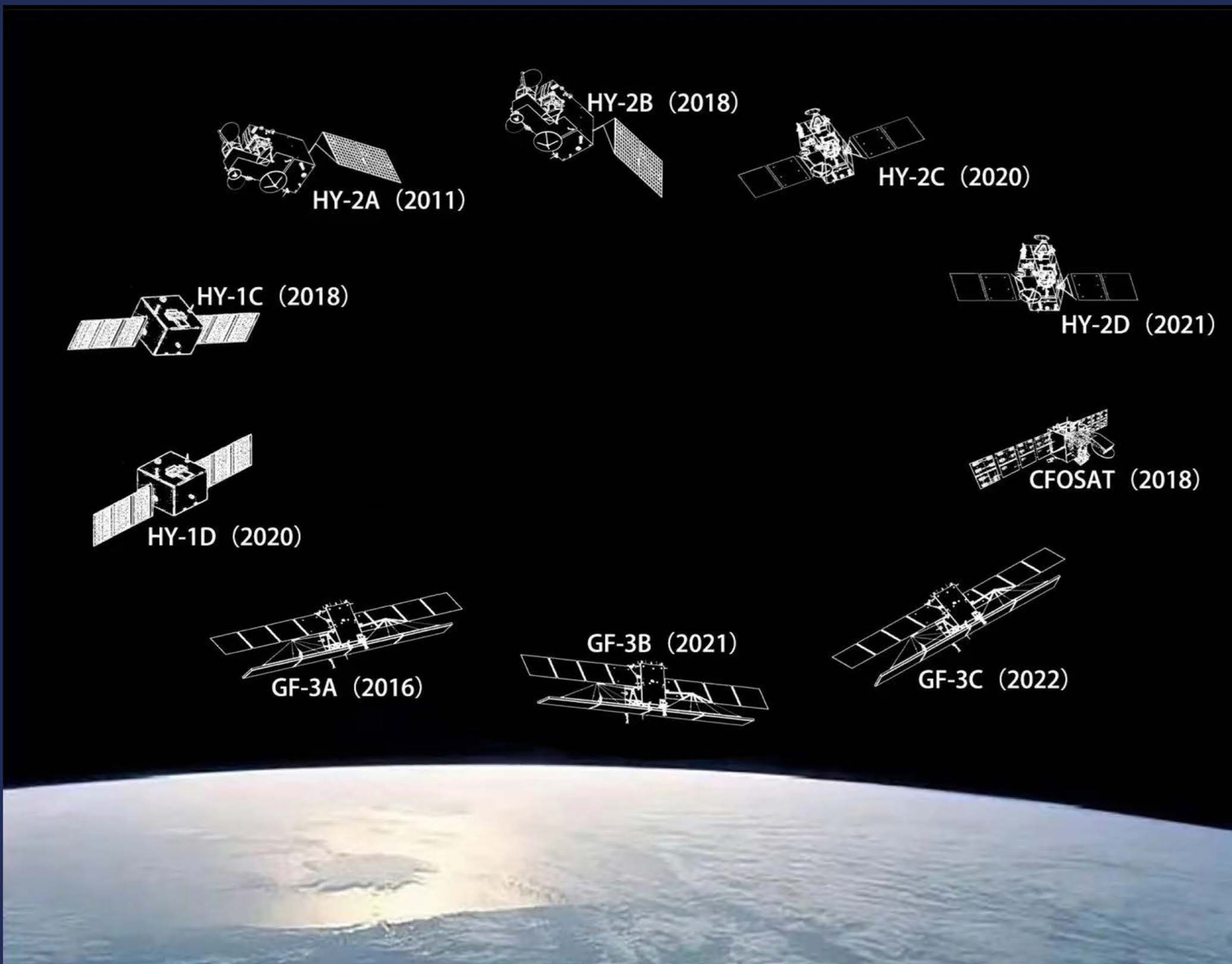
Space Science, Technology and Application Series

SPACECRAFT SYSTEM DESIGN



Edited by **Zhang Qingjun and Liu Jie**

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Spacecraft System Design

Drawing on practical engineering experience and the latest achievements in space technology in China, this book investigates spacecraft system design and introduces several design methods based on the model development process.

A well-established space engineering system with spacecraft as the core is integral to spaceflight activities and missions of entering, exploring, developing and utilizing outer space. This book expounds on the key phases in spacecraft development, including task analysis, overall plan design, external interface, configuration and assembly design and experimental verification. Subsystems that function as the nuclei of spacecraft design and important model development aspects are then examined, such as orbit design, environmental influence factor, reliability design, dynamics analysis, etc. In addition, it also discusses the digital environment and methods to improve the efficiency of system design.

The book will appeal to researchers, students and especially professionals interested in spacecraft system design and space engineering.

Zhang Qingjun is an expert engaged in the field of space remote sensing and currently works at the Institute of Remote Sensing Satellite (IRSS), CAST, China. He was the Chief Engineer and project manager of CBERS, Haiyang-2, Gaofen-3 satellites, etc., and has won several national science and technology awards.

Liu Jie is a researcher at the Beijing Institute of Spacecraft System Engineering (ISSE), a researcher at the China Academy of Space Technology and was the Director System Designer of Haiyang-2 satellite and Gaofen-3 satellite. He has been engaged in the fields of spacecraft system design and satellite remote sensing technology research since 2004.

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Spacecraft System Design

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Contents

Contributors

Tian Baiyi is a researcher at CAST, China. His research interests focus on spacecraft orbit design and analysis of space missions.

Hao Ganggang is a researcher at CAST, China. His research interests include spacecraft system design.

Fan Haitao is a researcher at CAST, China. His research interests include spacecraft digital design.

Shi Haitao is a researcher at CAST, China. His research interests include spacecraft system design.

Zhao Haitao is a researcher at CAST, China. His research interests include spacecraft reliability design.

Zhao Han is a researcher at CAST, China. His research interests include spacecraft digital design.

Xu Hao is a researcher at CAST, China. His research interests include spacecraft system design.

Liu Hongyu is a researcher at CAST, China. His research interests include spacecraft system design.

Wang Jiachun is a researcher at CAST, China. His research interests include spacecraft digital design.

Zu Jiaguo is a researcher at CAST, China. His main research interests include spacecraft system design, spacecraft mechanical design, spacecraft manufacturing and AIT, as well as spacecraft project management.

Zhu Jiantao is a researcher at CAST, China. His research interests focus on design of space environment engineering for spacecrafts.

Liu Jie is a researcher at the Beijing Institute of Spacecraft System Engineering (ISSE), a researcher at the China Academy of Space Technology and was the Director System Designer of Haiyang-2 satellite and Gaofen-3 satellite. He has been engaged in the fields of spacecraft system design and satellite remote sensing technology research since 2004.

Zhou Jing is a researcher at CAST, China. Her research interests focus on spacecraft orbit design and analysis of space missions.

Wang Jingyan is a researcher at CAST, China. Her research interests focus on design and analysis of reliability and safety for spacecrafts.

Liu Jiuli is a researcher at CAST, China. His research interests include spacecraft system design.

Zhang Lei is a researcher at CAST, China. His research interests focus on spacecraft orbit design and analysis of space missions.

Zhang Liang is a researcher at CAST, China. His research interests include spacecraft digital design.

Zhou Liping is a researcher at CAST, China. Her research interests include spacecraft system design.

Zhang Liqiang is a researcher at CAST, China. His research interests include spacecraft digital design.

Geng Liyin is a researcher at CAST, China. His research interests focus on design of space environment engineering for spacecrafts.

Huang Meili is a researcher at CAST, China. She specializes in orbit design and control spacecraft, and analysis of space missions.

Xiao Pengfei is a researcher at CAST, China. His research interests include spacecraft system design.

Yue Qunxing is a researcher at CAST, China. Her research interests include system engineering and project management.

Wang Rui is a researcher at CAST, China. His research interests include spacecraft system design.

Qu Shaojie is a researcher at CAST, China. His research interests focus on design of space environment engineering for spacecrafts.

Zheng Shigui is a researcher at CAST, China. His research interests focus on design of space environment engineering for spacecrafts.

Yu Songbai is a researcher at CAST, China. His research interests include spacecraft system design.

Liu Xia is a researcher at CAST, China. Her research interests include spacecraft digital design.

Xiong Xiao is a researcher at CAST, China. His research interests include spacecraft reliability design.

Li Xiaofei is a researcher at CAST, China. His research interests include spacecraft system design.

Zhang Xiaofeng is a researcher at CAST, China. His research interests include spacecraft system design.

Lv Xinqi is a researcher at CAST, China. Her research interests include spacecraft reliability design.

Cheng Yao is a researcher at CAST, China. Her research interests include spacecraft reliability design.

Yuan Yi is a researcher at CAST, China. His research interests include spacecraft digital design.

Gong Ying is a researcher at CAST, China. Her research interests include spacecraft reliability design.

Zhang Zhaoguo is a researcher at CAST, China. His research interests include spacecraft reliability design.

Xie Zheng is a researcher at CAST, China. His research interests include spacecraft digital design.

Yuan Zhi is a researcher at CAST, China. His research interests include spacecraft system design.

Cheng Zhuo is a researcher at CAST, China. Her research interests include spacecraft reliability design.

CHAPTER 1

Introduction

Yue Qunxing
CAST

CONTENTS

As a general term, space flight refers to various activities entering, exploring, developing, and utilizing space (i.e. the space beyond Earth's atmosphere) and celestial bodies outside the Earth. To perform activities in aerospace and fulfill specific space missions, humans need to establish massive space engineering system with spacecraft as the core based on space technology.

On October 4, 1957, the successful launch of the first man-made earth satellite ushered in a new era of human spaceflight and also marked the beginning of human exploration in the vast universe. The scope of human activities has gradually expanded from land to ocean, from ocean to atmosphere, and from atmosphere to space, indicating a great leap in human ability to understand and transform nature. Over the past 60 years, global space technology has made great progress, which has been widely used in scientific research, military activities, national economy, and social life. Some leading aerospace countries have

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built up a huge and systematic aerospace technology industry. Space activities are rapidly changing the way of human life. The wide application and service of navigation satellites, resource satellites, meteorological satellites, and communication satellites have produced great social and economic benefits.

In this chapter, spacecraft system engineering, composition, and classification of spacecraft systems are introduced, and the significance, general principles, and special requirements of spacecraft system design from a system engineering perspective are presented.

1.1 DEFINITION OF SPACECRAFT SYSTEM ENGINEERING

1.1.1 System Engineering

In 1978, an article *Technique of Organizational Management – System Engineering*, co-authored by Chinese scientists Qian Xuesen, Xu Guozhi, and Wang Shouyun, was published in *Wenhui* Newspaper, which expounded the concepts of system, system engineering, and engineering/project. It points out that a system is an organic whole with specific functions, which is composed of several interacting and interdependent components, while system engineering is a scientific method to organize and manage systems in planning, researching, designing, manufacturing, testing, and application, which has a universal significance for all systems. Under the guidance of decomposition-integration, system engineering studies the overall systematic problem by using the iterative process of analysis, synthesis, testing, and evaluation. System engineering combines optical, mechanical, thermal, electrical, communication, reliability, management, and other professional technologies to conduct system requirements analysis, scheme design, manufacturing and assembly, verification and use, etc. The goal is to develop a comprehensive and optimal system that meets the requirements of the system's entire lifecycle through the two parallel optimizations of system engineering technology and system engineering management.

System engineering is the application of system thinking in the field of engineering. In the modern science and technology system proposed by Qian Xuesen, system engineering belongs to the engineering application technology of system science, and its technical science is mainly based on general operations research, cybernetics, and information theory. According to the application field, system engineering can be divided into project system engineering, agricultural system engineering, military system engineering, economic system engineering, social system engineering, etc. Spacecraft system engineering belongs to the scope of project system engineering.

