in Table 5.14. For the system elements, the example space system illustrated in Section 6.1.1 with 6 satellites labelled from 1 to 6 and 5 types of subsystems labelled from A to E is also adopted in this section.

The results corresponding to V_1 and V_2 are presented respectively in Table 6.5 and Table 6.6. For reference, the optimal solutions are also acquired by the GS to verify the correctness, applicability, and effectiveness of the optimization algorithms. For comparison, all the results are normalised by the total spacecraft costs to give the cost fraction.

From the results presented in Table 6.5, we can see that the GA saves approximately 98% on average of the total calls that the GS requires, which is consistent across the designs. This indicates that GA can effectively and efficiently seek the optimal set of solutions, and can be applied for further designs.

Adopting V_1 as the input of the degree of derivation, the corresponding results of different cost models and cases are shown in Table 6.5. Since the elements of the System Derivation Matrix V_1 are uniform, the effects of fractionation are shown in this series of case studies.

Taking the FireSat II as an example, an optimal solution, its System Sequence Vector Q can be transformed into the corresponding System Configuration Matrix N, seen in Eq. (6.1), associated with the inputs of the degree of duplication and fractionation. The details of this transformation process are described and exemplified

Table 6.5: Solutions of replenishment cost fraction of spacecraft costs with input V_1 (The abbreviation used are respectively Pct for Percentage, Fcn for Function, GA for Genetic Algorithm, and GS for Global Search.)

Inputs	Optimum (%)	GA Fcn Calls	Pct of GS Fcn Calls (%)	System Sequence Vector
Ørsted	5.12	851	1.19	$\begin{bmatrix} 11 & 32 & 11 & 1 & 9 \end{bmatrix}^\intercal$
Freja	5.07	951	1.33	$\begin{bmatrix} 12 & 33 & 2 & 3 & 21 \end{bmatrix}^\intercal$
SAMPEX	4.74	1151	1.61	$\begin{bmatrix} 14 & 8 & 14 & 1 & 3 \end{bmatrix}^\intercal$
PoSAT-1	6.27	751	1.05	$\begin{bmatrix} 11 & 7 & 2 & 5 & 20 \end{bmatrix}^\intercal$
FireSat II	7.33	851	1.19	$\begin{bmatrix} 12 & 7 & 5 & 1 & 10 \end{bmatrix}^\intercal$
SCS	7.51	801	1.12	$\begin{bmatrix} 12 & 7 & 2 & 1 & 24 \end{bmatrix}^\intercal$
SSTL COTS	5.58	601	0.84	$\begin{bmatrix} 11 & 9 & 11 & 1 & 20 \end{bmatrix}^\intercal$
Gen	5.75	1151	1.61	$\begin{bmatrix} 15 & 9 & 4 & 5 & 9 \end{bmatrix}^\intercal$
Com	7.66	801	1.12	$\begin{bmatrix} 15 & 10 & 5 & 6 & 30 \end{bmatrix}^\intercal$
Obs	6.76	701	0.98	$\begin{bmatrix} 15 & 44 & 15 & 1 & 9 \end{bmatrix}^\intercal$
Nav	7.77	951	1.33	$\begin{bmatrix} 13 & 8 & 13 & 5 & 9 \end{bmatrix}^\intercal$
Sci	5.90	901	1.26	$\begin{bmatrix} 12 & 7 & 12 & 1 & 5 \end{bmatrix}^\intercal$
Tec	4.81	651	0.91	$\begin{bmatrix} 6 & 6 & 2 & 3 & 11 \end{bmatrix}^\intercal$
Hyb	7.33	1251	1.75	$\begin{bmatrix} 9 & 6 & 1 & 2 & 6 \end{bmatrix}^\intercal$

in Section 4.4.1.

$$N = \begin{bmatrix} 0 & 2 & 1 & 1 & 1 \\ 0 & 0 & 0 & 0 & 0 \\ 1 & 2 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 1 & 0 & 1 & 0 & 2 \end{bmatrix} \tag{6.1}$$

The System Configuration Matrix N expresses the configuration design in the form of matrices, shown in Fig. 6.1, where subsystem A to E are represented by the column 1 to 5 in N and satellite 1 to 6 are denoted by the row 1 to 6 in N.

Since all the satellites are equivalent (the degree of derivation is set to be uniform across satellites for the input V_1), such an optimal design configuration is mainly determined by the degree of duplication and fractionation, reducing the total replenishment costs by increasing the reliability and decreasing the cost penalty of fractionation.

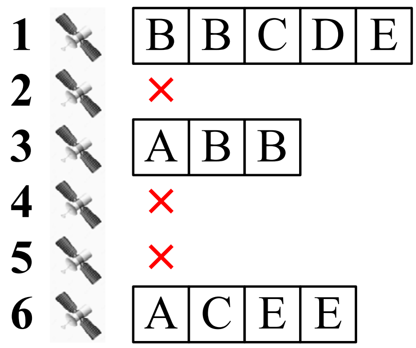


Fig. 6.1: Optimal configuration with minimum replenishment costs for input V_1

Adopting V_2 as the input of the degree of derivation, the corresponding results can be seen in Table 6.6. It is noted that since the value of the elements in the input degree of derivation increases from row 1 to row 6, instead of the uniform value in V_1, the optimization results exhibit the effects of different degree of derivation on system design configurations.

Also applying the example of the FireSat II, the System Configuration Matrix N, corresponding to the optimal System Sequence Vector Q, can be derived according to the degree of duplication and fractionation, as shown in Eq.(6.2).

$$N = \begin{bmatrix} 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 1 & 2 & 1 & 0 & 2 \\ 1 & 2 & 1 & 1 & 1 \end{bmatrix} \tag{6.2}$$

Similarly, the above System Configuration Matrix N can be interpreted into the design configuration of the space system with the minimum replenishment costs, as shown in Fig. 6.2.

The optimal solutions can be understood as controlling the mission risks by using mature design. Most subsystems tend to be allocated in the last row, namely, the last satellite, which is defined by V_2 to have the highest design reliability. If it were not for the System Fractionation Vector F that restricts the degree of fractionation for each subsystem, all of them would locate in the last row or satellite. In this series of cases, the System Derivation Matrix V is the dominant factor, regardless of different inputs of cost models.

Table 6.6: Solutions of replenishment cost fraction of spacecraft costs with input V_2 (The abbreviation used are respectively Pct for Percentage, Fcn for Function, GA for Genetic Algorithm, and GS for Global Search.)

Inputs	Optimum (%)	GA Fcn Calls	Pct of GS Fcn Calls (%)	System Sequence Vector
Ørsted	4.58	1651	2.31	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
Freja	4.56	1101	1.54	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
SAMPEX	4.24	1851	2.59	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
PoSAT-1	5.58	1651	2.31	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
FireSat II	6.53	1551	2.17	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
SCS	6.71	1851	2.59	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
SSTL COTS	4.89	1701	2.38	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
Gen	5.11	1301	1.82	$\begin{bmatrix} 15 & 44 & 15 & 6 & 29 \end{bmatrix}^\mathsf{T}$
Com	6.85	1501	2.10	$\begin{bmatrix} 15 & 44 & 15 & 6 & 5 \end{bmatrix}^\mathsf{T}$
Obs	5.96	1801	2.52	$\begin{bmatrix} 15 & 44 & 15 & 6 & 5 \end{bmatrix}^\mathsf{T}$
Nav	6.92	1901	2.66	$\begin{bmatrix} 15 & 44 & 15 & 6 & 5 \end{bmatrix}^\mathsf{T}$
Sci	5.22	1751	2.45	$\begin{bmatrix} 15 & 44 & 15 & 6 & 5 \end{bmatrix}^\mathsf{T}$
Tec	4.26	2151	3.01	$\begin{bmatrix} 15 & 44 & 15 & 6 & 5 \end{bmatrix}^\mathsf{T}$
Hyb	6.52	1751	2.45	$\begin{bmatrix} 15 & 44 & 15 & 6 & 5 \end{bmatrix}^\mathsf{T}$

It is also worth noting that, the solution provided is only one of the set of optimal solutions. In some cases, there might be many optimal solutions; even so, it is just a very tiny proportion of the total population. On the other hand, the diversity of the optimal solutions might be one of the favourable characteristics for the further optimization of system lifecycle costs.

6.1.4 Disposal Cost Implementation

Given the design configuration of a space system, the index of the design of a series of removal missions can be used as the variable to enable an optimization process in the retirement phase with the disposal costs as the objective function. Since the value flow from the overall mission designs to the total disposal costs has been completed and the appropriate optimization algorithm has been identified, a case study is conducted in this section to show the concrete solution process.

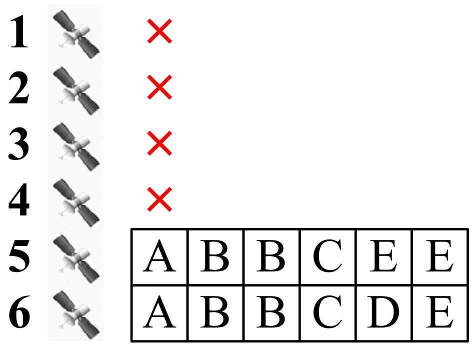


Fig. 6.2: Optimal configuration with minimum replenishment costs for Input V_2

The case is the disposal mission of all the Galileo FOC satellites after retirement, regardless of the EOL de-orbit devices equipped. During the simulation, the following assumptions are made to simplify the calculation process:

1) The time of the transfer trajectory is not taken into account, namely, no constraint of the transfer time for orbital manoeuvres.

2) The orbital elements the Right Ascension of the Ascending Node (RAAN) and the argument of perigee of all the Galileo FOC satellites are assumed to be the same, so that the solutions of the Hohmann transfer can be used to estimate the minimum velocity change between two orbits.

3) The liquid propellant with a typical specific impulse of 340s is applied for all the propulsion subsystems or de-orbit devices. The specific impulse is used to calculate the mass of the propulsion subsystems or de-orbit devices by the Tsiolkovsky equation, given by Eq. (5.38), with the input parameter of delta vee. Other values can also be applied.

The masses and orbit elements of the Galileo FOC satellites are listed in Table 6.7.

By applying the proposed approach, the optimal solution of the overall mission design is achieved with a minimum cost of $1648.0 million US to remove all the Galileo FOC satellites. The high launch costs mainly account for this huge disposal cost. Since the Galileo FOC satellites are in Medium Earth Orbits (MEO), there are only limited available launch vehicles with the adequate payload capability. The launching of the heavy mission spacecraft is necessary to carry the Galileo FOC satellites and support the atmosphere disposal manoeuvres required.