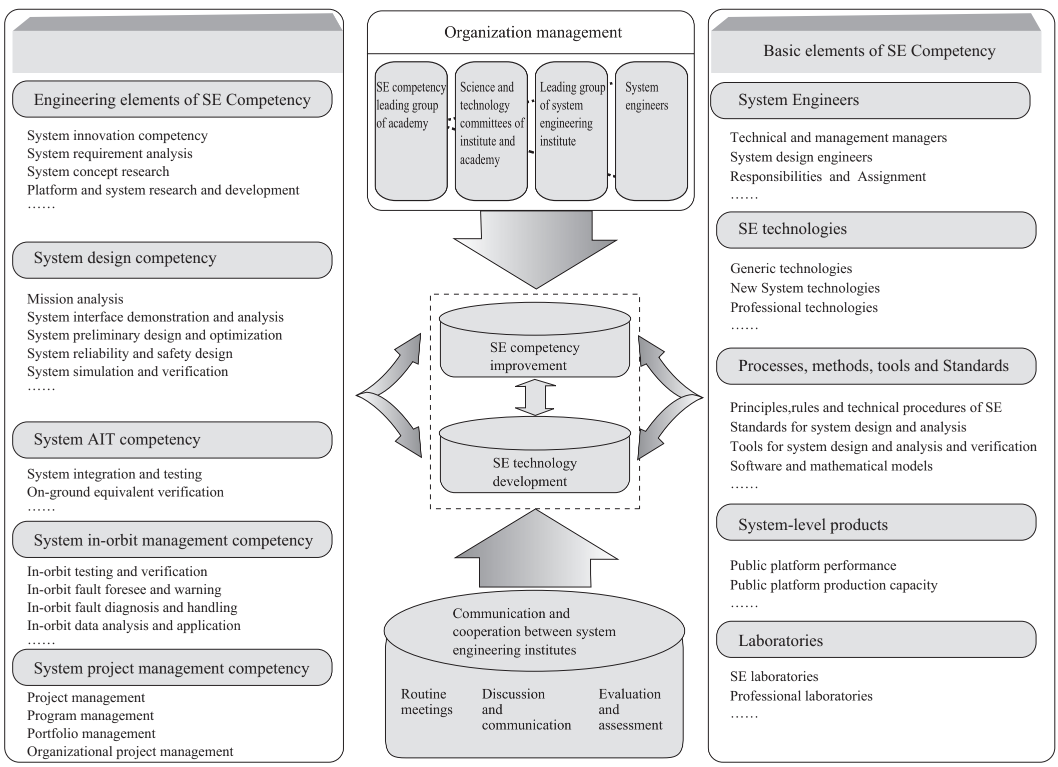
1.1.2 Spacecraft System Engineering

The purpose of spacecraft system engineering is to develop spacecraft systems with the best overall performance, meeting the life-cycle use requirements through the process of system planning, research, design, manufacture, test, and use by comprehensively using the theories and methods of spacecraft engineering technology and system science.

From the perspective of engineering process, spacecraft system engineering comprises system requirements demonstration, system research and development, system design, system integration and verification, system on-orbit management, etc. In addition, the fundamental elements of a spacecraft system engineering include five basic elements: system engineering team; system engineering technology; the processes, methods, tools, and standard specifications; the system-level products; and the laboratory foundation.

China Space System Engineering, founded by Mr. Qian Xuesen, has developed a set of Chinese-characterized space engineering system and methods in practice. It mainly includes the following five aspects:

1. Strengthen the construction of the system engineering department, overall plan design, technical management, and technical coordination.
2. Give full play to the two-line command management mode [one headed by the chief designer, and the other by the chief executive] to guarantee the smooth implementation of the aerospace model engineering.
3. Strengthen the spacecraft design and production organization system based on the system engineering academy.
4. Strictly follow the scientific research procedures and control the configuration at each stage to avoid big iteration.
5. Always adhere to the "Quality First" policy, which is an indispensable requirement to realize the goal of space system engineering (Figure 1.1).



Framework of the spacecraft system engineering of a certain system engineering unit of China Aerospace.

FIGURE 1.1 Framework of the spacecraft system engineering of a certain system engineering unit of China Aerospace.

1.2 COMPOSITION AND PROJECT RELATIONSHIP OF SPACECRAFT ENGINEERING SYSTEM

1.2.1 Spacecraft System

Spacecraft refers to the vehicles that perform specific tasks such as exploring, exploiting, and utilizing space and celestial bodies other than the Earth, also known as space vehicles.

1.2.1.1 Spacecraft Classification

Spacecraft can be classified into unmanned spacecraft and manned spacecraft according to whether they are manned, and civil spacecraft and military spacecraft according to utilization; based on fields, it can be divided into manned field, remote sensing field, navigation field, communication field, space science field, deep-space exploration field, etc. Different types of spacecrafts have different supporting systems. The specific classification is shown in Figure 1.2.

1.2.1.1.1 Unmanned Spacecraft Unmanned spacecraft orbiting the earth is referred to as artificial earth satellites, among which the man-made earth satellite features the largest number of launches and the most versatile, accounting for more than 90% of all spacecraft launches. Artificial earth satellites, according to utilization, can be categorized into scientific satellites, technological test satellites, and application satellites.

Scientific satellites are satellites for stellar and planetary observation, field and matter detection, mainly including near-Earth space physics exploration satellites, astronomical satellites, which can be used for scientific research on solar-terrestrial space, cosmic evolution, and cosmic physical fields; technical test satellites are satellites for principle or engineering tests of space technology and space application technology. Generally, new technologies, new principles, new programs, new instruments, and devices of spacecraft are used on high-value, commercial satellites only after they passed on-orbit verifications. Some tests, such as rendezvous and docking of cooperative objects, are conducted with entire spacecraft. Application satellites are man-made earth satellites that directly various fields of human activities such as national economy, military, and culture. Among all kinds of

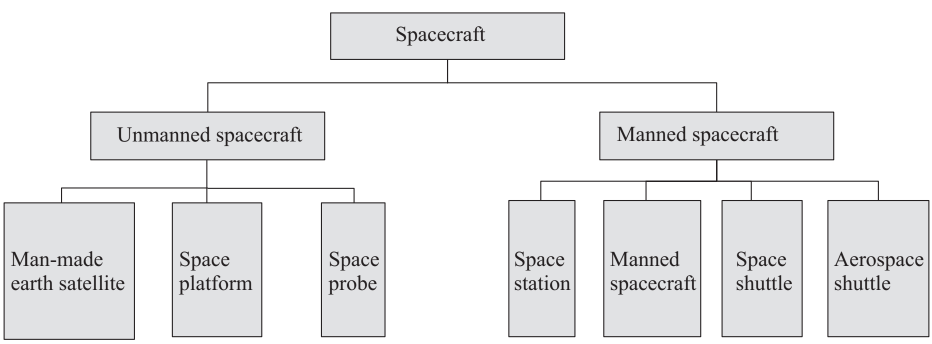


FIGURE 1.2 Spacecraft classification.

man-made earth satellites, application satellites have the largest lunch number and types, which, according to use, can be divided into navigation satellites, remote sensing satellites, communication satellites, etc. Application satellites are the most launched and the most varied among all kinds of artificial earth satellites, which can be divided into navigation satellites, remote sensing satellites, communication satellites, etc., according to their uses.

Current technologies for in-orbit servicing, refueling, resupplying consumables, or recovering various equipment or items in space are emerging, expanding the on-orbit functions of spacecraft, and extending the lifespan of the spacecraft.

1.2.1.1.2 Manned Spacecraft According to the mission, manned spacecrafts are classified into space stations and launch vehicles. The latter include manned vehicles and cargo spaceships; manned vehicles include manned spacecraft, space shuttles, and aerospace shuttles.

The manned spacecraft provides support for the astronauts to perform space missions in space and enables them to return to the spacecraft that landed on the ground. As a round-trip transport vehicle, manned spacecraft can be divided into near-Earth orbit manned spacecraft (to transport astronauts and cargo to and from the space station), lunar manned spacecraft, and planetary manned spacecraft according to their applications. China's Shenzhou series spacecraft are manned spacecraft in low-Earth orbit, while the American Apollo series spacecraft are manned spacecraft to the moon.

Space station is a manned spacecraft that provides necessary test or living conditions for astronauts to live and work and can perform a long-term operation on orbit.

A space shuttle is a space vehicle that carries people or cargoes traveling between the ground and the outer space, part of which can be reused multiple times. Only the United States and the Soviet Union have developed and launched space shuttles.

Aerospace shuttle is a new generation of multiple reusable spacecraft that combines technologies of aeronautics and astronautics, vehicle, and spacecraft. In addition to carrying a rocket engine, it uses aero-engine to work with air as an oxidant when launching, ascending, and re-entering the atmosphere.

1.2.1.2 Composition of Spacecraft

A spacecraft consists of several subsystems with different functions. In general, a spacecraft can be divided into two parts: payload and platform, as shown in Figure 1.3.

1.2.1.2.1 Spacecraft Platform Spacecraft platform is a basic part of the satellite, providing support and service (or guarantee) for payloads. It can support one or a combination of several payloads. The spacecraft platform can provide installation and support, working power, attitude and orbit control, condition monitoring and working mode control, thermal environment protection, information data management, and other services for the payload. Spacecraft platforms generally contain several subsystems such as structure and mechanism subsystem, thermal control subsystem, power subsystem, measurement and control data management subsystem, and attitude and orbit control subsystem.

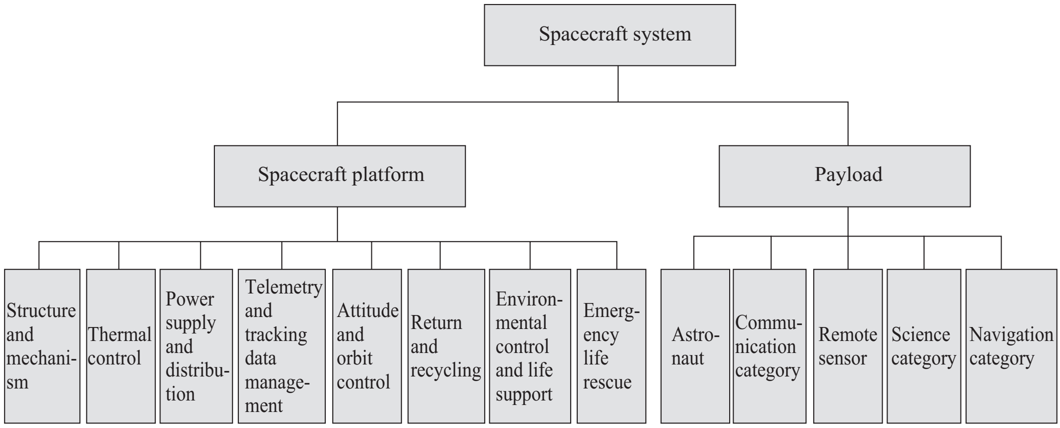


FIGURE 1.3 Composition of spacecraft subsystem.

The structure and mechanism subsystem includes structure subsystem and mechanism subsystem. The former is designed to provide an overall configuration for the spacecraft and support the on-board devices. During the course of the launch of the launch vehicle and the on-orbit maneuvering, it will support the entire spacecraft, transmitting loads to guarantee sufficient strength and stiffness. The mechanism subsystem enables the spacecraft or one of its components to complete the prescribed motion and make sure they are in a required operating state or position. A mechanism subsystem consists of unfolding and locking mechanism, separation and retraction mechanism, drive mechanism, rendezvous and docking mechanism, etc.

The task of the thermal control subsystem is to control the temperature of the on-board instruments and devices as well as the components of the satellite body during the flight of the spacecraft, to ensure that the operating temperature at each on-orbit stage is within the required range, thus a normal on-orbit operation can be guaranteed.

A power subsystem (power supply and distribution subsystem) provides power for the satellite during its on-orbit lifecycle (including sunlight and eclipse season). Due to the long working life of spacecraft in orbit, most of them use long-term power supply. A power subsystem should have functions such as power generation, energy storage, distribution, bus-voltage regulation, and battery charge and discharge control. In some cases, secondary power supply capable of transforming and stabilizing a variety of voltages is also required.

The TT&C data management subsystem performs satellite telemetry, remote control, orbit tracking and measurement, and data management with the cooperation of other subsystems and the ground station. The task of telemetry is to measure the operating status, engineering parameters, environmental parameters, and related data of the instruments and devices of the system relative to the satellite. The remote control is to send commands from the ground to control the operating status of the relevant system's instruments and devices and to infuse data or programs into the satellite. The orbit tracking and measurement refer to the course that radio waves are transmitted from the ground station and then sent back by the on-board transponder, during which the speed, distance, and angle of the satellite's motion can be measured according to the radio wave transmission characteristics, and finally the satellite orbital parameters are obtained through calculation. Data management refers to the comprehensive management of on-board data using on-board computers.

The function of attitude and orbit control subsystem (hereafter referred to as attitude-orbit-control subsystem) is to maintain or change the spacecraft's attitude and orbit during operation. To accomplish its special mission, each spacecraft has a specific nominal orbit and an expected attitude. However, due to launch errors, the attitude-orbit-control subsystem needs attitude adjustment or orbital maneuvers; during orbit operation, due to the interference force/torque of the external environment and that of the internal electromechanical components, the spacecraft will deviate from the nominal orbit and the expected attitude, in this case, the attitude-orbit-control subsystem will be in charge of attitude and orbit maintenance.

In addition, a recovery subsystem as well as an environmental control and life support system and an emergency rescue subsystem are also included in a recoverable spacecraft.

1.2.1.2.2 Payload The payload, which is the most critical subsystem for a spacecraft to accomplish its mission in orbit, refers to those directly loaded on the spacecraft, including instruments, devices, personnel, experimental organisms, samples, etc.

The payload refers to the instruments, equipment, personnel, experimental organisms, and samples that are loaded on the spacecraft to directly accomplish a specific space mission, and it is the most critical subsystem for the spacecraft to accomplish the space mission in orbit.

As the core of spacecraft, payloads are manifold and vary with their missions. It can be roughly classified into remote sensing (or information acquisition), communication (or information transmission), navigation (or information reference), science, and others. Even the same type of payloads could be different in performance.

Remote sensing payload refers to remote sensors for Earth observation, including visible light remote sensors, multispectral scanners, infrared remote sensors, microwave radiometers, synthetic aperture radars, and microwave scatterometers. These remote sensors can acquire various information about Earth's surface (water surface), atmosphere, space, etc.

Communication payloads refer to transponders and antennas, which can be used for satellite-to-Earth satellite communications and play a vital role in space activities.

The navigation payloads are various instruments and devices that provide space and time references. These payloads can be used for satellite navigation, such as highly stable atomic clocks, radio beacon machines.

The scientific exploration payloads are the instruments and devices that are used for space environment exploration, astronomical observation, and space science experiments, including X-ray telescope spectrometer, solar optical telescope, ion mass spectrometer, X-ray spectrometer, and all kinds of devices for measuring and monitoring space environments.

Other payloads mainly include new technology test payloads and special payloads. The former refers to some new spacecraft, subsystems, instruments, devices, and components. They are launched in a certain orbit by special satellite conducting new technology test to verify the theories, scheme, feasibility, compatibility, reliability, etc. Special payloads refer to non-technical payloads, such as space tourism (the payloads are tourists) and space souvenirs (the payloads are envelopes, flags, etc.).

1.2.2 Space Engineering System

The upper level of a spacecraft system is space engineering system. To perform its functions, the spacecraft must be launched by a vehicle at the launch site first, and then the ground-based TT&C system will perform TT&C on the launch vehicle and the spacecraft, making sure the spacecraft enters a predetermined orbit. Next, the attitude and the angle of orbit will be adjusted so that the spacecraft can carry out its task (except for recoverable satellites and scientific experiment satellites) with the cooperation of the ground application system. Then, the functional role of the spacecraft is fully utilized only when the operation is carried out with the cooperation of ground application system. Therefore, in addition to spacecraft system, the space engineering system also consists of the carrier system that pushes the spacecraft into orbit, the launch site that is used for the final assembly, testing, refueling and launch for launch vehicles and the spacecraft before launch, the ground-based TT&C system (TT&C center, TT&C station, TT&C ship, etc.) for the telemetry and tracking of launch vehicles and the spacecraft, the ground application system that cooperates with the on-orbit spacecraft to perform predetermined specific functions, etc. For deep-space exploration and scientific research probes, space engineering system, in comparison to an artificial earth satellite, have an extra application system for afterward scientific research but one less ground-based real-time application capable of continuous operating. For manned spacecraft, the space system has several more systems than the man-made earth satellite engineering system, such as astronauts system, escape and rescue system, and landing site system.

The spacecraft must operate in coordination with the carrier, launch site and recovery facilities, the ground-based TT&C systems, and the ground application system to jointly complete its space mission. The spacecraft is the main component and is the core of the space engineering system. Figure 1.4 shows a typical satellite engineering system.

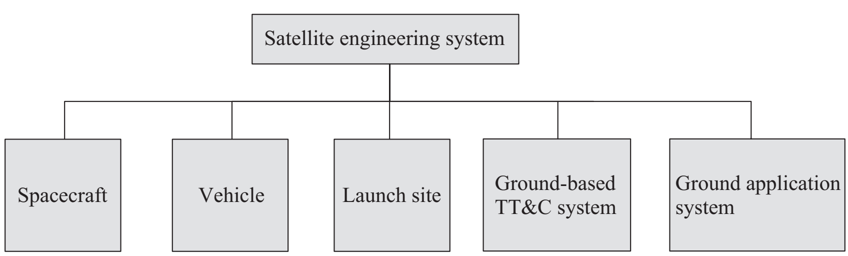


FIGURE 1.4 Composition of satellite engineering system.

All systems in the aerospace engineering system other than the spacecraft are external environmental systems of the spacecraft system. Therefore, when designing spacecraft systems, the space engineering system other than the spacecraft should be used as constraints of environment design.

1.3 CONCEPT OF SPACECRAFT SYSTEM DESIGN

1.3.1 Definition of Spacecraft System Design

A complex engineering system is composed of several subsystems with different functions. Since the entire engineering system is designed by different specialists, a team to design the overall plan first according to the requirements of the system tasks is required. According to the overall scheme, the requirements are decomposed, and the design requirements are put forward to the designers of each subsystem, so that the detailed design of the subsystem can be carried out. Finally, through the overall comprehensive design, each subsystem is integrated into a system that can meet the task requirements. Therefore, the overall design team must ensure the correlation, interaction, and coordination between subsystems, and correspondingly, the integrated system is an engineering system that meets the task requirements and is optimized. This indicates that system design acts as a “guideline” during the development of a complex engineering system.

Spacecraft system design is based on mission requirement analysis, coordinating and integrating the requirements and constraints of each system, adopting system engineering methods, effectively organizing the spacecraft platform and payloads, and coordinating the overall system design to form a satisfactory and optimized system. The core of the spacecraft system design is to provide optimal spacecraft solutions and products under the constraints of prescribed requirements, time, and expenses. Spacecraft system design is the core of spacecraft system engineering.

1.3.1.1 Design Features of Spacecraft System

Spacecraft system design is characterized by hierarchy, systematicness, process-oriented, and creativity.

1.3.1.1.1 Hierarchy A spacecraft is composed of multiple subsystems with different functions and performances, and the subsystems include various instruments and devices. That is the hierarchy of the spacecraft system. The hierarchy of the system design is embodied in the overall design of the system-level design according to the system level, without interfering with the design of large systems or subsystems.[1]

1.3.1.1.2 Systematicness The overall design of spacecraft should not only meet the requirements of the specific mission of the whole spacecraft but also ensure that the subsystems are interrelated, interacted, restricted, and coordinated.

1.3.1.1.3 Process-Oriented The design of spacecraft must be carried out according to the development process. A typical spacecraft development procedure can be divided into three phases, i.e. scheme design, prototype development, and flight model development. The corresponding product development processes are scheme design and prototype tests, detailed design and appraisal tests, as well as the development and test of the flight model. The process can be eliminated on a case-by-case basis, but, in principle, the order cannot be reversed.

1.3.1.1.4 Creativity The goal of spacecraft system design is to obtain an optimized "new" spacecraft system that meets the specific mission requirements.

1.3.2 Principles of Spacecraft System Design

1.3.2.1 Principles of Spacecraft System Design[2]

1.3.2.1.1 Satisfy Mission Requirements Spacecraft system design must follow the user's needs or the particular needs of the country. The final spacecraft developed must be in line with the requirements put forward by the user, that is, in addition to functions and performance indicators, the lead time and development cost requirements must be satisfied. If it cannot meet the user's requirements, it is necessary to coordinate with the user in time for adjustment.

1.3.2.1.2 Ensure the Whole System Is Optimal Spacecraft is a complex system, which is formed by the combination of related components (subsystems or independent components) (mutual correlation, interaction, and coordination). The overall function and performance of the system are not available in its various components, and it is not a simple superposition of the functions and performance of these components. Spacecraft system design should proceed from the overall function and performance of the spacecraft system or even the large aerospace system, handle the relationship between the local and the global, and prevent the pursuit of local high performance or partial low performance from the overall function and performance.

1.3.2.1.3 Standardize the Procedure The development of spacecraft systems must conform to the phased and systematic rules; the procedures must be planned according to the characteristics of each spacecraft project and cannot be reversed during the operation.

1.3.2.1.4 Pay Equal Attention to Inheritance and Innovation Innovation refers to the comprehensive utilization of existing and new technologies, processes, materials, etc. to develop new spacecraft systems and ultimately achieve an optimized and "new" spacecraft system and spacecraft orbit or constellation in space that meets the user's specific mission requirements at the least cost. Spacecraft system design emphasizes both innovation to continuously improve its performance and expand into new application fields and inheritance of the existing technologies to ensure the success of development. As a general requirement, the inherited technologies and products applied on a newly developed spacecraft should reach more than 70%.

1.3.2.1.5 Aim at High Efficiency and Low Risks The pre-design work accounts for about 15% of the cost of the system's entire lifecycle, but it determines 85% of the system's full life cycle cost. Therefore, the cost of the system is designable.

To obtain the maximum benefits, the spacecraft system is required to optimize the design and most effectively apply the existing mature technologies and products, rationally simplify the technical process, shorten the development cycle, and reduce the development cost through optimizing the design and applying the existing mature technologies and products in the highest efficiency manner. Only in this way, the spacecraft can be developed at the minimum cost and meet the requirements of the overall function and performance of the spacecraft, thus ensuring the best function of the system.

1.3.2.2 Special Requirements on Spacecraft System Design

Compared with the design of other engineering projects (vehicles, aircraft, missiles, and rockets), the following special requirements are raised in the design of spacecraft system:

1.3.2.2.1 Space Environment Adaptability[3] In the system engineering design, spacecraft should be capable of adapting various environment conditions, such as high and low temperatures, atomic oxygen, vacuum, solar electromagnetic radiation (heat, light, ultraviolet, etc.), space particle radiation (electrons, protons, cosmic rays, etc.), and mechanics in the launching phase. In addition to withstanding the above-mentioned environment and temperatures, the designed instrument and devices (especially electronic components) of each subsystem must have reasonable shapes and configurations to ensure the spacecraft has good heat-dissipating surfaces and environmental protection measures such as space radiation protection.

1.3.2.2.2 Long Service Life and High Reliability

Because of the high cost of spacecraft development and launch, harsh operating environment, and non-repairable or limited maintenance after launch, the design of the spacecraft system must ensure performance requirements such as long life and high reliability.

1.3.2.2.3 Satisfying the Constraints from other Systems

The spacecraft system can only perform its functions under the cooperation of other systems in the engineering system. Therefore, the spacecraft system must meet the constraints from other systems in the engineering system, including launch vehicle constraints, launch site constraints, ground-based telemetry and tracking ships/stations constraints, and ground application system constraints, etc.