The detailed mass and cost breakdowns of the spacecraft launched for removal missions can be seen in Table 6.8, and the corresponding mission design is exhibited in

Table 6.7: Masses and orbit elements of Galileo FOC satellites [32]

Satellite	Perigee(km)	Apogee(km)	Eccentricity	Inclination(°)	Mass(kg)
Galileo FOC 1	17231	25971	0.1560	49.8	733.0
Galileo FOC 2	13810	25918	0.2310	49.8	733.0
Galileo FOC 3	23516	23574	0.0010	55.0	733.0
Galileo FOC 4	23353	23382	0.0005	55.1	733.0
Galileo FOC 5	23218	23240	0.0004	57.1	733.0
Galileo FOC 6	23220	23239	0.0003	57.1	733.0
Galileo FOC 7	23265	23305	0.0007	54.6	733.0
Galileo FOC 8	23550	23618	0.0011	55.0	733.0
Galileo FOC 9	23551	23568	0.0003	55.0	733.0
Galileo FOC 10	23272	23280	0.0001	57.4	733.0
Galileo FOC 11	23483	23530	0.0008	57.4	733.0
Galileo FOC 12	23039	23055	0.0003	54.6	733.0
Galileo FOC 13	22982	22982	0.0000	54.6	733.0
Galileo FOC 14	23272	23296	0.0004	54.6	733.0

Table 6.8: Mass and cost breakdowns of optimal solution

Parameter	Mission 1	Mission 2	Mission 3
Mission spacecraft mass (kg)	4523.2	5992.0	8077.2
Spacecraft cost (FY2010$M)	241.8	252.5	274.2
Launch cost (FY2010$M)	282.4	290.4	306.6
Total cost (FY2010$M)	524.3	542.9	580.8

Table 6.9: Optimal solution of disposal mission design

Mission	Sequence	Target Satellite	Launch vehicle
1	1	Galileo FOC FM 1	Falcon 9 V1.0, 1.1, FT
1	2	Galileo FOC FM 2	
2	1	Galileo FOC FM 11	Falcon 9 V1.0, 1.1, FT
2	2	Galileo FOC FM 10	
2	3	Galileo FOC FM 6	
2	4	Galileo FOC FM 5	
3	1	Galileo FOC FM 4	Falcon 9 V1.0, 1.1, FT
3	2	Galileo FOC FM 8	
3	3	Galileo FOC FM 9	
3	4	Galileo FOC FM 3	
3	5	Galileo FOC FM 7	
3	6	Galileo FOC FM 14	
3	7	Galileo FOC FM 12	
3	8	Galileo FOC FM 13	

Table 6.9. In the table, the first column gives the combination or grouping of all the satellites. The satellites in the same group are removed by the same mission spacecraft. The second column gives the removal permutation of the satellites within the group, which represents the sequence to be removed by a mission spacecraft.

6.2 Lifecycle Cost Integration and Implementation

Having modelled, analysed, and optimized the major costs of a space system in different lifecycle phases, this section aims at integrating all the lifecycle phase costs into the entire lifecycle costs. Overall, the integrated value provides a standardised, comparable, and comprehensive metric for the design trade-offs between different lifecycle phases. By applying the integrated value as the objective function, the design configurations can be analysed and optimized to minimise the total costs in the system implementation from development to retirement.

6.2.1 Lifecycle Cost Modelling

Lifecycle cost modelling is the integration of the modelling previously established of development, launch, replenishment, and disposal costs. In other words, the different metrics of the development, launch, operation, and retirement phases are summed to generate an overall value for lifecycle cost analysis and optimization.

6.2.2 Lifecycle Cost Integration

Having formulated the costs in different lifecycle phases such as development, launch, operation, and retirement phases, the integration of these costs is executed to generate an appropriate value for the analysis and optimization of space system design configurations. Since the system value is integrated in the system characteristic space, the integration process builds the bridge between the design configurations and lifecycle costs. The corresponding calculation procedures are illustrated in Fig. 6.3.

1) The system design configuration is defined by the degree of duplication, fractionation, and derivation, which can be equivalently transformed into a more intuitive expression, that is, the System Configuration Matrix N.

2) The subsystem costs and intrinsic properties, e.g., mass, reliability, and orbit, are modelled and applied to calculate the costs and properties of all the subsystems of each satellite in a space system.

3) The costs and intrinsic properties of all the subsystems are then integrated to generate the costs of each satellite in different lifecycle phases such as development, launch, operation, and retirement phases, through the system configuration matrix. The calculation of the costs in different lifecycle adopts different value models with different parameters of the costs and properties as the inputs.

4) The total costs in each lifecycle phase are the sum of those of all the satellites in this system.

5) Finally, the system lifecycle costs are the sum of the total costs of the space system in all the four lifecycle phases.

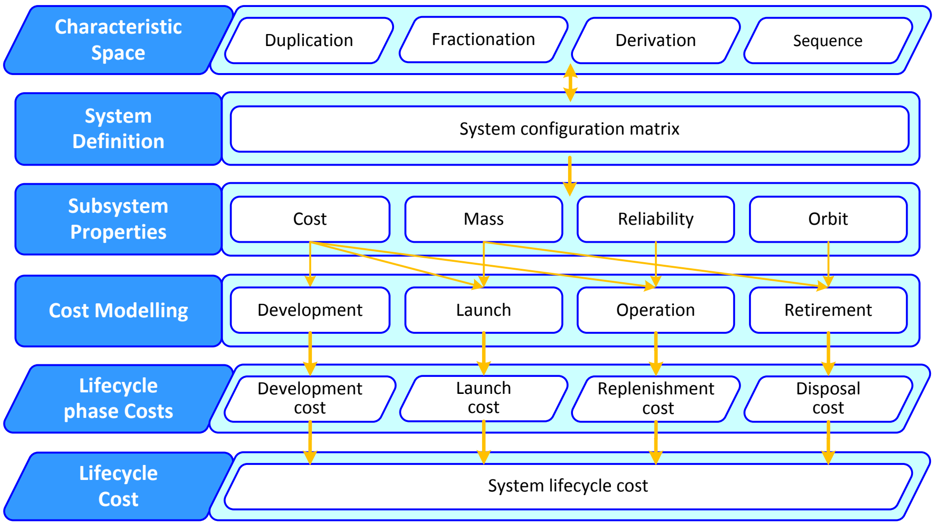


Fig. 6.3: System lifecycle cost integration process

6.2.3 Lifecycle Cost Optimization

Having established the mathematical relationships between design configurations and lifecycle costs, this section completes the final step for the proposed value-centric design loop, encouraging the system to be designed through a global optimization process. Since the value flow from the overall system design configurations to the costs of different lifecycle phases has been completed, the sum of these costs, that is, the total lifecycle costs, can be naturally derived from the overall system design configurations. This integrated analysis meets all the three conditions required by a perfect optimization process [122], namely:

1) Generate all possible designs. Each design of the overall space system configuration is labelled with an index, which can act as the variable of the optimization problem.

2) Evaluate each design. The lifecycle cost modelling, which can be integrated from the modelling of development, launch, replenishment, and disposal costs, can be used as the objective function of the optimization problem.

3) Choose the design with the highest value. The appropriate optimization algorithm can be applied to seek the optimal design solution with the minimum lifecycle costs.

Therefore, the system design configurations can be used as the variables of the entire lifecycle cost optimization. Accordingly, this optimization process can be executed in the system characteristic space by altering the design configurations. The value used as the objective function is the total lifecycle costs. Similar to the lifecycle phase optimization problems established, the variables of this optimization problem are also integers. This type of optimization problem can be modelled as a MINLP problem, with the general mathematical expression given by Eq. (5.4).

More specifically, the concrete calculation procedures are shown in Fig. 6.4. During the four lifecycle phases, the System Sequence Vector is consistent. This enables it as the design variable throughout the entire lifecycle of a space system. Associated with the input parameters of duplication, fractionation, and derivation, the System Sequence Vector can be transformed into the complete system configuration designs. Based on different system designs, the intrinsic properties such as mass, cost, and reliability, can be integrated into different lifecycle phase costs. The corresponding cost models of the four lifecycle phases, i.e., development, launch, operation, and retirement phases, have been illustrated in Chapter 5.

Since the four metrics are all converted into the monetary dimension, they can be summed into an overall metric, namely, the system lifecycle cost. Adopting the system lifecycle cost as an overall objective function, an optimization process of the entire lifecycle phase can be enabled. Similarly, the MI-LXPM algorithm can be applied to solve such an optimization problem of design configurations at the overall level.

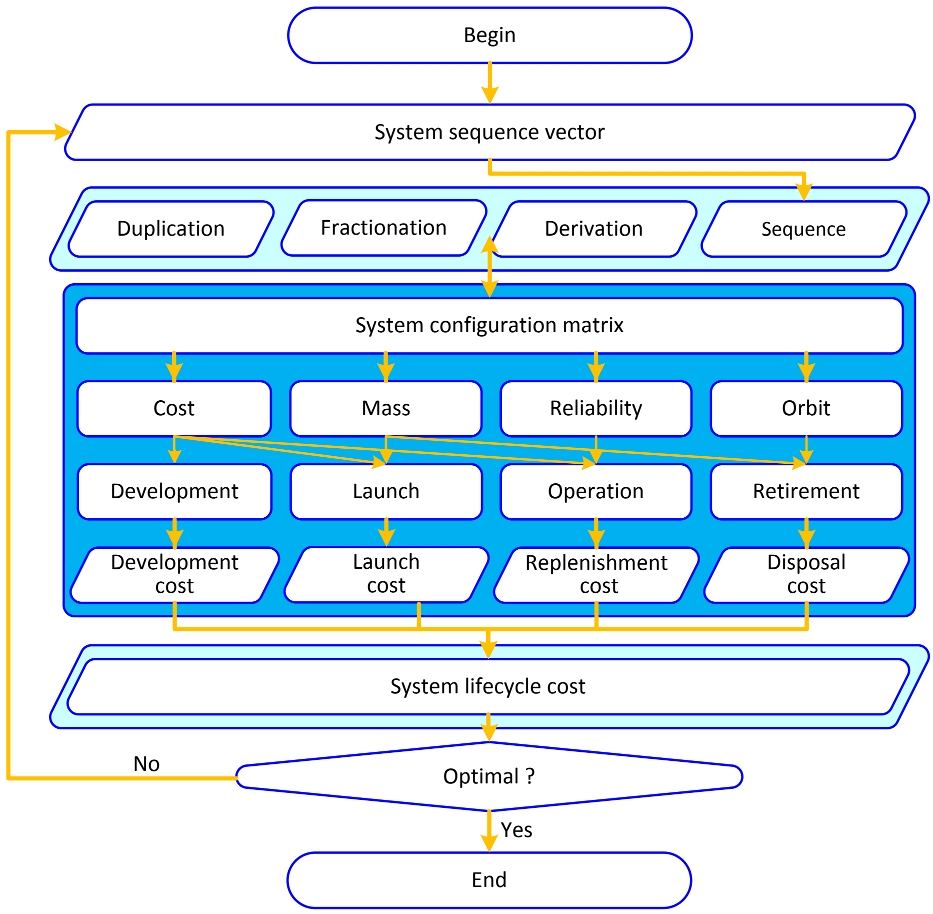


Fig. 6.4: System lifecycle cost optimization process

6.2.4 Lifecycle Cost Implementation

Having established the proposed optimization problem of the entire lifecycle costs and identified the appropriate optimization algorithm, two typical mission cases are conducted to show the concrete solution process. One case is the design of the Galileo FOC constellation, which has previously been used in the lifecycle phase cost analysis. The other case is the design of the LEO observation satellite constellation RapidEye.

(1) Galileo FOC constellation

The mass and orbit parameters of the Galileo FOC satellites have been presented in Section 6.1.2 and Section 6.1.4, which are summarised in Table 6.7. In terms of the detailed parameters of individual satellites, the mass breakdown of typical navigation satellites in the textbook Space Mission Engineering: The New SMAD [87] is adopted to calculate the masses of different subsystems, and these masses are further applied as the inputs to subsystem development cost models established in Section 5.1.1, namely, USCM and SSCM CERs, to calculate the development costs of each subsystem. Since satellite structure is not a subsystem that can be fractionated, the mass of structure is distributed into the other subsystems. In summary, the mass and cost breakdown of Galileo FOC satellites is listed in Table 6.10.