Launch vehicle restraints: Spacecraft is the payload of the launch vehicle. The designed parameters of the spacecraft, such as separation mass (weight), selected orbital parameters, the azimuth during launch, and the attitude accuracy and orbit accuracy during separation, are constrained by the capability of the launch vehicle. The maximum envelope size of the spacecraft is constrained by the effective space of the rocket fairing (the shape of the returnable satellite without fairing must meet the requirements of the aerodynamic characteristics of the launch vehicle); the overall longitudinal and lateral stiffness of the spacecraft must be no less than the launch vehicle's requirements; the deviation of the centroid and geometry center of the spacecraft must meet the launch vehicle's requirements; the spacecraft as a whole and the instruments and devices of subsystems as well as their installation and connection strength must be capable of withstanding the mechanical environments without damage. Such environments include overload, vibration, noise, and impact, which are generated by the launch vehicle during take-off, flight, and separation; the mechanical, electrical, and thermal interfaces between a spacecraft and its launch vehicle must be designed in detail and be well fitted, and the spacecraft and launch vehicle should also meet the requirements for electromagnetic compatibility.

Launch site constraints: The launch site is used for the final assembly, testing, refueling, and launch of the launch vehicle and the spacecraft before launch. It consists of technical zone, launch zone, and related parts. In the technical zone, spacecraft has specific requirements on the size of the plant area, air environment (temperature, humidity, cleanliness, etc.), hoisting devices, power supply, propellant supply, communication devices, electromagnetic environment, safety facilities (grounding, lightning protection, explosion-proof and fire protection), etc.; in launch zone, spacecraft has specific requirements on hoisting, testing, communication, tower, electromagnetic environment, meteorology, etc. In addition, the geographical location of the launch site and the launch direction restrictions of the launch vehicle (related to the carrying capacity and rocket landing point restrictions) are also the constraints that must be considered in the spacecraft system engineering design.

Ground-based TT&C ship/station restraints: The constraints of the ground-based TT&C ship/station include the constraints of the ground-based TT&C ship/station on the TT&C frequency band and the TT&C system of the spacecraft and that of the geographical location of the ground-based TT&C ship/station on the control arc, etc. The radio information transmission interface between the ground-based TT&C ship/station and the spacecraft, mainly including radio frequency, transmission power, antenna pattern and gain, the sensitivity of spacecraft TT&C devices, TT&C system, TT&C procedures, TT&C requirements, modulation methods, data processing, and encryption and decryption, must be regarded as the constraints of both parties after the conformation of investigations on design and existing conditions. Those conditions should be recorded in documents and followed by the two parties.

Ground application system constraints: Ground application system constraints include the technical requirements for the use of spacecraft and the radio interface requirements between the spacecraft and ground application systems. The real-time or delayed useful information delivery between an application satellite (except for recoverable satellites) and ground application systems is realized via radio. For instance, by taking use of radios, communication satellites transmit communication information for ground communication stations, navigation satellites conduct ranging and timing to determine the user's location; earth observing satellites use data transmission systems to send remote sensing information to the ground in real time or with a delay.

As a result, the radio frequency band, communication system, information transmission and performance indicators of spacecraft, and ground application systems are important constraints.

1.3.2.2.4 Taking Safety and Risk Control as the Top Priority Since spacecraft system contains flammable and explosive propellants and igniters, and the margins of design conditions are small, risks should be fully considered.

Safety must be ensured through safety design and safety management during the development process. In the safety design, the main load-bearing members are generally required to have sufficient stiffness and strength to have a certain safety margin; the propellant tank should be designed with a certain safety margin; the design of instrument circuit and cable network of spacecraft system should meet the requirements of electromagnetic compatibility; initiating explosives should be safe and insensitive; initiating device manager must be designed with multiple safety insurance; circuit welding, debugging, and various electrical tests must have safety protection measures; flammable and explosive propellant filling must have safety and fire protection measures.

Technical risk, cost risk, and schedule risk should be considered. In particular, the overall plan design must be thoughtful and meticulous to avoid large-scale rework. During the development process, various quality accidents must be avoided to reduce economic risks and time schedule risks. Special risk analysis and effective measures must be adopted to reduce the risk to a minimum or acceptable level.

1.3.2.2.5 Laying Emphasis on Fault Diagnosis and Autonomous Control Functions After the spacecraft enters its orbit, simply autonomous control is necessary for the following actions during the long-term on-orbit operation period: spacecraft attitude TT&C, the switching of the instrument and device backup parts of each subsystem, the charging/discharging of the battery, the shunt control of the residual current of the power supply, the fully regulated bus-voltage control, the power-on and power-off control of the electric heater of the thermal control subsystem, etc. In addition to the simple autonomous control functions described above, modern spacecraft can autonomously implement the measurement and control of their orbital position during orbiting operations. That is, even without ground support, it can eliminate all kinds of interference and adjust its orbit and attitude to a normal state through autonomous guidance, navigation, and control technology (intelligent control), so as to reduce the dependence on ground-based TT&C stations.

1.3.2.2.6 Considering the Design Requirements for Public Platforms It usually takes 3–5 years to develop a new type of spacecraft, while the platform, consisting of structure, power supply, attitude and orbit control, propulsion, TT&C, thermal control, and other subsystems, can continue to be used in other newly developed spacecraft of the same type or developing new spacecraft of comparable size. In this case, the platform is often referred to as a public platform. To shorten lead time and reduce costs, the spacecraft platform should be designed as a public platform that can match a variety of payloads.

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CHAPTER 2

Design Method and Process of Spacecraft System

Liu Jie

Beijing Institute of Spacecraft System Engineering

CONTENTS

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The purpose of spacecraft system design is to coordinate and integrate the requirements and constraints of each system based on the analysis of mission requirements, through a systems engineering approach, effectively organize spacecraft platforms and payloads, and collaborate on the overall system design to form an optimal system that meets the requirements. The fundamental task of spacecraft system design is to design an optimized spacecraft system scheme that meets the user's specific mission requirements, assign a development mission task book to each subsystem department, and complete the system comprehensive design.[1]

This chapter introduces the spacecraft system design method and development process, focusing on a detailed analysis of the system design process, and briefly introduces the standard system and tool system in spacecraft system design.

2.1 DESIGN METHOD OF SPACECRAFT SYSTEM

2.1.1 System Design Procedure

Spacecraft system design can be divided into two parts, namely, system scheme design and system verification (as shown in Figure 2.1). The system scheme design can be subdivided into three phases: conceptual research (system scheme assumption), system scheme demonstration, and system scheme detailed design. The system verification can be divided into two phases: prototype verification and flight model verification. The technical problems found in the verification phase should be fed back to the scheme design department and modified, so the prototype verification and flight model verification are also called prototype design and flight model design.

2.1.1.1 System Scheme Design

2.1.1.1.1 Basic Tasks of System Scheme Design The basic tasks of spacecraft system design can be summarized as follows:

1. On the basis of mission analysis, the technical approaches to realize spacecraft mission including payload selection, orbit or constellation selection, launch vehicle, launch site, TT&C center and application system selection and constraint conditions formulation (coordinating interface between large-scale systems), and spacecraft system scheme assumption are proposed.

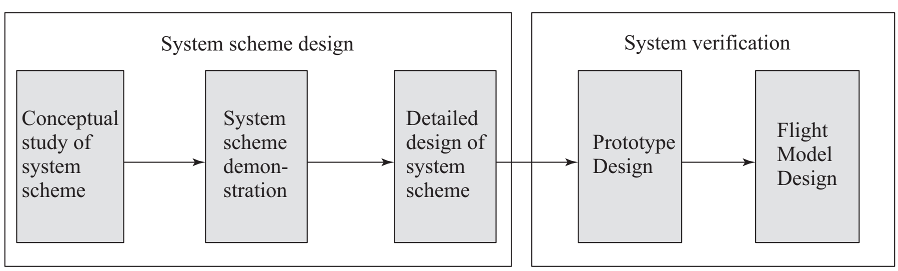


FIGURE 2.1 Spacecraft system design procedure.

2. Through scheme demonstration and selection, the user's requirements are transformed into the spacecraft system composed of several subsystems and the system function and performance parameters, so that the spacecraft system can adapt to the corresponding space environment, meet the constraints of large-scale system and other requirements, and complete the orbit or constellation design at the same time.
3. The function and performance parameters of spacecraft system are assigned into each subsystem and the next-level equipment; the optical, mechanical, thermal, and mechanical interfaces between the subsystems are designed and determined. Through analysis and coordination, the functional, physical, and program interfaces among the subsystems are matched, so that each subsystem can adapt to various corresponding environments.
4. Formulate the technological process of spacecraft system development. Determine the development phases in each process of spacecraft development, and development work (design, manufacturing, test, and verification), system models, tests, and needed ground test equipment and subsystem. Clarify the main and auxiliary lines and the sequence etc.

2.1.1.1.2 The Role of System Scheme Design Spacecraft system scheme design is the comprehensive and top-level design of spacecraft, which plays a very important role.[2]

Spacecraft system design is to design the system scheme based on mission analysis after receiving the model development mission, and then put forward the development mission requirements (including scheme, function, performance index, interface design, environmental test, technical process, and quality assurance engineering) to each subsystem department in combination with the requirements of engineering development. Then, the subsystem department can carry out the next step of research and development. Therefore, the system scheme design is a pioneering design from scratch, at the top, leading, leading position.

In addition to various significant contents in the system (such as orbit design, system scheme demonstration, system performance index analysis, configuration design, environmental condition analysis and formulation, environmental test requirements, large-scale system selection and interface design, electrical performance test requirements, reliability and safety analysis, and electromagnetic compatibility analysis), there are also relevant contents of each subsystem, such as the selection of subsystem scheme, subsystem functional performance index, various interface design, electrical performance test requirements, and environmental test requirements. Therefore, the system scheme design is multidisciplinary and comprehensive.[3]

In the system scheme design, the direction, overall situation, scheme, and subsystem design requirements for spacecraft development are determined. The advantages and disadvantages of the system scheme design directly affect the overall performance and quality of the spacecraft and the cycle and cost of spacecraft development. The system scheme design plays an important role in the whole spacecraft development. In the research institutes of the aerospace system industry department, the systems engineering department has been set up to undertake the system scheme design, system comprehensive design, and related system-specific research and development, and plays the role of technical management in the development of their respective spacecraft.

2.1.1.2 System Verification

System verification can be divided into five aspects according to its tasks:

2.1.1.2.1 Integrated Design Verification The verification of integrated design includes two aspects: one is mechanics-integrated design (general assembly design), which includes the design of bracket, welding, installation, and packaging according to the requirements of general layout, so as to connect the instruments, equipment, cables, and pipelines into an organic whole in mechanical aspect; the other is its electrical integration design (system circuit design, including power flow and information flow design), in which the distributor, initiating explosive device manager, cable network, etc. are designed according to the requirements of the system layout, connecting the instruments and equipment into an organic whole in terms of power supply and electrical information transmission.

2.1.1.2.2 Test Design Verification Test design verification mainly includes acceptance test design for subsystem and system-level test design for spacecraft. There are also system-level EMC test design and various tests (such as precision measurement, leakage detection, and quality characteristic measurement) in the final assembly. Through various function and performance tests, the integrity of spacecraft function, performance, and quality is verified.

2.1.1.2.3 Verification of Environmental Test Design Environmental test design verification is to put forward various environmental test conditions and requirements for instruments and equipment of subsystem and system-level products (spacecraft as a whole). Environmental tests mainly include the space environment tests, such as vacuum, particle irradiation, ultraviolet irradiation, magnetic environment, and mechanical environment tests, such as sinusoidal vibration, random vibration and noise, and environmental tests, such as thermal balance and thermal vacuum. Environmental test design is to determine test items, test conditions, and requirements according to various types of orbit, life, launch vehicle, and spacecraft platform. Based on the established conditions, through environmental tests and other tests, it is verified that the spacecraft can withstand various environmental tests and ensure the product quality.

2.1.1.2.4 Development Quality Assurance Engineering Requirements The above-mentioned system scheme design and system comprehensive design only design and put forward requirements from the aspects of scheme, function and performance, interface, technical process, final assembly, test, and experiment, but it is not enough for the development of qualified spacecraft. Based on years of experience at home and abroad, a set of quality assurance engineering requirements should be developed. Different requirements should be put forward for different spacecraft, including outline and specification of product quality assurance, reliability and safety, availability, testability, maintainability, electromagnetic compatibility, test coverage, and software assurance of the spacecraft products developed, to ensure the development quality of each subsystem and spacecraft system.

2.1.1.2.5 Launch Implementation Requirements The selection and interface coordination of the major systems have been basically completed in the scheme phase, but the formal document requirements should be put forward in the system comprehensive design phase to put forward the launch implementation requirements to the major systems. These requirements include: launch requirements for launch vehicles (such as spacecraft mass, orbit parameters, and orbit accuracy); technical requirements for launch site (such as propellant injection and launch window); requirements for TT&C Center (such as satellite ground docking, TT&C program, and data processing); requirements for a ground application center (such as satellite ground docking and on-orbit test).

2.1.1.3 System Optimization Design

Because the design can meet the requirements of users with a specific function of the system that can have many programs, therefore, in the process of scheme design, the optimal program should be designed by optimization.

In fact, optimization has been taken into account in the process of general scheme design. For example, the composition should be reasonable, the function and performance should meet the user's and overall requirements, the interface relationship should be complete and correct, and the development technology process and environmental simulation test should be reasonable and necessary. However, the multiple schemes obtained through the above general scheme design are not necessarily the best and may have their own advantages and disadvantages. This requires further detailed optimization design, or that the better part of some schemes can be recombined into a better scheme. It should be pointed out that the advantages and disadvantages sometimes vary with the evaluation factors. For example, the use of new technologies can achieve good performance, which is an advantage in terms of performance factors, but it is a disadvantage in terms of cost.

Optimal design, whether qualitative or quantitative, should use the following six indicators as factors to evaluate the pros and cons of the scheme or as the objective function of the optimal design:

1. Technical performance (whether it meets the user's and general requirements)
2. Interface coordination (whether all subsystems and major systems are coordinated or not)
3. Development cost (lowest or not)

4. Development progress (fastest)
5. Constraints (adaptation)
6. Degree of risk (minimum or not)
7. Advanced (competitive)

There are many theories and methods of optimization design, which have been applied in some fields. However, there is no recognized mature method, and systems engineering is still under discussion in the field of spacecraft. This is mainly due to the fact that there are many types of spacecrafts, systems, and specialties; more importantly, there are many optimization objective functions for the system design of spacecraft, and it is difficult to establish a general model to complete quantitative optimization. At present, we often use the method of trade-off to make decisions through qualitative and quantitative comparison.

In general, the process of space system optimization design is shown in Figure 2.2.

2.1.1.4 Position and Function of Spacecraft System Design

According to the function, task, and feature of the systems engineering, it is not difficult to understand that spacecraft system design is the top-level comprehensive design of spacecraft. Therefore, it has an important position and role.

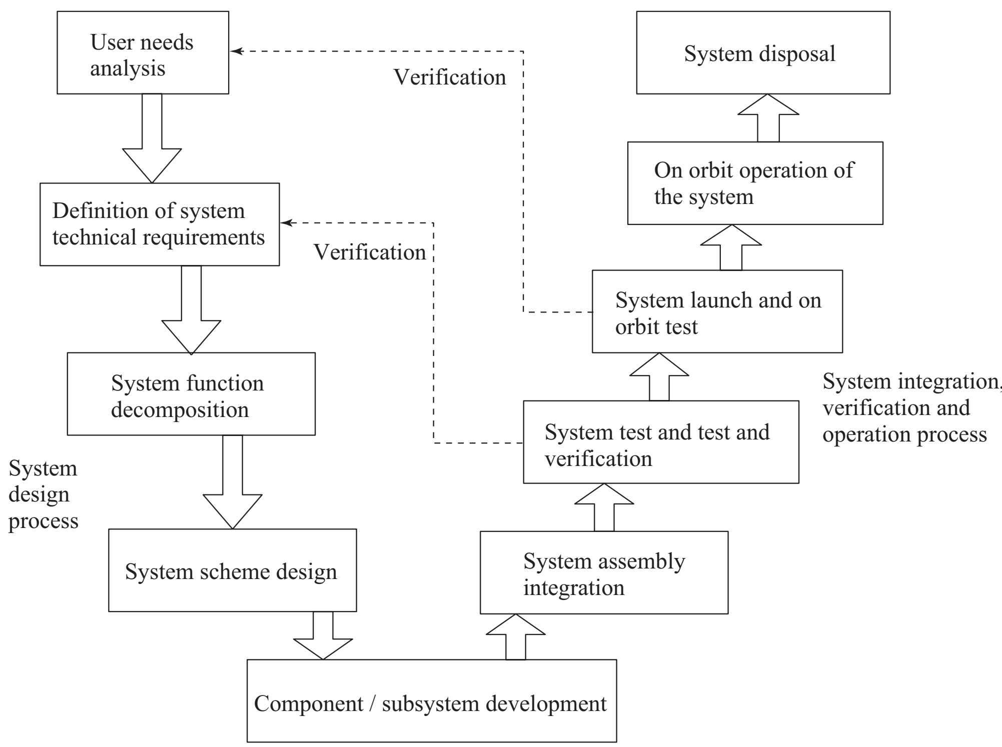


FIGURE 2.2 Optimization design process of spacecraft system.

Top-level design is determined by the hierarchy of the system. After the systems engineering department of spacecraft gets the spacecraft model development task, the systems engineering designer first carries out the system scheme design on the basis of the task analysis. After the system scheme (detailed) design is completed, the development task requirements (including scheme, function, performance index, interface design, environmental test, environmental test, technical process, and quality assurance engineering) shall be put forward to each subsystem department in combination with the requirements of engineering development. Then, the subsystem department can carry out the next step of research and development. Therefore, the system scheme design is a pioneering design within the spacecraft system level, that is, the top-level design.

The so-called comprehensive design refers to a lot of significant content involved in the system scheme design. In addition to the contents of various specialties (such as orbit design, system scheme demonstration, system performance index analysis, spacecraft configuration design, environmental condition analysis and formulation, environmental test requirements, large-scale system selection and interface design, electrical performance test requirements, reliability and safety analysis, and electromagnetic compatibility analysis) in the system level, the contents of system scheme design are not included. There are also relevant contents of each subsystem (selection of subsystem scheme, analysis of subsystem function and performance index, various interface design, electrical performance test requirements, environmental test requirements, etc.). Therefore, the system scheme design is a multidisciplinary and multi-professional comprehensive design, and it should be widely coordinated and cooperated with many departments, that is, it is in the position of "technical management".

From the position of top-level design and the content of comprehensive design, we can see that the system scheme design is a design that sets the orientation, overall situation, scheme, and subsystem design requirements of spacecraft research and development. Therefore, the system scheme design plays an innovative, decision-making, leading, and comprehensive role. From the nature and status of the systems engineering, it can be seen that whether the system scheme design is good or not directly affects the systems engineering performance and quality of the spacecraft, as well as the development cycle and cost of the entire spacecraft.

In conclusion, the system scheme design plays an important role in spacecraft development. Because of this, the systems engineering department has been set up in all industrial departments of the space system, including the research institutes for developing spacecraft, launch vehicles, and various missiles. Each systems engineering department, through system scheme design, system comprehensive design, and related system-specific development, has played the role of "technical management" in the development of their own aerospace models. The spacecraft development units implement spacecraft engineering development and complete space missions through the technical management of the systems engineering design department. This is the most effective method of modern complex engineering system, which is also in line with the system and systems engineering principles described in Chapter 1.

2.1.2 General Framework of System Design

Systems engineering manuals have been compiled by major space agencies such as the United States and Europe. In particular, NASA systems engineering manuals are mainly written for spacecraft systems. It is found that these manuals focus on systems engineering phase division, process definition, activity description, and method tool introduction, including both systems engineering technology and management contents.

Through a comprehensive analysis of the domestic and foreign spacecraft system design process, the spacecraft systems engineering technology architecture (as shown in Figure 2.3) can be obtained. Its technical elements are divided into "One Core, Two Supports and One Guarantee".

1. One core is systems engineering technology system core, that is, systems engineering technology process and activities.
2. Two supports include the important support of systems engineering technology activities, that is, system engineering-specific technology, systems engineering tool method, and systems engineering standard specification.
3. It is an important support for the sustainable development of systems engineering technology, including systems engineering experience summary, foreign systems engineering technology research and reference, and on-orbit performance analysis and application of spacecraft.
4. One guarantee refers to the important guarantee for the sustainable development of systems engineering technology, which is the perfect organization of systems engineering research, system designer training, and systems engineering technology exchange.

2.2 SPACECRAFT SYSTEM DEVELOPMENT PHASE

The development of spacecraft system is a complex work, which often takes a long time from the beginning of feasibility study to the final launch of spacecraft into orbit. In order to facilitate management, the development process is often divided into several different phases, each with specifying development goals and work content, and is gradually and iteratively progressed. The development program reflects the scientific law of the space system development process, ensures that a long period of development process can orbit and control the target in phases, and makes the development process orderly. The development phases of China Aerospace's spacecraft system are shown in Figure 2.4.

The work contents and milestone of each phase are shown in Table 2.1.

In general, the spacecraft development process should strictly abide by the standard development procedures. The next phase can only be carried out after the successful completion of the development work specified in the previous phase. However, due to the progress of research and development, sometimes it is necessary to go beyond the limitation of research and development phase. For example, in order to start the long-term work as soon as possible, some subsystems that are still in the scheme phase may be used in the initial phase of the prototype phase, and some technical development work that should