Table 6.10: Mass and cost breakdown of Galileo FOC satellites

Subsystem	Mass percentage (%)	Mass (kg)	Cost (FY2010$M)
PLS	32.7	239.6	19.3
EPS	50.2	367.7	18.2
TTC	4.6	33.9	4.8
CDH	3.8	27.7	3.9
ACS	8.7	64.1	11.8
TOT	100.0	733.0	58.0
WSS	3.0	22.0	1.7

In this case, the fractionated modules of a satellite cannot be shared with any other satellite, since the Galileo FOC satellites are in different orbits or not in range of near field WSS. Overall, the configuration of the Galileo FOC constellation is the constellation of identical spacecraft. The fractionation design of such a system configuration is only within the satellite rather than across the satellites. Therefore, the configuration of each satellite can be designed independently.

The input parameters are listed in Table 6.11, and the corresponding optimal solution is presented in Table 6.12. For comparison, the monolithic and completely fractionated solutions are also provided in Table 6.12, associated with the optimal solution.

Since each satellite is identical and designed independently, the design of one satellite is considered in this case, which can be used for all the 14 satellites. Thus, the elements of the Satellite Duplication Vector P can be set to be all 1, namely, one for each subsystem A to E in a satellite. The Satellite Derivation Matrix V is set to be 0.5 for all the elements, which means the satellites are set to be half derivative to the traditional designs. Thus, the average of STD and SSAD reliability models established in Section 5.3.1 is adopted as the reliability of each subsystem.

As shown in Table 6.12, the launch and disposal costs of the optimal solution of the Galileo FOC constellation are consistent with the results previously obtained in the launch and retirement phases. Since the satellites are launched into different orbits, the overall constellation configuration is inalterable in this case, which restricts the

Table 6.11: Inputs of lifecycle cost optimization of Galileo FOC constellation

Parameters	Symbols	Values
Number of satellites	m	14
Number of subsystems	n	5
System Duplication Vector	P	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$
System Fractionation Vector	F	$\begin{bmatrix} 1 & 1 & 1 & 1 & 1 \end{bmatrix}^\intercal$
System Derivation Matrix	V	$\begin{bmatrix} 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \\ 0.5 & 0.5 & 0.5 & 0.5 & 0.5 \end{bmatrix}$

Table 6.12: Breakdown and comparison of different solutions of Galileo FOC constellation (FY2010$M)

Lifecycle phase	Optimal average	Optimal	Monolithic	Fractionated
Development	58.9	824.8	805.4	977.8
Launch	47.3	661.9	655.9	708.8
Operation	15.3	214.0	756.2	225.4
Retirement	117.7	1648.0	1648.0	1648.0
Overall	239.2	3348.7	3865.5	3560.0

feasible domain of system designs.

In the development phase, the optimization of the design configuration of a satellite reduces the IAT costs and the cost penalties of fractionation, while the learning curve factor decreases the manufacturing costs of the entre constellation. In the operation phase, the design configuration optimization helps reduce the replenishment costs, through the fractionation of functionalities and the improvement of the overall reliability. Due to the location in MEO, the launch missions and the atmosphere entry removal missions are costly. Thus, the EOL de-orbit devices are essential for such satellites. The corresponding costs are estimated in the development phase, namely, the development costs of the EOL de-orbit devices, which are much economical than the active removal missions.

Table 6.12 also compares the expected costs of different solutions of design configurations in different lifecycle phases. The monolithic design has the lowest development costs, but simultaneously has the highest replenishment costs. The completely fractionated design can effectively reduce the costs in the operation phase, but suffers the cost penalties of the WSS in the development phase. The optimal design with the minimum lifecycle costs has a balance between the development and the replenishment costs. It saves most of the replenishment costs of the monolithic design, and has fewer cost penalties than the completely fractionated design.

Through the backward transformation, the optimal System Configuration Matrix N can be obtained from the inputs of the System Sequence Vector Q, the System Duplication Vector P, and the System Fractionation Vector F, seen in Eq. (6.3). The details of this transformation process have been described and exemplified in Section 4.4.1.

$$N = \begin{bmatrix} 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 1 & 0 & 0 & 0 & 0 \\ 0 & 1 & 1 & 1 & 1 \\ 0 & 0 & 0 & 0 & 0 \end{bmatrix} \tag{6.3}$$

The System Configuration Matrix N expresses the design configuration in the form of matrices, as intuitively shown in Fig. 6.5, where subsystem A to E, namely, PLS, EPS, TTC, CDH, and ACS, are represented by the column 1 to 5 in N, and satellite 1 to 5 are denoted by the row 1 to 5 in N.

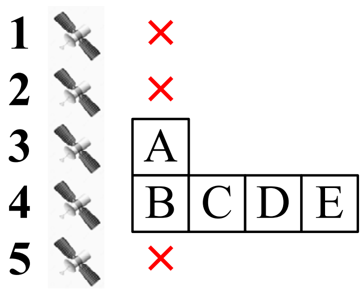


Fig. 6.5: Optimal configuration per Galileo FOC satellite with minimum lifecycle costs

It can been seen in Fig. 6.5 that the optimal design configuration separates the Subsystem A, namely, PLS, from the infrastructure modules, namely, EPS, TTC, CDH, and ACS. This is an effective way to control the risks in the operation phase, since the PLS is the most significant and expensive module in a Galileo FOC satellite. The integration of different infrastructure modules can reduce the cost penalties of the WSS, compared with the completely fractionated design that fractionates all the subsystems.

However, the fractionation of the ACS has not been realised in the current space industry, the above optimal solution only exists theoretically. Practically, the ACS is required to be integrated with the PLS. Therefore, such a constraint is applied for a further optimization process to seek a practical optimal solution.

This constraint can be addressed in two ways:

a) View the PLS and the ACS as a virtual subsystem. In this condition, such an integrated subsystem associated with the other three subsystems, i.e., EPS, TTC, and CDH, consists of a new 4-subsystem optimization process. The remainder of the solution process is same with the above 5-subsystem optimization process.

b) Introduce a penalty function to rule out the design configuration without the ACS to be integrated in the PLS. This way is easier to be implemented, but takes more calculation time, compared with the first one.

Either way can effectively reach the practical optimal solution shown in Fig. 6.6, while the difference of the cost breakdowns between the constrained and monolithic solution is highlighted in Table 6.13.

Table 6.13: Breakdown and comparison of different constrained solutions of Galileo FOC constellation (FY2010$M)

Lifecycle phase	Constrained average	Constrained optimal	Monolithic
Development	57.5	805.1	805.4
Launch	46.8	655.8	655.9
Operation	20.6	288.7	756.2
Retirement	117.7	1648.0	1648.0
Overall	242.7	3397.6	3865.5

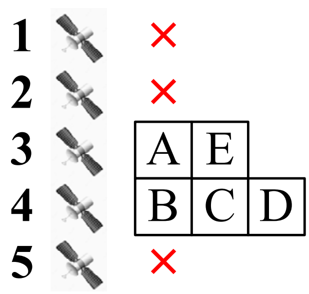


Fig. 6.6: Constrained optimal configuration per Galileo FOC satellite with minimum lifecycle costs

As Table 6.13 shows, the constrained optimal solution costs more than the previous theoretical optimal solution, but still much less than monolithic solution. The fractionated solution is not listed in Table 6.13, since it cannot meet the constraint of the integrated PLS and ACS. Furthermore, the above two ways can also be applied to the integration of other subsystems.

(2) RapidEye constellation

As previously described in Section 3.3, the LEO constellation RapidEye is a constellation consisting of five identical observation satellites flying in a close range. In this case, the number of satellites is kept, but the satellites are not required to be identical. Thus, the configuration of all the satellites can be designed and optimized altogether.

The mass and orbit parameters of the RapidEye satellites are listed in Table 6.14.

Table 6.14: Masses and orbit elements of RapidEye satellites [32]

Satellite	Perigee(km)	Apogee(km)	Eccentricity	Inclination(°)	Mass(kg)
RapidEye 1	613	646	0.0024	97.9	175.0
RapidEye 2	621	638	0.0012	97.9	175.0
RapidEye 3	621	637	0.0011	97.9	175.0
RapidEye 4	621	638	0.0012	97.9	175.0
RapidEye 5	617	642	0.0018	97.9	175.0

Similar to the Galileo FOC constellation case, the mass breakdown of typical observation satellites in the textbook *Space Mission Engineering: The New SMAD* [87] is adopted to calculate the masses of different subsystems. These masses are further applied as the inputs to SSCM CERs to calculate the development costs of each subsystem, since the RapidEye satellites are small satellites. In summary, the mass and cost breakdown of the RapidEye satellites is exhibited in Table 6.15.

Table 6.15: Mass and cost breakdown of RapidEye satellites

Subsystem	Mass percentage (%)	Mass (kg)	Cost (FY2010$M)
PLS	47.7	83.5	7.1
EPS	32.3	56.5	8.6
TTC	3.1	5.4	1.0
CDH	7.7	13.5	3.2
ACS	9.2	16.2	4.9
TOT	100.0	175.0	24.8
WSS	3.0	5.3	0.5

The input parameters are listed in Table 6.16, and the corresponding optimal solution is presented in Table 6.17. For comparison, the solution of the constellation of identical spacecraft is also provided in Table 6.17.

In this case, all the 5 satellites in the RapidEye constellation with 5 types of subsystems are designed together. Thus, the elements of System Duplication Vector P are set to be 5 respectively for subsystem A to E. Since RapidEye satellites are all small satellites, the System Derivation Matrix V is set to 0, which means that the satellites are not derivative from traditional satellites. Thus, the SSAD model established in Section 5.3.1 is applied to calculate the reliability of each subsystem.

Table 6.16: Inputs of lifecycle cost optimization of RapidEye constellation

Parameters	Symbols	Values
Number of satellites	m	5
Number of subsystems	n	5
System Duplication Vector	P	$\begin{bmatrix} 5 & 5 & 5 & 5 & 5 \end{bmatrix}^{\mathsf{T}}$
System Derivation Matrix	V	$\begin{bmatrix} 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \end{bmatrix}$

The lifecycle costs of the optimal solution of the RapidEye Constellation are decomposed into the costs of different lifecycle phases in Table 6.17. Since the RapidEye satellites are small satellites, the development costs are lower than those of the Galileo FOC satellites. Due to the location in the LEO, the average launch, replenishment, and disposal costs of the RapidEye satellites are significantly saved, compared with the Galileo FOC satellites.

Table 6.17 also compares the cost breakdowns between the optimal configuration and the constellation of identical spacecraft that is the one actually used. The major cost difference is the replenishment costs, while the other costs are similar. The high replenishment costs of the latter are caused by the reliability decrease from the integration of different subsystems.

Similarly, through the backward transformation, the optimal System Configuration

Table 6.17: Breakdown and comparison of different solutions of RapidEye constellation (FY2010$M)

Lifecycle phase	Optimal average	Optimal	Identical
Development	56.1	280.4	279.2
Launch	25.0	124.9	124.5
Operation	18.7	93.5	287.0
Retirement	18.6	92.8	92.8
Overall	118.3	591.6	783.5

Matrix N, seen in Eq. (6.4), can be obtained from the inputs of the System Sequence Vector Q, the System Duplication Vector P, and the System Fractionation Vector F.

$$N = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 \\ 1 & 5 & 0 & 0 & 0 \\ 1 & 0 & 5 & 0 & 0 \\ 1 & 0 & 0 & 5 & 0 \\ 1 & 0 & 0 & 0 & 5 \end{bmatrix} \tag{6.4}$$

The optimal System Configuration Matrix N of the RapidEye constellation can also be interpreted into an intuitive form as shown in Fig. 6.7. The subsystem A to E, namely, PLS, EPS, TTC, CDH, and ACS, are represented by the column 1 to 5 in N, and satellite 1 to 5 are denoted by the row 1 to 5 in N.