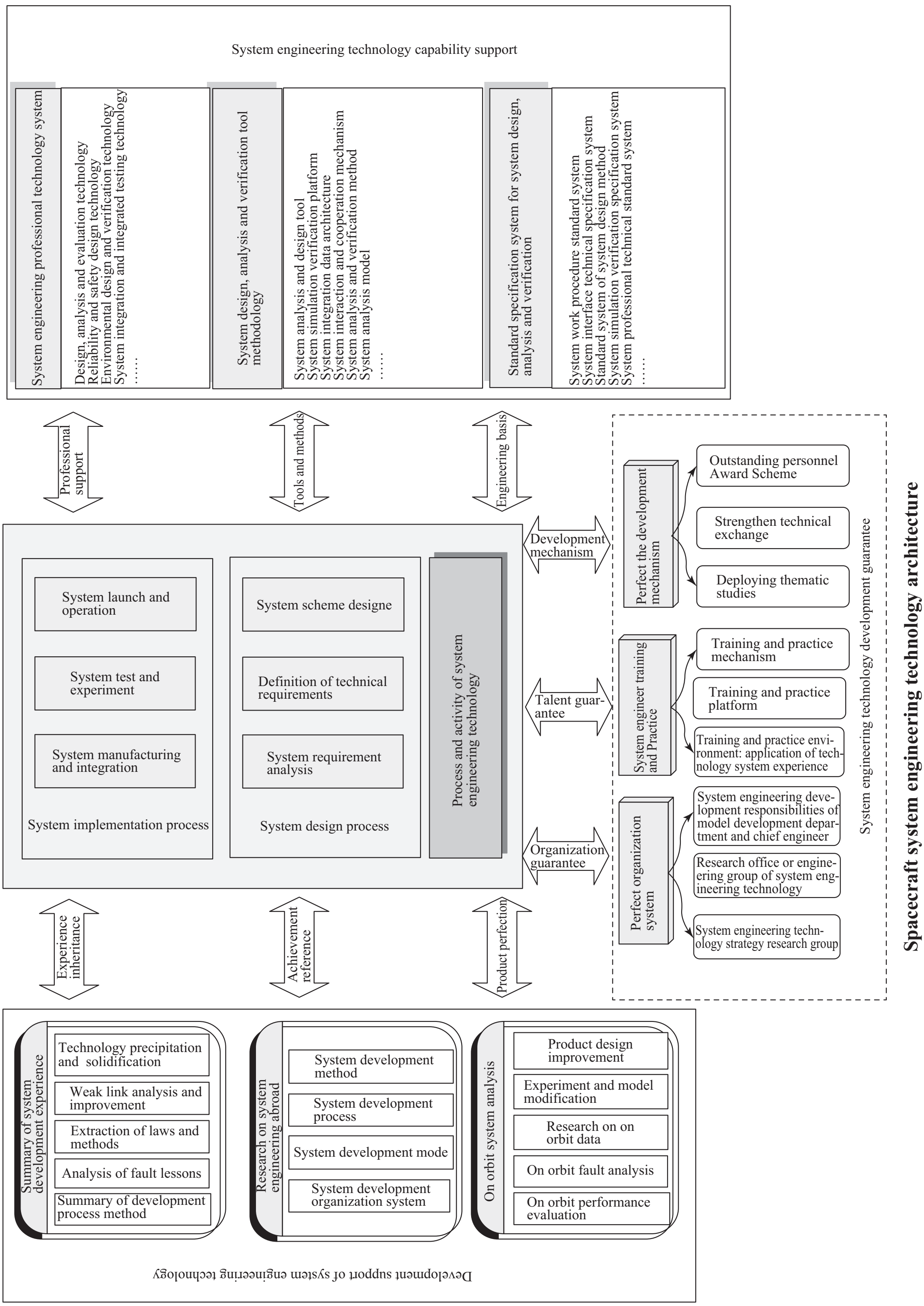


FIGURE 2.3 Spacecraft systems engineering technology architecture.

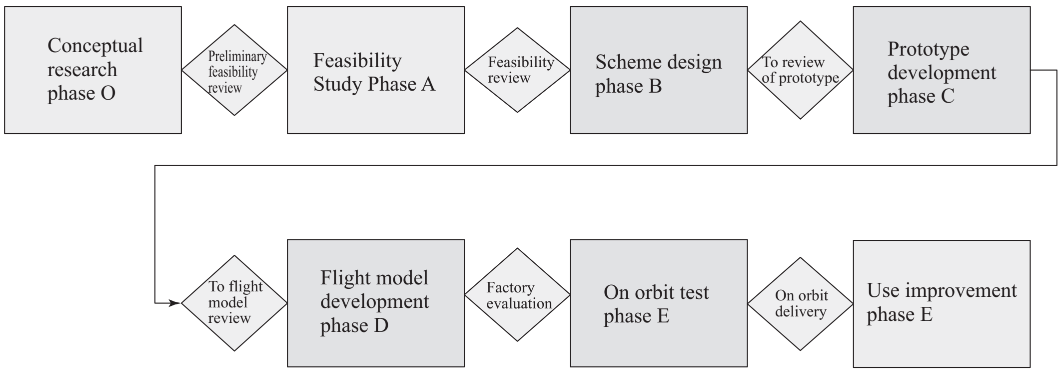


FIGURE 2.4 Life cycle of China's aerospace products.

TABLE 2.1 Phase Division of Spacecraft System Development

Project Phase Division		• **Main Milestone**
Project establishment and start-up phase	• Conceptual research	a. Completion of task requirement analysis and review b. Preliminary determination of spacecraft preliminary use requirements and technical requirements c. Project comprehensive demonstration report d. Demonstration report of spacecraft feasibility preliminary scheme
	• Feasibility demonstration	a. Clear conclusion of key technology tackling b. Feasibility study report of spacecraft c. Economic feasibility study report
Project implementation phase	• Scheme design	a. Complete the system scheme design report of spacecraft and pass the review b. Interface control document between spacecraft system and other systems c. Preparation of subsystem development assignment d. Complete reliability work plan e. Sign development contract or agreement
	• Development of prototype	a. Complete reliability and safety design b. The final assembly of the satellite prototype and various large-scale ground tests were completed and passed the review c. All the quality problems found have been solved d. The technical status of satellite flight model is determined and the list of flight model components is put forward e. Complete the prototype development summary report, prototype reliability report, and whole satellite flight model design report
	• Development of a flight model	a. Passed special reviews on components, software, technical status, reliability and safety, and zero quality problems b. Passed factory review
Long-term operation phase	• Satellite on-orbit testing	a. Completed on-orbit testing of platform and payload b. Complete the summary of spacecraft on-orbit test and pass the review c. On-orbit delivery of spacecraft to users
	• Use improvement	a. Long-term operation management of spacecraft on orbit b. Modification design and production of follow-up spacecraft

Note: The phase division of spacecraft system project in the table refers to that the design of common platform is not adopted in the development of new spacecraft, and the phase division can be changed after the design of common platform is adopted.

normally be finished in the scheme phase may also continue to the prototype phase or the flight model phase. In recent years, with the continuous development and maturity of public platform, the spacecraft based on public platform, or the equipment model or business replacement model with mature technology and good inheritance, can be tailored according to the actual development basis. For example, at present, many equipment models adopt the one-step flight model development mode, directly across the prototype development phase.

2.3 GENERAL DESIGN FLOW OF SPACECRAFT SYSTEM

In real spacecraft system development, system designers are most familiar with the technological process and planning process of spacecraft system development. In the actual operation process of technological process, due to the mismatch of resource allocation and the objective existence of short-term projects, the implementation of technological process is not carried out according to the serial or parallel relationship of design, some technical activities are often put in front or behind according to the development progress, and even some activities are not operated and completed at the actual phase. In addition, system designers often confuse the relationship between “spacecraft system development technology process” and “system design technology process”. In fact, the technological process of spacecraft system development is not equal to the technological process of system design. The technological process of system design is a subset of the technological process of model development. If it is abstracted, the technological process of system design of each spacecraft is basically the same.

This section only discusses the general process of feasibility demonstration phase, scheme design phase, prototype development phase, and flight model development phase in spacecraft system design, and does not describe on-orbit test and utilization improvement phase.

2.3.1 Process of Conceptual Demonstration Phase

2.3.1.1 Primary Coverage

From the perspective of spacecraft system development process, conceptual demonstration and feasibility demonstration are generally carried out together. The purpose of the mission in the conceptual (or feasibility) demonstration phase is to determine the capability and technical indicators of the satellite platform and payload through user’s needs analysis and satellite mission analysis and complete the preliminary feasibility scheme to realize the mission, which is used to support the comprehensive demonstration of the project.

At the beginning of the conceptual demonstration phase, the user usually organizes the development party to participate in the mission requirement analysis of the satellite, and formulate the preliminary use requirements and technical requirements of the satellite. The development party carries out the preliminary research of the spacecraft on the basis of the feasibility analysis of the preliminary use requirements.

The main work of the conceptual demonstration includes that: first, the user’s needs and their realizability are analyzed according to the user’s initial requirements and technical indicators; second, on the basis of user’s needs analysis, the mission analysis is carried out to analyze the attainable technical indicators and technical level, and the user indicators are transformed into spacecraft technical indicators and requirements. On this basis, the functional baseline of the spacecraft system is established, and the technical indicators of the functional baseline and their realization ways are analyzed and demonstrated to complete the decomposition of the whole satellite technical indicators; Third, on the premise of determining the technical indexes of spacecraft, the feasibility demonstrations of system and subsystem are carried out, the key technologies of system and subsystem level are analyzed and determined, the technical performance, uncertainty, and risk degree are predicted and analyzed, the preliminary solutions are proposed, and the key technology tackling is organized. In addition, the interface coordination and analysis of large-scale system should be carried out at the same time, and the development program and development cycle should be proposed. Finally, they cooperate with relevant departments to complete the comprehensive project approval and assist users to complete the preparation of "general requirements for spacecraft development".

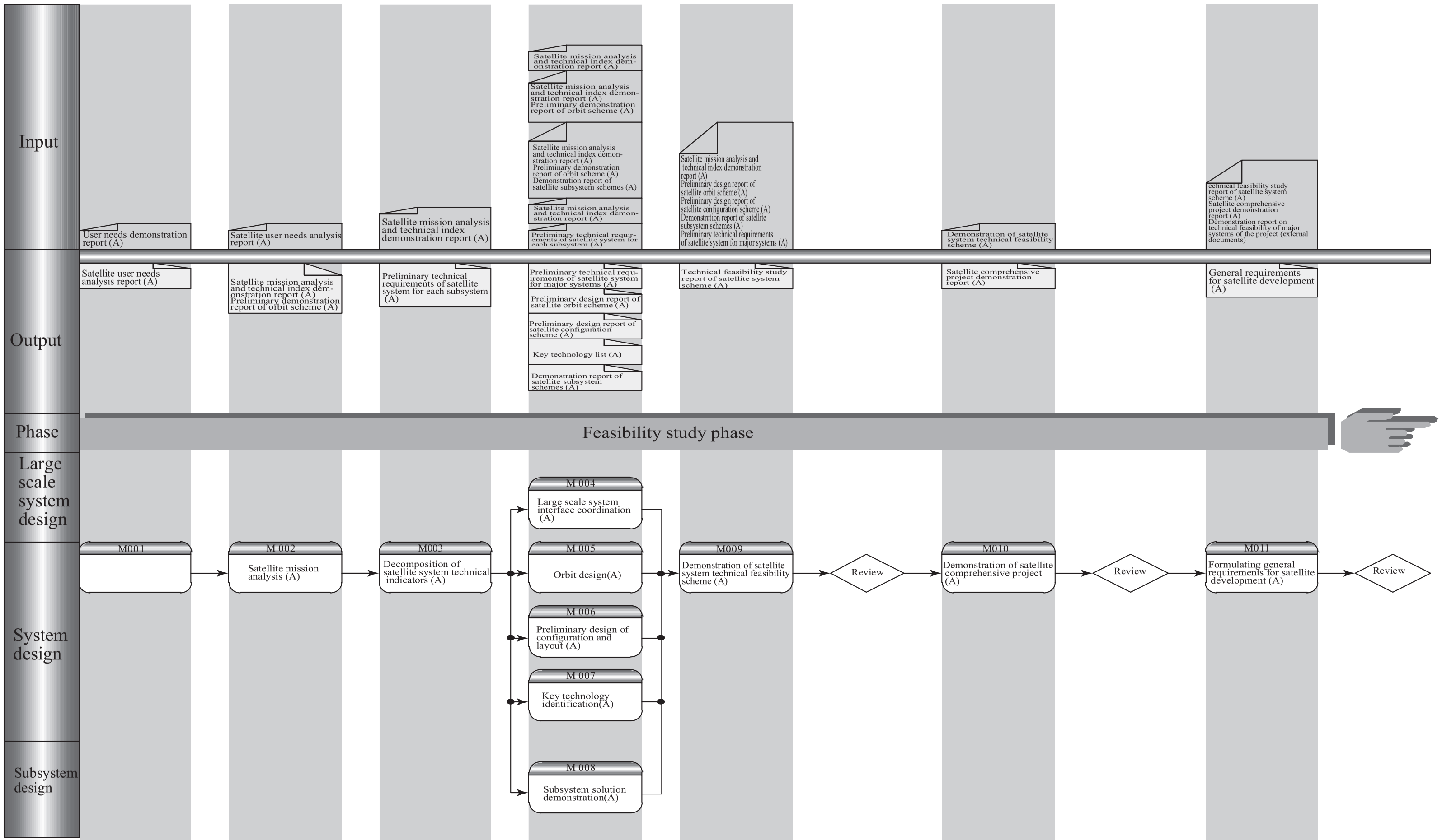
2.3.1.2 General Design Process

The general design process of spacecraft system design concept demonstration phase is shown in Figure 2.5. The milestone of this phase includes completing the "spacecraft feasibility scheme demonstration report" and passing the review, proposing the key technology solutions, and summarizing the key technology in tackling various situations.

2.3.1.3 Key Links Description

2.3.1.3.1 User's Needs Analysis and Task Analysis User's needs analysis is the premise and foundation of spacecraft design and feasibility demonstration. The purpose of the analysis is to transform the user's needs into the requirements of engineering applications, and to analyze the realizability of user's needs. In the process of analysis, the systems engineering technician must reach a consensus with the user about the concept, connotation, and extension of relevant contents in the comprehensive demonstration report of user's needs. According to the relevant standards and specifications in application field, the use requirements shall be transformed into the functional and task requirements of the system, and the technical indexes shall be converted into the performance index requirements of spacecraft system.

The purpose of task analysis is to establish a clear logical relationship between user's needs and system scheme design through the analysis and decomposition of user function and performance requirements. Based on the analysis of different technical approaches, the process of task analysis is often to form multiple system solutions, from which the optimal system solution shall be found through multi-scheme analysis and comparison. Task analysis activity is to establish the top-level baseline of the system from a kind of technical "chaos", which needs the support of multidisciplinary technology. In the analysis activities, the input of satellite payload preliminary scheme is generally required, and the preliminary demonstration of orbit scheme should be carried out at the same time. If necessary, the preliminary scheme demonstration of key platform subsystem should also be followed up at the same time. The technical execution of this activity is not a single line and needs to be iterated continuously.



General process of spacecraft system design in conceptual demonstration phase

FIGURE 2.5 General process of spacecraft system design in conceptual demonstration phase.

2.3.1.3.2 Decomposition of Spacecraft System Technical Specifications The purpose of the spacecraft system technical index decomposition is to develop a technical index system that will guide the subsystem for the next step of demonstration and scheme design. Task analysis is based on index decomposition at the system level. The system technical specifications decomposition of satellite is to decompose a more detailed index system that each subsystem should meet under the clear system index. Some indexes are obtained through demonstration, some indexes are obtained through prior knowledge, and some indexes are enveloping in the conceptual demonstration phase, which is not necessarily very accurate, but it must meet the requirements of system index.

2.3.1.3.3 Large-Scale System Interface Coordination Large-scale system interface coordination is to define the interface relationship between spacecraft and other systems (carrier system, TT&C system, launch site system, ground application system, etc.) and the conditions that need the support of other systems. The satellite system and other major systems successively put forward interface requirements to each other. After several rounds of negotiation and index confirmation, the preliminary technical requirements or specifications for major systems put forward were finally confirmed. Large-scale system interface coordination can be carried out in parallel on the basis of the certain constraints between them. The interface coordination of large-scale system in the conceptual demonstration phase is mainly to compare multiple schemes, demonstrate whether the large-scale system interface is feasible, select the best, and provide external conditions for the design work in the satellite scheme phase.

2.3.1.3.4 Orbit Design The connotation of spacecraft orbit design is the process of determining spacecraft flight orbit and orbit-related launch, operation, and control procedures according to spacecraft mission and related constraints.

Spacecraft orbit design includes mission requirement analysis, flight orbit design, operation orbit design, return orbit design, orbit maneuver design, orbit maintenance design, launch window design, and orbit analysis and simulation. The main process of spacecraft orbit design is as follows:

Analyzing the mission profile of spacecraft,

Extracting the design constraints of the flight orbit,

Giving several orbit schemes that initially meet the mission objectives,

Comparing and analyzing multiple schemes from the perspectives of economy, inheritance, mission requirements satisfaction and realizability, and

Analyzing and calculating the conditions of TT&C, illumination, etc.,

Finally, the orbit design scheme suitable for engineering application is given.

2.3.1.3.5 Preliminary Design of Configuration and Layout Through the design analysis and technical coordination of the spacecraft's shape, structural form, preliminary system layout, mass characteristics, and the interface relationship with the launch vehicle, one or more of the spacecraft's overall form and equipment installation positions are given to meet the mission requirements under the conditions of given space environment, orbit conditions, mission target characteristic requirements, and large-scale system constraints. In the conceptual demonstration phase, configuration and layout design mainly solves the configuration problem of spacecraft and the layout of important equipment related to flight mission and supports the system feasibility scheme of spacecraft from the aspect of integrated design.

The configuration design part includes the overall shape and parameters of the spacecraft, the division of cabin and functional components, and the main structural forms and configurations of the spacecraft in different states. The layout design is generally carried out according to the divided spacecraft cabin, the position of spacecraft equipment in the whole satellite is arranged and coordinated, and the satisfaction of various constraints and requirements is analyzed.

2.3.1.3.6 Key Technology Identification Key technology identification is an important system technical activity in the phase of conceptual demonstration. The realization of any system scheme usually needs a series of key technologies. Only when the key technologies are solved, the system scheme can become a feasible scheme. The accuracy of key technology identification determines the risk and difficulty of the subsequent scheme.

2.3.1.3.7 Demonstration of Subsystem Scheme The purpose of subsystem scheme demonstration activity is to form scheme demonstration report or preliminary scheme report of each subsystem according to the system technical specifications requirements of the subsystem, so as to support the spacecraft system scheme. In the conceptual demonstration phase, the demonstration work of subsystem scheme is generally rough, and there are generally two output modes. One is to form a complete and feasible subsystem scheme, and the other is to realize the feedback of system requirements through subsystem scheme demonstration, so as to complete the optimal scheme of the system.

2.3.1.3.8 Demonstration of System Technical Feasibility Plan of Spacecraft
The demonstration report of system technical feasibility plan of spacecraft is the final system-level technical achievement in the conceptual demonstration phase. Through the output of this activity, it is proved that the key technology of the project has been broken through and has the model approval conditions. The demonstration report of the system technical feasibility plan of spacecraft includes mission requirements, task analysis, satellite system scheme, subsystem scheme, key technology breakthrough, inheritance analysis, technical process, planning process, etc. The report is a summary of the realization of all technologies in the feasibility demonstration phase, and also gives a brief outlook on the model development after the project is established.

2.3.2 Process of Scheme Design Phase

2.3.2.1 Primary Coverage

In the scheme design phase, the work is carried out on the basis of feasibility demonstration, mainly through the decomposition of system technical indicators, formulation of relevant specifications, orbit design, configuration layout and assembly design, subsystem scheme design, etc., to complete the design of satellite system scheme and provide input for satellite prototype research.

The work in the scheme design phase mainly includes that the demonstration and decomposition of the system technical specifications of the spacecraft are first carried out on the basis of the feasibility demonstration to obtain the system technical specifications of the spacecraft and the technical indexes of the corresponding subsystems; then, the preparation of relevant specification documents (including the formulation of spacecraft design and construction specification, space environmental protection design specification, and electromagnetic compatibility specification and environmental specification) is carried out, and the reliability and safety outline of spacecraft is prepared to provide basis and input for system scheme and subsystem scheme design; next, the work, such as the large-scale system interface coordination, detailed orbit design, satellite configuration layout and final assembly design, satellite development test plan and satellite development technical process formulation, and detailed design of each subsystem scheme, is carried out; the technical requirements for each major system are put forward; satellite orbit change and on-orbit maintenance scheme are obtained; satellite configuration and layout design report and final assembly design report are completed; and spacecraft development and test plan and technological process are developed; finally, the spacecraft system scheme design report is completed, and the prototype phase is commenced after the review is passed.

The main marks of the completion of the scheme phase include the preparation of the spacecraft system scheme design report and passing the review; completion of the interface control documents with other systems; and preparation of the mission statement of spacecraft subsystem development.

Scheme design is an important phase in spacecraft development, which directly determines the design level and comprehensive performance of spacecraft. In the scheme phase, the system and subsystem personnel are required to work together, and the system design and analysis elements are comprehensively considered. The optimized system scheme is obtained through repeated iteration to ensure the correctness, rationality, and advanced nature of the design.

2.3.2.2 General Design Process

The general design process of spacecraft system design scheme phase is shown in Figure 2.6. The milestone of this phase includes completing the "spacecraft system scheme design report" and passing the review, and summarizing the work of spacecraft scheme phase.

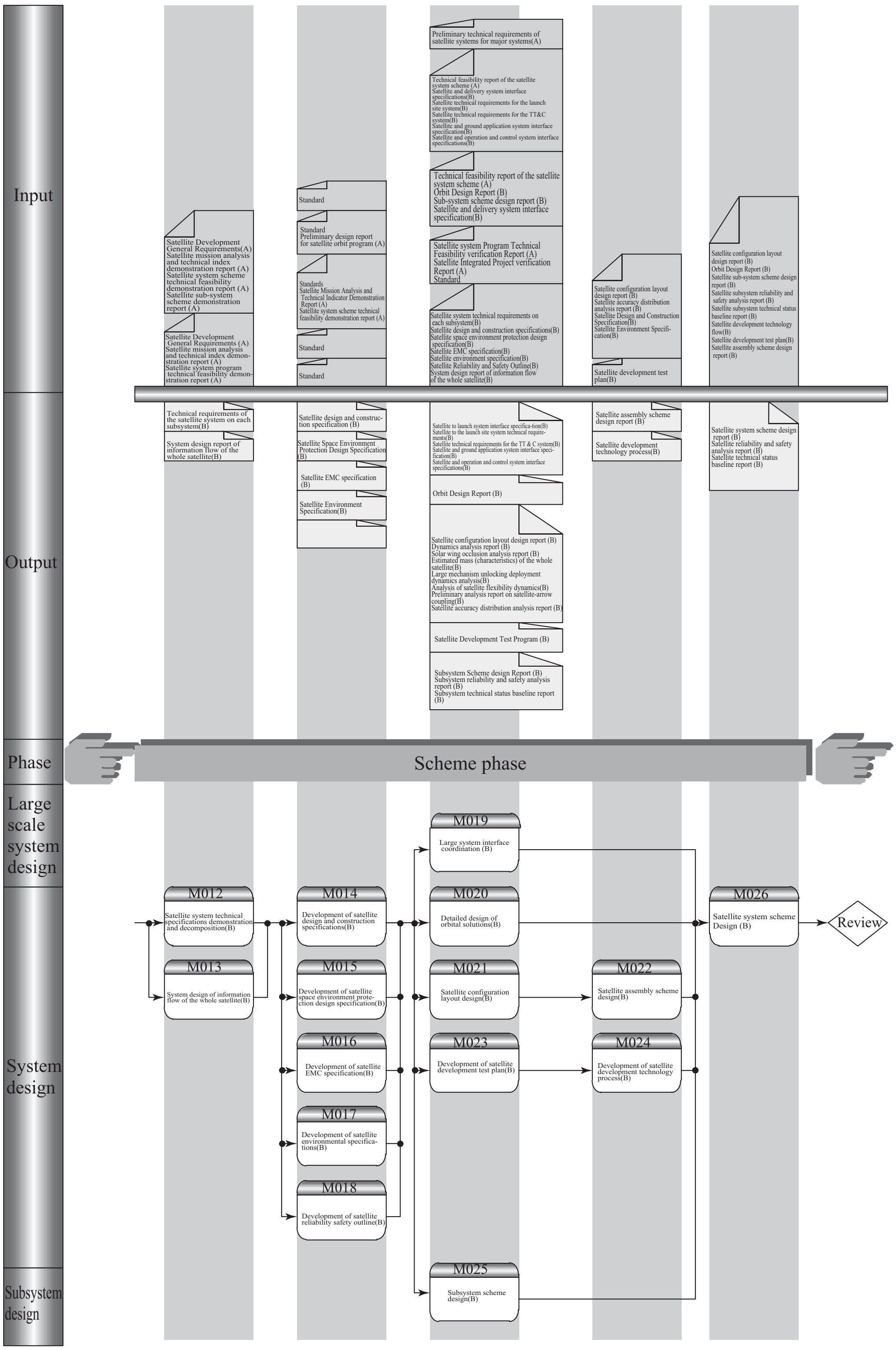


FIGURE 2.6 General flow of spacecraft system design in the scheme phase.