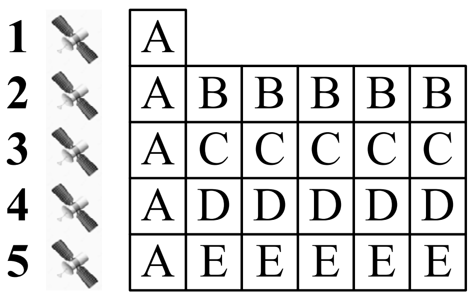


Fig. 6.7: Optimal configuration of RapidEye constellation with minimum lifecycle costs

As can be seen from Fig. 6.7, the optimal constellation configuration can be viewed as separating the infrastructure modules from the payload modules, where subsystem A to E are respectively PLS, EPS, TTC, CDH, and ACS. The effects of such a design are dual: on the one hand, the integration of the same infrastructure modules can improve the reliability of that module; on the other hand, such integration can reduce the cost penalties of the WSS, compared with the completely fractionated design that fractionates all the subsystems.

Similar to the Galileo case, considering the practical constraint of the integration of the PLS and the ACS, a constrained optimization process can be applied to seek a practical optimal solution. The ways of adding the constraints are same with the ones used in the Galileo case. The corresponding optimal configuration design is shown in Fig. 6.8, with the difference of the cost breakdowns between the constrained optimal and the identical solution highlighted in Table 6.18.

Table 6.18: Breakdown and comparison of different constrained solutions of RapidEye constellation (FY2010$M)

Lifecycle phase	Constrained average	Constrained optimal	Identical
Development	56.2	281.0	279.2
Launch	25.0	125.1	124.5
Operation	25.0	125.0	287.0
Retirement	18.6	92.8	92.8
Overall	124.8	624.0	783.5

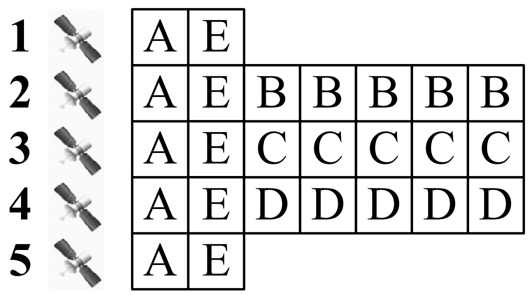


Fig. 6.8: Constrained optimal configuration of RapidEye constellation with minimum lifecycle costs

As can be seen from Table 6.18 that, the constrained optimal configuration design has a higher lifecycle cost than the theoretical optimal design, while it has a lower lifecycle cost than the design of the constellation of identical spacecraft. This indicates that the potential fractionation of some parts of a system can reduce the entire lifecycle costs.

Furthermore, this approach can also be applied for other subsystems such as EPS and TCS, while multiple constraints can be introduced to enable the integration of multiple subsystems into one module.

6.3 Summary

The optimization process of different lifecycle phase costs are implemented, based on the value-centric processes developed in the previous chapter. Typical mission cases have been reviewed and redesigned to illustrate how to minimise different lifecycle phase costs respectively, namely, the development, launch, replenishment, and disposal costs.

Based on the knowledge of different lifecycle phases, the methodology of analysing and optimizing the entire lifecycle costs of a space system has been established. The modelling of the costs of different lifecycle phases is integrated to generate the entire lifecycle costs. It acts as a comprehensive metric for system configuration design to minimise the total costs from development to retirement. It also provides a standardised and comparable value to make design trade-offs between different lifecycle phases.

Two representative mission cases are studied to illustrate the concrete solution process. They are the Galileo FOC and the RapideEye missions. Since the Galileo FOC satellites are located in different orbits, the methodology is applied to optimize the configuration design of individual satellites, while the overall system configuration is a constellation of identical spacecraft. The RapidEye satellites fly in a formation, and therefore the methodology is applied in this case to optimize the overall system configuration. Both cases provide a new insight in designing a space system to achieve the minimum lifecycle costs with the corresponding mission requirements.

Chapter 7

Conclusions

As the continuous advancements in micro- and nanotechnology that has given birth to the miniaturisation of electronic systems, space systems are facing an opportunity of scaling down, with enhanced capabilities and competitiveness. This has enabled the rapid development of small and inexpensive platforms, such as CubeSats, which may stimulate a revolution in space system design and development.

Apart from these technical advancements, emerging design concepts have also promoted a big breakthrough in spacecraft capabilities, some even overturning the conventional wisdom of space design and development. Modularity enables the decoupling of design requirements, which can effectively lower the difficulty in system designs and thus shorten the design cycle. Introducing modularity into space system design can overcome the drawbacks inherent to traditional designs, e.g., subsystem failures, cost variation, and over-complex structures. Such design philosophy embraces the mass production of COTS products, which have been widely-used in small satellites to increase the cost-effectiveness of CubeSats.

Fractionation offers another innovative system configuration for satellites and an ideal carrier for modularity, mass production, and the use of COTS products. The flexibility, robustness, and cost-effectiveness [12] offered by fractionation are believed to fill the vacancies and introduce additional value for space systems.

If these design concepts were appropriately developed, it would probably realise the dream of making satellites more accessible and affordable to develop easily like building blocks. However, the traditional requirement-centric methodologies focus on the design of large, complex, and bespoke space systems.

The associated labour-intensive development and production processes typically spend considerable time and money on the integration and testing phases. This does not naturally fit with the emerging modular, standardised concepts, and the incorporation of mass-produced technologies that could result in reduction of satellite sizes and stimulate the growth of the space market. Therefore, there is a significant potential benefit in establishing and adopting a new design paradigm to effectively solve the problems rooted in the traditional design methodologies and deliver the novel capabilities in both traditional and innovative system designs.

The traditional design problems generally appear in the process of turning system formulation into implementation. Thus, the particular constraints of system designs and the significant effects of different lifecycle phases in the system implementation are necessary to be included in the proposed paradigm. This enables the requirements between different lifecycle phases to be traded off in the overall system level, and further improves the design configuration to achieve the minimum system lifecycle costs.

7.1 Conclusions of the Developed Design Architecture

Aiming at overcoming the drawbacks that are rooted in traditional requirement-centric design methodologies, this research has presented a categorisation, characterisation, and value-centric design architecture to enable the exploration of the effect of emergence of the latest concepts and technologies.

Based upon the categorisation of different configurations, a characterisation of space systems is proposed, comprising of duplication, fractionation, and derivation. The three primary characteristics capture the overall characteristics of system configurations, enabling the promotion of hybrid designs with the potential to improve performance or reduce cost. Complying with the general philosophy of value-centric design, a value-centric design architecture for the design and development of a wide range of space systems is established, acting as a systematic guideline for quantitative system design and analysis.

Under this architecture, the three dimensions of system characterisation are formulated as a design space in accordance with their definitions. The function of the design space is to integrate the cost or intrinsic properties, e.g., mass, reliability, and orbit, from subsystem level to system level, based on system design variables. Applying appropriate value models, these properties can be measured in the singular monetary dimension. Different properties are used for the cost modelling of different lifecycle phases of a space system, e.g., development, launch, operation, and retirement phases. The sum of the costs of these four lifecycle phases is further applied as the objective function for the optimization process of system design configurations to minimise the entire lifecycle costs.

1) **A new categorisation and characterisation have been presented as a methodology to quantify the configuration designs for both traditional and innovative space systems**.

In terms of the features of configuration designs, space systems have been classified into four major categories: monolithic spacecraft, constellations of identical spacecraft, fractionated spacecraft, and hybrid spacecraft. Based upon the categorisation of different system configurations, three primary configuration characteristics, namely, duplication, fractionation, and derivation, have been identified to construct the system characteristic space.

The three primary characteristics reflect different system qualities, thus configuration designs can be effectively explored using the proposed design space. The definitions of the three characteristics are illustrated by a qualitative analysis of a series of typical space missions. Through this space, the connection has been established between system conceptual designs and configuration characteristics. This enables the qualitative analysis of the feasibility and applicability of different system configurations for different mission scopes.

More specifically, a mission concept can be analysed and designed in the system characteristic space, where each dimension represents different configuration characteristics. This system characteristic perspective may provide designers or users with better understanding of the limits and potentials of a space mission, beneficial for further mission modification or extension. Furthermore, this characterisation can also be formulated as a design space for the quantitative analysis of system configurations, under a value-centric design architecture.

2) **A value-centric design architecture based on the system characterisation has been established to enable the analysis and assessment of a wide range of space system configuration designs**.

Based upon the general descriptions of value-centric philosophy, a value-centric design architecture for designing and developing a wide range of space systems has been established, enabling the use of both traditional and innovative design concepts and technologies. Overall, such a design architecture acts as a systematic guideline for the quantitative space system design and analysis.

Under this architecture, the three dimensions of system characterisation are formulated as a design space, complying with the definitions and the qualitative understanding. This enables the establishment of the one-to-one correspondence between system design configurations and the three dimensions of system characterisation. The function of the design space is to integrate the cost or intrinsic properties, e.g., mass, reliability, and orbit, from subsystem level to system level, based on system design variables. This enables the design space to be applied to improve a given property by exploring the effects of different system configurations and subsystem selections, beneficial for the applications of the quantitative analysis and optimization of design configurations.

Through the modelling of the properties of subsystems, the design space has been explored to analyse the effects of different configuration characteristics. The integration of the properties from subsystem to system level has been implemented in the proposed value-centric architecture to establish the mapping relations between design configurations and intrinsic properties. This enables the applications of optimization algorithms to search for the optimal set of design configurations in the system characteristic space, with a certain property as the objective function.

More specifically, the mapping relations have been applied on the properties such as mass, reliability, and orbit. By applying appropriate value models, these properties can be measured in the singular monetary dimension. Different properties can be used for the cost modelling of different lifecycle phases of a space system, e.g., development, launch, operation, and retirement phases. On one hand, each of these costs or their combinations can act as a metric to evaluate the system configuration designs. On the other hand, the metric can be used as the objective function for the optimization process of configuration designs. The concrete analysis and optimization of the cost of different lifecycle phases have been executed in the related chapters. The sum of the costs of these four lifecycle phases can be further applied as the objective function for the optimization process of configuration designs to minimise the lifecycle costs.

The quantification of system values and the adoption of optimization algorithms are two primary advantages, distinguishing the proposed architecture from traditional requirement-centric methodologies. These advantages enable the designers to design, analyse, and compare new and innovative system configurations on a level playing field.

Particularly in the era of "smaller, faster, and cheaper New Space", this design architecture can provide an effective vehicle for the application of modular, standardised, and mass-produced technologies and COTS products. Under the proposed architecture, the design of system configurations can be converted into the design of the combination and permutation of subsystems. The optimization of design configurations can also be realised in the design space, the system characteristic space, by identifying specific system value as the objective function, thus helping to address problems inherent in traditional design methodologies such as capability uncertainty, cost growth, and schedule slippage.

3) **A methodology has been developed under the proposed value-centric architecture to identify improved design configurations and optimize system lifecycle value**.

Under the proposed value-centric design architecture, a methodology for analysing system design configurations and optimizing different lifecycle phase costs or the overall lifecycle costs has been developed. Given the same performance level, the optimization of the lifecycle costs can be converted into that of the lifecycle value for a space system.

Through the use of the mapping relations established, the system properties can be analysed based on the design configurations. Applying appropriate value models, these properties can be measured in the singular monetary dimension. Different properties can be used for the cost modelling of different lifecycle phases of a space system, e.g., development, launch, operation, and retirement phases.

The development costs quantify the costs of developing a space system, with subsystem cost models as inputs. Using the system characteristic space, subsystem costs are integrated into an overall system cost, which measures the efforts made in the development phase. The learning curve factor is considered, when a subsystem is reproduced. By adopting the development costs as the objective function, the optimization of system configurations is realised. The case studies reveal the different effects that monolithic and distributed configurations have on development costs.

The launch costs formulate the costs of launching a space system, with subsystem cost and mass properties as inputs. Real data of various launch vehicle families, such as cost, total attempts, and success rate, are initially collected to develop a launch vehicle database. Based on this database, we estimate the reliability of each launch vehicle family, using a modified two-level Bayesian analysis. The factors of both launch cost and reliability are subsequently merged into the expected launch cost, acting as the comprehensive metric to evaluate the cost of launch activities. Similarly, the optimization process of the launch costs is built, with a series of typical space systems reviewed and redesigned to illustrate how to better control launch costs.

The replenishment costs estimate the running costs of a space system in the operation phase, with subsystem cost and reliability properties as inputs. Since on-orbit servicing is extremely costly and operationally complex, system maintenance can be executed by replacing a failed or degraded satellite with a new one, as is routinely done in the network GPS. The modelling of subsystem reliability, associated with the subsystem cost models in the development phase, are merged into the expected replenishment costs. Based upon the modelling, the effects of different design configurations on the replenishment costs can be analysed. Reversibly, by adopting the replenishment costs as the objective function, the design configurations can be optimized in the system characteristic space. A few mission cases have been studied to illustrate how different system design configurations influence the lifecycle replenishment costs.

Due to the looming space debris problem, the disposal costs assess the costs of the retirement of space systems. The active removal activities are applied for the space systems without end-of-life de-orbit or orbital lifetime reduction devices, or with these devices failed. It is estimated by calculating the costs associated with building and launching a series of spacecraft to send the de-orbit packages to implement disposal activities. To minimise the costs, the change in velocity is optimized for a single orbital transfer between any two targets, while the combination and permutation strategies are developed for missions of multiple targets. Case studies have been conducted on typical space missions, which reveal the significance of pre-design of de-orbit devices.

Overall, the above four costs are integrated or summed into the lifecycle costs, acting as an objective function for space system design and analysis in order to minimise the mission costs. Thus, different system properties or design requirements can be converted into a standardised dimension, solving design selection problems by turning the multi-objective optimization problem into a single-objective optimization.

7.2 Recommendations for Future Work

While the proposed value-centric design architecture has been demonstrated with the abilities to solve the traditional problems and provide a vehicle for both traditional and innovative design concepts and technologies, a number of improvements and further developments have also been identified based on this research. The recommendations on the improvements and the potential opportunities for further developments are discussed in the following.

1) **System Cost Modelling**

As previously presented in Section 3.1, the cost modelling used in this research can only assess the comparative costs according to the system or subsystem masses. More specifically, the CERs of the USCM, the SSCM, and the spacecraft cost fraction are applied to estimate the costs of different subsystems and the entire spacecraft. The standard deviations of these cost models are relatively high, although limited to the cost models publicly available. Thus, more practical design solutions can be reached through the integration of improved cost models into the analysis and optimization process of system design configurations.

The cost modelling can effectively support the cost analysis of this research at the subsystem level. However, more comprehensive cost modelling is an essential element for the further integration of the design of more detailed system configurations. More accurate cost estimates at more detailed system levels are required in this comprehensive cost modelling.

2) **Launch Cost Modelling**

Due to the limited public data of the launch costs, the average costs of launch missions of different families of launch vehicles are used as the cost parameter of single launch attempt in this research. The costs are rough estimates, since different launch missions can have different requirements.

A number of factors can contribute to the variation of launch costs. The difference in payload mass and volume results in the use of different types of launch vehicles in a launch vehicle family. The orbit parameters cause the different masses of propellant to be consumed for a space mission.

A more comprehensive launch cost modelling can be utilised to generate more accurate launch costs, beneficial to the assessment of the costs in the launch, operation, and retirement phases. This can contribute to the demonstration of more practical solutions in the process of analysing and optimizing system configurations to minimise the lifecycle costs.

3) **Combination and Permutation Strategies**

The combination and permutation strategy adopted for the ADR analysis is only practical when the number of objects is less than 15. When more than 15, the corresponding Stirling number of the second kind is over 10^{10}. This causes the low efficiency in labelling all the 10^{10} combinations. Moreover, the process of labelling all the possible permutations of each of the 10^{10} combinations is more severe.

Fortunately, there is a solution in the combination and permutation strategy adopted. Since the numbering and labelling is the function of the number of objects, all the possible combinations and permutations corresponding to the number of objects can be pre-calculated and stored. The stored combinations and permutations can be read, when the global searching or optimization algorithms is applied on the corresponding number of objects.

However, the pre-calculation of the list of all the possible combinations and permutations takes days or months for the situations where the number of objects is greater than 15. It also causes a data storage problem.

Therefore, there is a significant requirement for an improved combination and permutation strategy for the application of global searching or optimization algorithms. This might be achieved by either data pre-processing that decreases the dimension of the objects or improved labelling strategy that reduces the amount and time of calculations.

4) **Detailed Design Process Integration**

In this research, a value-centric methodology for the exploration of the design of space system configurations has been demonstrated from the subsystem level to the system level. Since the modelling, analysis, and optimization of the costs of different lifecycle phases are implemented at the subsystem level, the analysis and optimization of system configurations are realised at the system level. This does not involve the considerations of many other detailed aspects of system configuration design. To increase the available knowledge and understanding in the concept phase, the implementation of the proposed design methodology needs to include and benefit from these detailed design aspects.

The detailed design processes are also the essential components of the proposed design methodology, which have not been fully explored in this research. Moreover, the proposed methodology provides a structure where these processes can be incorporated. Therefore, the implementation of the proposed design methodology incorporating more detailed design processes within the subsystems needs to be investigated.

The cost models at the component or more detailed levels are necessary in the detailed design processes. Applying the cost models of components and subsystems, the configuration designs of different subsystems can be optimized simultaneously with the design of the overall system. This is the merit of value-centric design that keeps multiple trade parameters consistent across all the components, subsystems, and systems.

Consequently, under the overall design architecture, the objective function of the lifecycle costs can be applied to seek the improved or optimal set of design solutions that more closely represent the truth, through the integration of more detailed design processes.

5) **Sensitive Analysis**

It is worth noting that assumptions and estimations, such as cost models and subsystem mass fraction models, are necessary to all the implementation in the presented design architecture. However, it is still unclear how sensitive the output values are to the input ones used and their potential variations, e.g., across the standard deviations of cost models. At least, it is essential to figure out whether the variations representing appropriate levels of uncertainty in the inputs make a significant difference to the outputs.

The outcomes of the analyses will affect the interpretation of the results. On the one hand, more attention should be paid to the input parameters that the outputs are quite sensitive to, since a tiny deviation may have a significant impact on the results. On the other hand, once the results are not very sensitive to the input values under an appropriate uncertainty level, it is very likely to be an optimum system in the case.

Moreover, the sensitive analyses are also beneficial to the accuracy improvement of cost models. This may promote it to be implemented associated with cost modelling, including subsystem or more detailed levels. The sensitive analyses can be conducted at different system levels, based on which the overall analysis across all the levels is enabled. In conclusion, the research of the detailed design process provides the improved solutions more close to the truth, while the overall sensitive analysis estimates the corresponding uncertainty.

References

[1] D. Salt, “NewSpace - Delivering on the Dream,” *Acta Astronautica*, vol. 92, no. 2, pp. 178–186, 2013.

[2] European Space Agency, “What is Space 4.0?,” 2016.

[3] European Space Agency, “Resolution “Towards Space 4.0 for a United Space in Europe” Council Meeting at Ministerial Level,” tech. rep., Lucerne, 2016.

[4] N. Antoni, M. Adriaensen, A. Papadimitriou, C. Giannopapa, and K.-U. Schrogl, “Re-affirming Europe’s Ambitions in Space: Past, Present and Future Perspectives,” *Acta Astronautica*, vol. 151, pp. 772–778, 2018.

[5] A. K. Cebrowski and J. W. Raymond, “Operationally Responsive Space: A New Defense Business Model,” tech. rep., 2005.

[6] L. Doggrell, “Operationally Responsive Space: A Vision for the Future of Military Space,” tech. rep., 2006.

[7] M. G. Richards, L. Viscito, A. M. Ross, and D. E. Hastings, “Distinguishing Attributes for the Operationally Responsive Space Paradigm,” in *AIAA 6th Responsive Space Conference*, (Los Angeles, CA), pp. 1–11, 2008.

[8] A. Poghosyan and A. Golkar, “CubeSat Evolution: Analyzing CubeSat Capabilities for Conducting Science Missions,” *Progress in Aerospace Sciences*, vol. 88, pp. 59–83, 2016.

[9] P. Davison, D. Kellari, E. F. Crawley, and B. G. Cameron, “Communications Satellites: Time Expanded Graph Exploration of a Tradespace of Architectures,” *Acta Astronautica*, vol. 115, pp. 442–451, 2015.

[10] D. Rossetti, B. Keer, J. Panek, B. Reed, F. Cepollina, and R. Ritter, “Spacecraft Modularity for Serviceable Spacecraft,” in *AIAA SPACE 2015 Conference and Exposition*, (Pasadena, CA), pp. 1–12, 2015.

[11] M. Martin and M. Stallard, “Distributed Satellite Missions and Technologies -

The TechSat 21 Program," in *AIAA Space Technology Conference & Exposition*, (Albuquerque NM), pp. 28–30, 1999.

[12] M. D. Graziano, "Overview of Distributed Missions," in *Distributed Space Missions for Earth System Monitoring* (M. D'Errico, ed.), ch. 12, pp. 375–386, 2013.

[13] N. H. Crisp, K. Smith, and P. Hollingsworth, "Launch and Deployment of Distributed Small Satellite Systems," *Acta Astronautica*, vol. 114, pp. 65–78, 2015.

[14] M. T. Hicks and C. Niederstrasser, "Small Sat at 30: Trends, Patterns, and Discoveries," in *30th Annual AIAA/USU Conference on Small Satellites*, (Logan, UT), pp. 1–13, 2016.

[15] A. Shaw and P. Rosher, "Micro Satellites : The Smaller the Satellites , the Bigger the Challenges ?," *Air and Space Law*, vol. 41, no. 4, pp. 311–328, 2016.

[16] C. Mathieu and A. Weigel, "Assessing the Flexibility Provided by Fractionated Spacecraft," in *AIAA SPACE 2005 Conference & Exposition*, (Long Beach, CA), pp. AIAA 2005–6700, 2005.

[17] S. Nag and L. Summerer, "Behaviour Based, Autonomous and Distributed Scatter Manoeuvres for Satellite Swarms," *Acta Astronautica*, vol. 82, no. 1, pp. 95–109, 2013.

[18] A. Shao, E. A. Koltz, and J. R. Wertz, "Quantifying the Cost Reduction Potential for Earth Observation Satellites," in *Proceedings of the 12th Reinventing Space Conference*, (London), pp. 199–210, 2014.

[19] Q. Xu, P. Hollingsworth, K. Smith, and W. Zheng, "Space System Concept Design: A Value-Centric Architecture Based on System Characteristic Space," in *67th International Astronautical Congress (IAC)*, (Guadalajara), pp. 1–12, 2016.

[20] Q. Xu, P. M. Hollingsworth, and K. L. Smith, "Value-Centric Design Architecture Based on Analysis of Space System Characteristics," *Acta Astronautica*, vol. 144, pp. 69–79, 2018.

[21] Q. Xu, M. Zhang, Z. Hao, P. Hollingsworth, and K. Smith, "Small Satellite Launch Opportunity: Statistical Analysis and Trend Forecast," in *67th International Astronautical Congress (IAC)*, (Guadalajara), pp. 1–10, 2016.

[22] Q. Xu, P. M. Hollingsworth, and K. L. Smith, "Space System Launch Cost Analysis : A Value-Centric Architecture Based on System Characteristic Space," in

31st International Symposium on Space Technology and Science, (Matsuyama), pp. 1–8, 2017.

[23] Q. Xu, P. Hollingsworth, and K. Smith, “Launch Cost Analysis and Optimization Based on Analysis of Space System Characteristics,” *Transactions of the Japan Society for Aeronautical and Space Sciences*, pp. 1–9, under review.

[24] Q. Xu, P. Hollingsworth, and K. Smith, “A Value-Centric Design and Certification Architecture for Innovative Space Systems,” in *31st International Symposium on Space Technology and Science*, (Matsuyama), pp. 1–9, 2017.

[25] Q. Xu, P. Hollingsworth, and K. Smith, “A Value-Centric Design and Certification Architecture for Space Systems,” *Transactions of the Japan Society for Aeronautical and Space Sciences*, pp. 1–10, accepted for publication.

[26] J. Bouwmeester and J. Guo, “Survey of Worldwide Pico- and Nanosatellite Missions, Distributions and Subsystem Technology,” *Acta Astronautica*, vol. 67, no. 7-8, pp. 854–862, 2010.

[27] M. Swartwout, “Attack of the CubeSats: A Statistical Look,” in *25th Annual AIAA//USU Conference on Small Satellites*, (Logan, UT), pp. 1–15, 2011.

[28] S. W. Janson, “25 Years of Small Satellites,” in *25th Annual AIAA/USU Conference on Small Satellites*, (Logan, UT), pp. 1–13, 2011.

[29] Satellite Industry Association, “State of the Satellite Industry Report,” tech. rep., Satellite Industry Association, Washington, DC, 2016.

[30] SpaceWorks Enterprises, “2017 Nano/Microsatellite Market Forecast,” tech. rep., SpaceWorks, Atlanta, GA, 2017.

[31] International Astronautical Federation, “23rd IAA Symposium on Small Satellite Missions,” 2016.

[32] Union of Concerned Scientists, “UCS Satellite Database,” 2017.

[33] E. Kahr, O. Montenbruck, K. O’Keefe, S. Skone, J. Urbanek, L. Bradbury, and P. Fenton, “GPS Tracking on a Nanosatellite - The CanX-2 Flight Experience,” in *8th International ESA Conference on Guidance, Navigation & Control Systems*, (Karlovy Vary), pp. 1–13, 2011.

[34] C. Underwood, G. Richardson, and J. Savignol, “SNAP-1: A Low Cost Modular COTS-Based Nano-Satellite - Design, Construction, Launch and Early Operations Phase,” in *15th Annual AIAA/USU Conference on Small Satellites*,

(Logan, UT), pp. 1–7, 2001.

[35] von Karman Institute, "QB50," 2017.

[36] E. Gill, P. Sundaramoorthy, J. Bouwmeester, B. Zandbergen, and R. Reinhard, "Formation Flying within a Constellation of Nano-satellites: The QB50 Mission," *Acta Astronautica*, vol. 82, no. 1, pp. 110–117, 2013.

[37] Planet Labs, "Flock Imaging Constellation," 2018.

[38] L. Johnson, M. Whorton, A. Heaton, R. Pinson, G. Laue, and C. Adams, "NanoSail-D: A Solar Sail Demonstration Mission," *Acta Astronautica*, vol. 68, no. 5-6, pp. 571–575, 2011.

[39] National Aeronautics and Space Administration, "Solar Sail Stunner," 2011.

[40] D. C. Alhorn, J. P. Casas, E. F. Agasid, C. L. Adams, G. Laue, C. Kitts, and S. O'Brien, "NanoSail-D: The Small Satellite That Could!," in *25th Annual AIAA/USU Conference on Small Satellites*, (Logan, UT), pp. 1–15, 2011.

[41] R. L. Staehle, B. Anderson, B. Betts, D. Blaney, C. Chow, L. Friedman, H. Hemmati, D. Jones, A. Klesh, P. Liewer, J. Lazio, M. W.-Y. Lo, P. Mouroulis, N. Murphy, P. J. Pingree, J. Puig-Suari, T. Svitek, A. Williams, and T. Wilson, "Interplanetary CubeSats: Opening the Solar System to a Broad Community at Lower Cost," *Journal of Small Satellites*, pp. 1–30, 2012.

[42] O. Brown and P. Eremenko, "The Value Proposition for Fractionated Space Architectures," in *AIAA SPACE 2006 Conference & Exposition*, (San Jose, CA), pp. 1–22, 2006.

[43] N. Shah and O. Brown, "Fractionated Spacecraft: Changing the Future of Risk and Opportunity for Space System," *High Frontier*, vol. 5, no. 5, pp. 29–36, 2008.

[44] J. Enright, C. Jilla, and D. Miller, "Modularity and Spacecraft Cost," *Journal of Reducing Space Mission Cost*, vol. 1, no. 2, pp. 133–158, 1998.

[45] J. Kingston, "Modularity as an Enabler for a More Efficient Commercial Small Satellite Program," in *17th Annual AIAA/USU Conference on Small Satellites*, (Logan, UT), pp. 1–17, 2003.

[46] J. C. Lyke, "Plug-and-play Satellites," *IEEE Spectrum*, vol. 49, no. 8, pp. 37–42, 2012.

[47] C. Alves and A. Finkelstein, "Challenges in COTS Decision-Making: A Goal-Driven Requirements Engineering Perspective," in *14th International Conference on Software Engineering and Knowledge Engineering*, (Ischia), pp. 789–794, 2002.

[48] C. Soanes and A. Stevenson, *Oxford Dictionary of English.* Oxford: Oxford University Press, 2 ed., 2005.

[49] D. Selva and D. Krejci, "A Survey and Assessment of the Capabilities of Cubesats for Earth Observation," *Acta Astronautica*, vol. 74, pp. 50–68, 2012.

[50] B. Klofas, J. Anderson, and K. Leveque, "A Survey of CubeSat Communication Systems," in *5th Annual CubeSat Developers' Workshop*, pp. 1–36, 2008.

[51] B. Klofas and K. Leveque, "A Survey of CubeSat Communication Systems: 2009-2012," in *10th Annual CubeSat Developers' Workshop*, pp. 1–41, 2013.

[52] G. B. Shaw, D. W. Miller, and D. E. Hastings, "Generalized Characteristics of Communication, Sensing, and Navigation Satellite Systems," *Journal of Spacecraft and Rockets*, vol. 37, no. 6, pp. 801–811, 2000.

[53] O. Brown and P. Eremenko, "Application of Value-Centric Design to Space Architectures : The Case of Fractionated Spacecraft," in *AIAA SPACE 2008 Conference & Exposition*, (San Diego, CA), pp. AIAA–2008–7869, 2008.

[54] J. F. Castet and J. H. Saleh, "On the Concept of Survivability, with Application to Spacecraft and Space-based Networks," *Reliability Engineering and System Safety*, vol. 99, pp. 123–138, 2012.

[55] G. A. Orndorff, B. F. Zink, and J. D. Cosby, "Clustered Architecture For Responsive Space," in *AIAA 5th Responsive Space Conference*, (Los Angeles, CA), pp. 1–14, 2007.

[56] W. Edmonson, J. Chenou, N. Neogi, and H. Herencia-Zapana, "Small Satellite Systems Design Methodology: A Formal and Agile Design Process," *Systems Conference (SysCon), 2014 8th Annual IEEE*, pp. 518–524, 2014.

[57] D. Barker and L. Summer, "Analysis of Near-field Wireless Power Transmission for Fractionated Spacecraft Applications," in *62nd International Astronautical Congress (IAC)*, (Cape Town), pp. 1–8, 2011.

[58] F. Alibay, V. R. Desaraju, J. E. Duda, and J. A. Hoffman, "Fractionated

Robotic Architectures for Planetary Surface Mobility Systems," *Acta Astronautica*, vol. 95, no. 1, pp. 15–29, 2014.

[59] J. Guo, D. Maessen, and E. Gill, "Fractionated Spacecraft: The New Sprout in Distributed Space Systems," in *60th International Astronautical Congress (IAC)*, (Daejeon), pp. 1–11, 2009.

[60] O. Brown, P. Eremenko, and M. Bille, "Fractionated Space Architectures: Tracing the Path to Reality," in *23rd Annual AIAA/USU Conference on Small Satellites*, (Logan, UT), pp. 1–10, 2009.

[61] J. Chu, J. Guo, and E. K. a. Gill, "Fractionated Space Infrastructure for Long-Term Earth Observation Missions," in *IEEE Aerospace Conference*, (Big Sky, MT), pp. 1–9, 2013.

[62] A. Martínez de Aragón, "Future Applications of Micro/Nano-Technologies in Space Systems," *ESA Bulletin*, vol. 85, pp. 65–72, 1996.

[63] T. Mosher and B. Stucker, "Responsive Space Requires Responsive Manufacturing," in *AIAA 2nd Responsive Space Conference*, (Los Angeles, CA), pp. 1–8, 2004.

[64] National Aeronautics and Space Administration, "On-Orbit Satellite Servicing Study," tech. rep., 2010.

[65] National Aeronautics and Space Administration, "James Webb Space Telescope (JWST): Independent Comprehensive Review Panel (ICRP) Final Report," tech. rep., National Aeronautics and Space Administration, Washington, DC, 2010.

[66] United States Government Accountability Office, "Defense Acquisition: Assessments of Selected Weapon Programs," tech. rep., United States Government Accountability Office, Washington, DC, 2015.

[67] Congressional Budget Office, "The Budgetary Implications of NASA's Current Plans for Space Exploration," tech. rep., Congressional Budget Office, Washington, DC, 2009.

[68] R. Pielke, "The Rise and Fall of the Space Shuttle," *American Scientist*, p. 32, 2008.

[69] National Aeronautics and Space Administration, "Cost-Benefit Analysis Used in Support of the Space Shuttle Program," tech. rep., National Aeronautics and Space Administration, Washington, DC, 1972.

[70] National Aeronautics and Space Administration, "Analysis of Cost Estimates for the Space Shuttle and Two Alternate Programs," tech. rep., National Aeronautics and Space Administration, Washington, DC, 1973.

[71] R. Pielke and R. Byerly, "Shuttle Programme Lifetime Cost," *Nature*, vol. 472, p. 38, 2011.

[72] United States Government Accountability Office, "The Content and Uses of Shuttle Cost Estimates," tech. rep., United States Government Accountability Office, Washington, DC, 1993.

[73] European Parliament, "Galileo Note," tech. rep., European Parliament, Brussels, 2007.

[74] British Broadcasting Corporation, "Galileo Price Rises 1.9bn Euros," 2011.

[75] European Commission, "Galileo, Europe's GPS, Opens Up Business Opportunities and Makes Life Easier for Citizens," 2013.

[76] United States Government Accountability Office, "NASA Assessments of Selected Large-Scale Projects," tech. rep., United States Government Accountability Office, Washington, DC, 2009.

[77] British Broadcasting Corporation, "Boeing Hopes 787 Flight Marks End of Troubles," 2009.

[78] National Aeronautics and Space Administration, "Mars Science Laboratory Landing," tech. rep., National Aeronautics and Space Administration, Washington, DC, 2012.

[79] European Commissison, "REGULATION (EC) No 683/2008 OF THE EUROPEAN PARLIAMENT AND OF THE COUNCIL of 9 July 2008 on the Further Implementation of the European Satellite Navigation Programmes (EGNOS and Galileo)," 2008.

[80] European Commission, "Political Go-ahead for Galileo," 2007.

[81] European Commission, "Commission Awards Major Contracts to Make Galileo Operational Early 2014," 2010.

[82] European Space Agency, "Two New Satellites Join The Galileo Constellation," 2015.

[83] British Broadcasting Corporation, "US warns against European satellite system," 2001.

[84] British Broadcasting Corporation, "Galileo Companies Given Deadline," 2007.

[85] National Aeronautics and Space Administration, *NASA Systems Engineering Handbook*. Washington, DC: National Aeronautics and Space Administration, 2007.

[86] J. R. Wertz and W. J. Larson, *Space Mission Analysis and Design*. El Segundo, CA: Microcosm Press, 3 ed., 1999.

[87] J. R. Wertz, D. F. Everett, and J. J. Puschell, *Space Mission Engineering: The New SMAD*. Hawthorne, CA: Microcosm Press, 2011.

[88] B. Fox, K. Brancato, and B. Alkire, "Guidelines and Metrics for Assessing Space System Cost Estimates," tech. rep., RAND Corporation, Santa Monica, CA, 2008.

[89] The Aerospace Corporation, "Small Satellite Cost Model," 2016.

[90] E. E. Burger, N. J. Schuch, O. S. Durao, L. L. Costa, and T. R. C. Stekel, "Small Satellites Current Situation for Access to Space Orbits," in *61st International Astronautical Congress (IAC)*, (Prague), 2010.

[91] A. Webb, A. Bonnema, and J. Paffett, "Launching Nanosats Affordably, Problems and Solutions," in *64th International Astronautical Congress (IAC)*, (Beijing), 2013.

[92] N. Crisp, K. Smith, and P. Hollingsworth, "Small Satellite Launch to LEO: A Review of Current and Future Launch Systems," *Transactions of the Japan Society for Aeronautical and Space Sciences, Aerospace Technology Japan*, vol. 12, pp. 1–9, 2014.

[93] NanoRacks LLC, "Smallsat Deployment," 2016.

[94] National Aeronautics and Space Administration, "NanoRacks CubeSat Deployer," 2016.

[95] National Aeronautics and Space Administration, "Meet Space Station's Small Satellite Launcher Suite," 2016.

[96] J. E. Lumpp, D. M. Erb, T. S. Clements, J. T. Rexroat, and M. D. Johnson, "The CubeLab Standard for Improved Access to the International Space Station," in *2011 Aerospace Conference*, (Big Sky, MT), pp. 1–6, 2011.

[97] British Broadcasting Corporation, "Vega Rocket Set for Maiden Voyage," 2012.

[98] Delft University of Technology, "Small Satellite Projects and Their Cost," 2016.

[99] K. Karuntzos, "United Launch Alliance Rideshare Capabilities for Providing Low-Cost Access to Space," in *IEEE Aerospace Conference Proceedings*, (Big Sky, MT), pp. 1–9, 2015.

[100] N. Crisp, K. Smith, and P. Hollingsworth, "Small Satellite Launch to LEO: A Review of Current and Future Launch Systems," *Transactions of the Japan Society for Aeronautical and Space Sciences*, vol. 12, no. ists29, pp. Tf_39–Tf_47, 2014.

[101] A. d. S. Curiel and G. Webb, "The Changing Launch Solutions for the Small Satellite Sector," in *62nd International Astronautical Congress (IAC)*, (Cape Town), 2011.

[102] A. Zak, "The Dnepr Launcher," 2016.

[103] Chinese Academy of Sciences, "Long March 6 Carrier Rocket to Send 20 Small Satellites into Space," 2016.

[104] The Times of India, "ISRO Creats History, Launches 104 Satellites in One Go," 2017.

[105] Planet Labs, "Planet Labs Homepage," 2017.

[106] H. Heidt, J. Puig-suari, A. S. Moore, S. Nakasuka, and R. J. Twiggs, "CubeSat: A new Generation of Picosatellite for Education and Industry Low-Cost Space Experimentation," in *14th Annual AIAA/USU Conference on Small Satellites*, (Logan, UT), pp. 1–19, 2000.

[107] A. L. Weigel and D. E. Hastings, "Evaluating the Cost and Risk Impacts of Launch Choices," *Journal of Spacecraft and Rockets*, vol. 41, no. 1, pp. 103–110, 2004.

[108] S. D. Guikema and M. E. Pate-Cornell, "Bayesian Analysis of Launch Vehicle Success Rates," *Journal of Spacecraft and Rockets*, vol. 41, no. 1, pp. 93–102, 2004.

[109] M. Zhang, Q. Xu, Q. Tang, and Q. Zhang, "Aggregated Preference Value Analysis on Small Satellite Launch Opportunities," in *31st International Symposium on Space Technology and Science*, (Matsuyama), pp. 1–9, 2017.

[110] M. Zhang, Q. Xu, and Q. Zhang, "Aggregated Preference Value Analysis of Small Satellite Launch Opportunities," *Transactions of the Japan Society for Aeronautical and Space Sciences*, vol. 61, no. 2, pp. 68–78, 2018.

[111] O. Brown, P. Eremenko, and P. D. Collopy, "Value-Centric Design Methodologies for Fractionated Spacecraft: Progress Summary from Phase 1 of the DARPA System F6 Program," in *AIAA SPACE 2009 Conference & Exposition*, (Reston, VA), pp. AIAA 2009–6540, 2009.

[112] M. G. O'Neill and A. L. Weigel, "Assessing Fractionated Spacecraft Value Propositions for Earth Imaging Space Missions," *Journal of Spacecraft and Rockets*, vol. 48, no. 6, pp. 974–986, 2011.

[113] European Space Agency, "How Many Space Debris Objects are Currently in Orbit," 2013.

[114] National Aeronautics and Space Administration, "Orbital Debris Quarterly News," Tech. Rep. 1, 2018.

[115] T. Schildknecht, R. Musci, M. Ploner, S. Preisig, J. de Leon Cruz, and H. Krag, "Optical Observation of Space Debris in the Geostationary Ring," in *Proceedings of the Third European Conference on Space Debris*, (Darmstadt), pp. 89–93, 2001.

[116] J. Huang, W. Hu, Q. Xin, and W. Guo, "A Novel Data Association Scheme for LEO Space Debris Surveillance Based on A Double Fence Radar System," *Advances in Space Research*, vol. 50, no. 11, pp. 1451–1461, 2012.

[117] Inter-Agency Space Debris Coordination Committee, "IADC Space Debris Mitigation Guidelines," tech. rep., 2007.

[118] M. Shan, J. Guo, and E. Gill, "Review and Comparison of Active Space Debris Capturing and Removal Methods," *Progress in Aerospace Sciences*, vol. 80, pp. 18–32, 2016.

[119] J. C. Liou, N. L. Johnson, and N. M. Hill, "Controlling the Growth of Future LEO Debris Populations with Active Debris Removal," *Acta Astronautica*, vol. 66, pp. 648–653, 2010.

[120] C. Bonnal, J.-M. Ruault, and M.-C. Desjean, "Active Debris Removal: Recent Progress and Current Trends," *Acta Astronautica*, vol. 85, pp. 51–60, 2013.

[121] T. Yamamoto, H. Okamoto, and S. Kawamoto, "Cost Analysis of Active Debris Removal Scenarios," in *7th European Conference on Space Debris*, (Darmstadt), pp. 1–15, 2017.

[122] P. D. Collopy, "Economic-Based Distributed Optimal Design," in *AIAA Space*

2001 Conference and Exposition, (Albuquerque, NM), pp. 2001–4675, 2001.

[123] B. Gunston, *The Cambridge Aerospace Dictionary.* Cambridge: Cambridge University Press, 2 ed., 2009.

[124] International Union of Pure and Applied Chemistry, *Compendium of Chemical Terminology.* Oxford: Blackwell Science, 2 ed., 1997.

[125] M. Williamson, *Dictionary of Space Technology.* Bristol: Hilger, 1990.

[126] G. F. Dubos and J. H. Saleh, "Comparative cost and utility analysis of monolith and fractionated spacecraft using failure and replacement Markov models," *Acta Astronautica*, vol. 68, no. 1-2, pp. 172–184, 2011.

[127] National Aeronautics and Space Administration, "Sputnik 1," 2014.

[128] J. C. Cook, D. Agle, and D. Brown, "NASA Spacecraft Embarks on Historic Journey Into Interstellar Space," 2013.

[129] National Aeronautics and Space Administration, "Earth's Reflection in Dawn Spacecraft," 2009.

[130] European Space Agency, "Fact Sheet," 2015.

[131] European Space Agency, "Hubble in Orbit," 2011.

[132] A. Darrin and B. O'Leary, *Handbook of Space Engineering, Archaeology, and Heritage.* Boca Raton: CRC Press, 2009.

[133] E. Kaplan and C. Hegarty, *Understanding GPS: Principles and Applications.* Norwood, MA: Artech House, 2 ed., 2005.

[134] National Aeronautics and Space Administration, "Tracking and Data Relay Satellite (TDRS) Fleet," 2014.

[135] United States Government, "GPS Block IIF Satellite," 2014.

[136] European Space Agency, "30-Satellite Galileo Constellation," 2014.

[137] C. Mathieu and A. Weigel, "Assessing the Flexibility provided by an On-orbit Infrastructure of Fractionated Spacecraft," in *56th International Astronautical Congress (IAC)*, (Fukuoka), 2005.

[138] C. Mathieu and A. Weigel, "Assessing the Fractionated Spacecraft Concept," in *AIAA SPACE 2006 Conference & Exposition*, (San Jose, CA), pp. AIAA 2006–7212, 2006.

[139] A. Salado and R. Nilchiani, "Fractionated Space Systems: Decoupling Conflicting Requirements and Isolating Requirement Change Propagation," in *AIAA*

SPACE 2013 Conference & Exposition, (San Diego, CA), pp. 1–15, 2013.

[140] Defense Advanced Research Projects Agency, "System F6," 2015.

[141] U.S. Naval Observatory, "Current GPS Constellation," 2016.

[142] C. H. Yinger, "Operation and Application of the Global Positioning System," *Crosslink*, vol. 3, no. 2, pp. 12–16, 2002.

[143] National Aeronautics and Space Administration, "SM3A," 2006.

[144] Lockheed Martin, "Hubble Space Telescope Servicing Mission 3A - Media Reference Guide," tech. rep., Lockheed Martin, 1999.

[145] United States Government Accountability Office, "GLOBAL POSITIONING SYSTEM: Significant Challenges in Sustaining and Upgrading Widely Used Capabilities," tech. rep., United States Government Accountability Office, Washington, DC, 2009.

[146] United States Government Accountability Office, "GLOBAL POSITIONING SYSTEM: Challenges in Sustaining and Upgrading Capabilities Persist," tech. rep., United States Government Accountability Office, Washington, DC, 2010.

[147] G. Tyc, J. Tulip, D. Schulten, M. Krischke, and M. Oxfort, "The RapidEye Mission Design," *Acta Astronautica*, vol. 56, no. 1-2, pp. 1–7, 2005.

[148] A. M. Baker, B. Stocker, J. Gebbie, M. Oxfort, G. Tyc, J. Steyn, and N. Hannaford, "RapidEye - A cost-effective Earth Observation Constellation," in *59h International Astronautical Congress (IAC)*, (Glasgow), 2008.

[149] J. Gebbie, P. Davies, A. d. S. Curiel, G. Tyc, L. Boland, and P. Palmer, "Spacecraft Constellation Deployment for the RapidEye Earth Observation System," in *60th International Astronautical Congress (IAC)*, (Daejeon), 2009.

[150] P. D. Collopy and P. M. Hollingsworth, "Value-Driven Design," *Journal of Aircraft*, vol. 48, no. 3, pp. 749–759, 2011.

[151] J. R. Wertz and W. J. Larson, *Reducing Space Mission Cost.* Hawthorne, CA: Microcosm Press, 1996.

[152] N. Lao, T. Mosher, and J. Neff, *Small Satellite Cost Model Version 98 INTRO.* Los Angeles, CA: The Aerospace Corporation, 1998.

[153] Surrey Satellite Technology Ltd, "Surrey Online Shop for Small Satellite Platforms, Subsystems & Payloads," 2016.

[154] G. D. Krebs, "Gunter's Space Page," 2016.

[155] M. Wade, "Encyclopedia Astronautica," 2016.

[156] E. Kyle, "Space Launch Report," 2016.

[157] S. Pillai and A. Papoulis, *Probability, Random Variables and Stochastic Processes*. McGraw-Hill, 2 ed., 2001.

[158] R. L. Scheaffer and J. T. McClave, *Probability and Statistics for Engineers*. Boston: P.W.S.-Kent Publishing Company, 1990.

[159] R. A. Howard, "Decision analysis: Perspectives on Inference, Decision, and Experimentation," *Proceedings of the IEEE*, vol. 58, no. 5, pp. 632–643, 1970.

[160] G. Thompson, *Improving Maintainability and Reliability through Design*. London: Professional Engineering Publishing, 1999.

[161] J. Davidson, *The Reliability of Mechanical Systems*. London: Mechanical Engineering for the Institution of Mechanical Engineers, 1988.

[162] J.-F. Castet and J. H. Saleh, "Satellite Reliability: Statistical Data Analysis and Modeling," *Journal of Spacecraft and Rockets*, vol. 46, no. 5, pp. 1065–1076, 2009.

[163] J.-F. Castet and J. H. Saleh, "Satellite and Satellite Subsystems Reliability: Statistical Data Analysis and Modeling," *Reliability Engineering and System Safety*, vol. 94, no. 11, pp. 1718–1728, 2009.

[164] J. Guo, L. Monas, and E. Gill, "Statistical Analysis and Modelling of Small Satellite Reliability," *Acta Astronautica*, vol. 98, no. 1, pp. 97–110, 2014.

[165] D. A. Vallado and W. D. McClain, *Fundamentals of Astrodynamics and Applications*. Hawthorne, CA: Microcosm Press, 4 ed., 2013.

[166] I. A. Stegun and M. Abramowitz, *Handbook of Mathematical Functions: With Formulas, Graphs, and Mathematical Tables*. Washington, DC: U.S. Dept. of Commerce, National Bureau of Standards, 1972.

[167] S. Kirkpatrick, C. D. Gelatt, and M. P. Vecchi, "Optimization by Simulated Annealing," *Science*, vol. 220, no. 4598, pp. 671–680, 1983.

[168] M. F. Cardoso, R. L. Salcedo, S. Feyo de Azevedo, and D. Barbosa, "A Simulated Annealing Approach to the Solution of MINLP Problems," *Computers & Chemical Engineering*, vol. 21, no. 12, pp. 1349–1364, 1997.

[169] B. V. Babu and R. Angira, "A Differential Evolution Approach for Global Optimization of MINLP Problems," in *Proceedings of 4th Asia-Pacific Conference*

on Simulated Evolution and Learning, (Singapore), pp. 880–884, 2002.

[170] Y. Luo, X. Yuan, and Y. Liu, “An Improved PSO Algorithm for Solving Non-convex NLP/MINLP Problems with Equality Constraints,” *Computers and Chemical Engineering*, vol. 31, no. 3, pp. 153–162, 2007.

[171] K. Deep, K. P. Singh, M. L. Kansal, and C. Mohan, “A Real Coded Genetic Algorithm for Solving Integer and Mixed Integer Optimization Problems,” *Applied Mathematics and Computation*, vol. 212, no. 2, pp. 505–518, 2009.

[172] K. Deep and M. Thakur, “A New Crossover Operator for Real Coded Genetic Algorithms,” *Applied Mathematics and Computation*, vol. 188, no. 1, pp. 895–911, 2007.

[173] K. Deep and M. Thakur, “A New Mutation Operator for Real Coded Genetic Algorithms,” *Applied Mathematics and Computation*, vol. 193, no. 1, pp. 211–230, 2007.

[174] British Broadcasting Corporation, “EU awards Galileo satellite -navigation contracts,” 2010.

[175] International Council on System Engineering, *Systems Engineering Handbook*. San Diego, CA: International Council on System Engineering, 2011.

[176] European Cooperation for Space Standardization Secretariat, “Space Product Assurance - Safety,” tech. rep., European Cooperation for Space Standardization, Noordwijk, 2009.

Appendix A

Review of System Life Cycles

An overview of three useful system life cycles are summarised in Fig. 2.2, which has been presented in our previous research [24]. Overall, the International Council on Systems Engineering (INCOSE) system life cycle is described as a general baseline of various systems. The NASA and ESA system life cycles are exact for space systems, providing the comprehensive understanding of the space engineering and an excellent reference for spacecraft design process.

A.1 INCOSE System Life Cycle

Specific design and certification process varies according to different missions or products, while one of the widely-used generic frameworks is described by INCOSE [175].

In the exploratory research phase, the critical outcome is to acquire a clear understanding of the users' needs, an accurate evaluation of the current technology readiness level, and a rough estimation of the mission cost and schedule. A large amount of creative work is executed to develop new concepts, enabling technologies and required capabilities in this phase, as a starting point of the entire project.

Alternative designs and the corresponding demonstration of the system, subsystems, and key components are identified in the concept phase, associated with the users' expectations and requirements. The selected baseline design of the system, subsystems and key components are further designed, built, verified, and validated throughout the development phase, meanwhile the management strategies of the mission cost and risk are also optimized and refined.

Subsequently, the system-of-interest is manufactured in the production phase, based on the baseline design [175]. Sometimes, product modifications may be applied to resolve manufacture problems, reduce manufacture cost, or improve system capabilities.

Maintenance and support are carried out in the utilisation and support phase, to keep the system offering continuous services in nominal circumstances. Until the retirement phase, the system and the corresponding services are safely removed from operation.

Throughout the development and modification of a system, verification and validation are two critical technical activities performed to examine the satisfaction of system requirements. The purpose of system verification is to comprehensively demonstrate the system capabilities to satisfy all the mission requirements in advance of the production and the utilisation phase. System validation is conducted to confirm the mission requirements and the system implementation as an appropriate solution to the users' questions.

A.2 NASA System Life Cycle

NASA [85] has developed a typical framework for space design and certification, regardless of human flights or robotic missions. Under this framework, two major phases are defined as formulation and implementation with system approval gate in-between, which are further divided into 7 incremental phases shown in Fig. A.1.

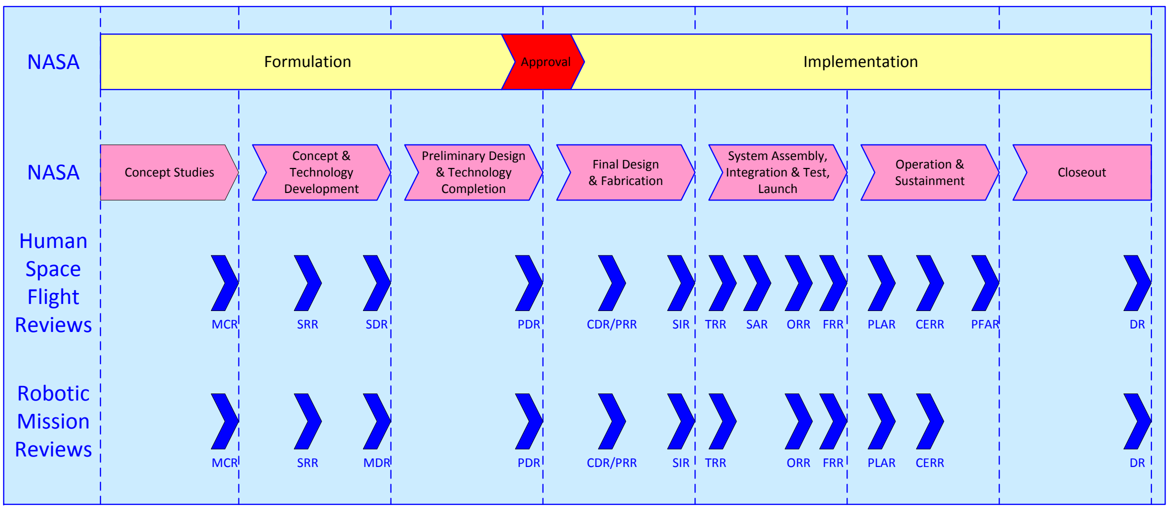


Fig. A.1: NASA system life cycle [85]

Prior to the concept and technology development, various feasible mission concepts are explored as the Mission Concept Review (MCR), as well as the feasibility verification and the programmatic assessment. The subsequent system analysis and design are correspondingly limited by the depth and the scope of these alternative concepts.

In the concept and technology development phase, the activities are performed to completely identify the functional, performance and schedule requirements for the system as the System Requirements Review (SRR), undertake the required technical responsibilities, and make engineering management plans for the project downstream processes, as the System Definition Review (SDR) or the Mission Definition Review (MDR).

During the preliminary design and technology completion phase, the major efforts focus on establishing the preliminary alternatives with the complete functions satisfying mission objectives and requirements, namely, the Preliminary Design Review (PDR). Both technical and engineering readiness of the system design are reviewed, assessed, and improved in the final design and fabrication phase, known as the Critical Design Review (CDR) or the Production Readiness Review (PRR). Before entering into integration, all the preparations are examined in the System Integration Review (SIR).

In the system assembly, integration and test, launch phase, the assembly, integration, testing, and launch activities are implemented for the following four reviews.

1) Test Readiness Review (TRR) confirms the system is ready for testing, and arranges the data acquisition, filtration, and governance.

2) System Acceptance Review (SAR) validates the completeness and maturity level of the entire system to satisfy the mission needs and expectations.

3) Operation Readiness Review (ORR) evaluates both characteristics of the system and the procedures of the project.

4) Flight Readiness Review (FRR) analyses, tests, and verifies the readiness of the system for a successful and reliable flight, as well as subsequent flight operations.

In the operation and sustainment phase, the Post-Launch Assessment Review (PLAR) is performed to observe the status, characteristics, and capabilities of the spacecraft, the Critical Events Readiness Review (CERR) deals with the readiness of a project for critical mission activities, and the Post-Flight Assessment Review (PFAR) identifies and solves all the anomalies appearing in the flight and operation test. Until the closeout phase, the system decommissioning plan is conducted, and the returned data analysis is presented in the Decommissioning Review (DR).

A.3 ESA System Life Cycle

In parallel with NASA, ESA has also defined its own design and certification process [176], consisting of six similar lifecycle phases with NASA's, offering another traditional paradigm of space design and certification.

Initially, the proposed system performance and relevant risk of alternative mission options are identified and assessed, in the mission analysis/needs identification phase. Based on the needs and requirements, the alternative system and operation concepts are explored, and the corresponding the feasibility and the risk level are analysed and evaluated in the feasibility phase to accomplish the mission objectives.

After maximising the lifecycle value and minimising the risk level, system architectures and mission operations are detailed designed in the preliminary definition phase, with further technical requirements and corresponding applicability confirmed. Such system architectures and mission operations are later optimized and confirmed in the detailed definition, production, and qualification testing phase. Simultaneously, the relative technical requirement implementation and the risk level are evaluated to support design modifications, resource allocation, and management strategies.

Throughout the utilisation phase, the mission operations are implemented within the acceptable risk level, and mission data is collected, transferred, and analysed to meet the mission objectives. Near the end of this phase, the disposal plan is made, or an extended mission is proposed, demonstrated, and executed for the disposal phase.