2.3.2.3 Key Link Description

2.3.2.3.1 Spacecraft System Technical Specifications Demonstration and Decomposition The purpose of the demonstration and decomposition of the spacecraft system technical specifications in the scheme phase is to deepen the demonstration and verification of the system specifications based on the results of the "mission analysis", and to form the technical requirements of the spacecraft system to the subsystems in the scheme phase by decomposing the system specifications and subsystem specifications in more detail to guide the demonstration and design of the subsystem in the scheme phase.

2.3.2.3.2 Spacecraft Top-Level Design Specification Spacecraft top-level design specifications include the design and construction specifications, space environment protection design specifications, electromagnetic compatibility specifications, and environmental test specifications.

The design and construction specification defines at the system level the design principles that each subsystem must follow during the model development process and the mechanical, electrical, and thermal interface requirements for the whole satellite. For spacecraft development, the design and construction specification is the technical "constitution", in which the common principles and interface requirements must be strictly followed by each subsystem during the design and development process. The design and construction specification includes two levels of constraints: the design level and the construction level. In the scheme phase, more attention is paid to the principles and interface requirements of the "design layer", and in the prototype/flight model phase, more attention is paid to the principles and interface requirements of the "build layer".

Based on the preliminary scheme of the spacecraft system and the initial determined operation orbit, the space environment protection design specification mainly elaborates the space environment profile and its space environment effect in the satellite operation on orbit, and accordingly puts forward the corresponding design requirements and measures for the satellite. According to the space environment and effects, the space environment protection design specification mainly puts forward the requirements for the selection and control of components and materials, the space environment test requirements, the space environment effect analysis requirements for the system, subsystems and equipment, and the related system design requirements.

The system EMC specification mainly stipulates the EMC requirements at system level, subsystem level, and equipment level of spacecraft systems, including the basic principles and design methods of satellite EMC design, electromagnetic interference control, electrical overlap, electrical grounding, cable selection and cabling, EMC testing and management, as well as the requirements, methods, qualification criteria, and acceptance principles of EMC testing at equipment level.

The environmental specifications are mainly proposed for the key equipment that needs to be put into operation in the scheme phase, with the purpose of providing a basis for the development and testing of key equipment in the scheme phase. The specification mainly specifies the environmental test items, test conditions, test sequence, and other test requirements for the relevant individual machines, and gives the environmental test matrix.

2.3.2.3.3 Develop a Spacecraft Reliability and Safety Assurance Outline Reliability and safety assurance outline is the reliability work items and their requirements that are formulated in accordance with the specifications and standards related to the reliability and safety assurance of spacecraft at the early phase of spacecraft development to ensure the reliability and safety of spacecraft to meet the characteristics of spacecraft missions. The main contents to be considered in the reliability outline are development of reliability work plan, reliability modeling, allocation and estimation of reliability indexes, reliability design guidelines, environmental impact analysis and its protection design, derating design, determination and control of reliability key items, software reliability assurance, reliability verification, reliability test, reliability control of outsourcing units, reliability evaluation, etc. The main contents considered in the safety outline are relevant requirements for safety design (including design requirements for safety of propulsion subsystem, safety design requirements for power supply and distribution, and safety design requirements for pyrotechnics and ignition circuits), definition and analysis of hazard sources, safety assurance during ground operation (including production, test site, and launch site), risk evaluation, verification of safety and its evaluation, etc.

2.3.2.3.4 Large System Interface Coordination Compared with the conceptual demonstration phase, the coordination of large system interfaces in the scheme phase mainly focuses on the design of spacecraft interfaces with other large systems and the refinement of the main technical indicators. After several rounds of consultation and confirmation on indicators, the initial technical requirements of spacecraft systems to other large systems in the scheme phase are initially confirmed to ensure the compatibility of large system interfaces.

2.3.2.3.5 Detailed Design of the Orbit Program The detailed design of the orbital program in the spacecraft scheme design phase is a more detailed design of the orbital program based on the preliminary analysis and design of the orbital program in the feasibility demonstration phase and the development progress and new input conditions. According to the requirements of users, the mission of spacecraft, the constraints of spacecraft platform and payload, and the constraints of large-scale systems such as launch vehicle, launch site, TT&C network, landing site, and ground application, in the detailed design of orbit scheme, the orbit design scheme is determined in detail, the TT&C conditions, illumination conditions, and orbit perturbation are analyzed, and the orbit control strategy, propellant consumption estimation, launch window scheme, and other orbit-related problems are proposed.

2.3.2.3.6 Spacecraft Configuration Layout Design In the configuration layout design in the scheme phase, the configuration layout design of the feasibility phase is deepened by re-evaluating and analyzing the configuration of the feasibility phase based on the new requirements of the spacecraft mission proposed in the scheme phase, the orbit situation and the deeply coordinated large system constraints (including the capacity, mechanical environmental conditions, and spacecraft available volume), and adding spacecraft accessories based on the new requirements. If the new requirements of the scheme phase still cannot be met, the configuration and analysis must be re-conformed. The layout of this phase focuses on the system layout based on the mission constraints (flight mode and procedures), launch constraints, orbit constraints, and payload and equipment constraints, and the system layout is based on the requirements on the subsystem providing on-satellite instrument dimensions, quality, center of mass, power consumption, heat generation, field of view, orientation, installation position, electrical connection between the instruments, and the mechanical interface relationship between the satellite and the launch vehicle. It is necessary to focus on the layout of the payload, the equipment and outriggers arranged on the satellite surface, and the moving parts, such as thrusters, antennas, solar wings, and attitude sensors. The configuration layout analysis generally includes mass characteristics analysis, optical components field of view occlusion analysis, antenna beam angle occlusion analysis (can also be classified as field of view occlusion analysis), solar wing occlusion analysis, thruster plume analysis, moving parts motion envelope interference analysis, spacecraft stiffness analysis, assembly and test operability analysis, and EMC analysis to confirm the effectiveness of the layout. For spacecraft containing large flexible unfolding attachments, flexible dynamics analysis and mechanism unlocking and unfolding dynamics analysis are required.

2.3.2.3.7 Spacecraft Final Assembly Scheme Design The purpose of spacecraft final assembly scheme design is to ensure the spacecraft assembly and the qualified mechanical interfaces required for the implementation of various work items for development, to realize the connection between the spacecraft and various types of ground mechanical support equipment and the work items such as transportation and docking with delivery and to have the feasibility of guaranteeing reassembly and fault repair. The spacecraft assembly scheme design is to plan the spacecraft final assembly design as a whole, propose feasible realization methods for each design work of the assembly, and to provide the basis for the detailed design of the spacecraft assembly. The main work is based on the system layout of the spacecraft, to provide the assembly scheme design, including the decomposition and docking scheme, satellite lifting and parking and transportation scheme, instrument installation and disassembly scheme, propulsion piping layout, orientation, welding and leak detection scheme, precision measurement scheme, high- and low-frequency cable orientation scheme, grounding scheme, assembly and test operation scheme, design scheme and analysis of the accessories of the final assembly, selection of various tooling and ground fasteners, and safety protection.

2.3.2.3.8 Develop Spacecraft Development Test Plans and Technical Processes The main contents involved in the research and test plan document include the development strategy, system development, platform development, payload development, ground support equipment development, assembly, test and test plan, and quality and risk management. The starting point of the entire spacecraft development process system is the beginning of the scheme phase, and the termination point is the delivery of the spacecraft to the orbit (such as communication satellites and meteorological satellites) or the delivery of the test program to the user after the spacecraft returns to the ground (such as return satellites and spacecraft). The whole spacecraft development process includes the scheme phase, the prototype phase, and the flight model phase.

2.3.2.3.9 Subsystem Design The design of each subsystem is carried out almost in parallel in the scheme phase, and each subsystem department carries out subsystem-level and stand-alone scheme design work under the constraints of system top-level specifications and technical requirements for system to the subsystem, and continuously feeds back problems and mismatched links to the systems engineering department in the process of each subsystem scheme design. The design work at each level is completed in top-down and bottom-up iterations.

2.3.2.3.10 System Scheme Design of Spacecraft The purpose of the system design of spacecraft is to transform the user's requirements into the functions and performance parameters of the spacecraft system composed of several subsystems, and make the spacecraft system adapt to the corresponding space environment, so as to meet the constraint requirements of large-scale systems (launch vehicle, launch site, TT&C center, and application system) and other requirements. The basic contents of the system scheme design include the task analysis, orbit design, demonstration of subsystem scheme, system scheme demonstration, the system performance index determination, typical parameter budget, configuration design, large-scale system coordination, key technology analysis, reliability design, technical process development, and funding and cycle determination. Finally, the research and development mission statement is proposed to each subsystem department, and the corresponding technical requirements are proposed to each major system department.

2.3.3 Prototype Development Process

2.3.3.1 Primary Coverage

The prototype development phase follows immediately after the scheme phase. In the prototype phase, the scheme design of the system and subsystem in the scheme phase is implemented in the project, the system and subsystem equipment are developed and integrated into various test models, and the correctness and rationality of the system and subsystem design (including the interface design between systems), that is, the extent to which the whole satellite design meets the mission requirements is verified through various models and tests.

The main design work of the prototype development includes that:

The specifications or requirements at the satellite and subsystem level are established or revised as the technical basis for the detailed design of the system and subsystem; during the detailed design of the subsystem, the mechanical, electrical, thermal, and other interfaces are coordinated at the equipment level, the subsystem is designed and analyzed in detail, the equipment interface table at the equipment level is determined, entire satellite layout is designed and analyzed, and the specific requirements for the implementation of system integration are proposed, which are output as the basis of system integration design. The work in this phase is the basis of system top-level design and subsystem top-level design, and the system top-level design specification and subsystem detailed design need to be confirmed by way of review to prevent errors or deviations in the system top-level design, which may lead to repeated system integration and equipment development.

Design and analyze the mechanical, electrical, thermal, and other interfaces between the spacecraft system and subsystem, complete the design of data interface between the system and subsystem, electrical interface design and analysis, and reliability and safety analysis, and put forward the specific requirements for system integration design and implementation, complete the design of ground support equipment of spacecraft system and subsystem and the integrated design of various whole satellite models, including the system circuit design, telemetry and remote control channel allocation, flight scheme design, power balance analysis, satellite system test coverage analysis, satellite system reliability, safety analysis, failure mode analysis, and assembly design.

According to the technical status and process, the scheme design, test outline, detailed rules and test coordination, and design of large-scale system interface test and satellite model test are carried out, and the correctness and rationality of system and subsystem design, that is, the degree to which the system design of the spacecraft meets the mission requirements, are verified through the test, and the parts that do not meet the requirements are changed and verified by experiments. Finally, the technical status of the satellite flight model is determined, and the prototype development summary report, prototype reliability report, satellite flight model design report, and prototype test coverage inspection report are completed and passed the review.

2.3.3.2 General Design Process

The general design process of spacecraft system in the prototype design phase is shown in Figure 2.7. The milestone of this phase includes that the "summary report of spacecraft prototype development" has been completed and passed the review, and the satellite test coverage meets the requirements.

2.3.3.3 Description of Key Links

2.3.3.3.1 Revision of Large System Interface Specifications According to the current status of spacecraft system and individual machine development, the interface indicators between spacecraft and other systems (launch system, TT&C system, launch site system and ground application system, etc.) are coordinated to ensure the compatibility of large system interfaces, and the large system interfaces in the prototype phase need to be confirmed by signing interface control documents.

2.3.3.3.2 Revision of Spacecraft System Top-Level Design Specifications The revision of design and construction specifications in this phase is more focused on the "construction" aspect. Through the work in the scheme phase, the design principles shared by the system can be revised adaptively, and the pending and tentative indicators in the scheme phase must be clarified, so as to guide the subsequent product development.

The space environment protection design specifications in the scheme phase are adaptively revised to further clarify various types of space environment effects and

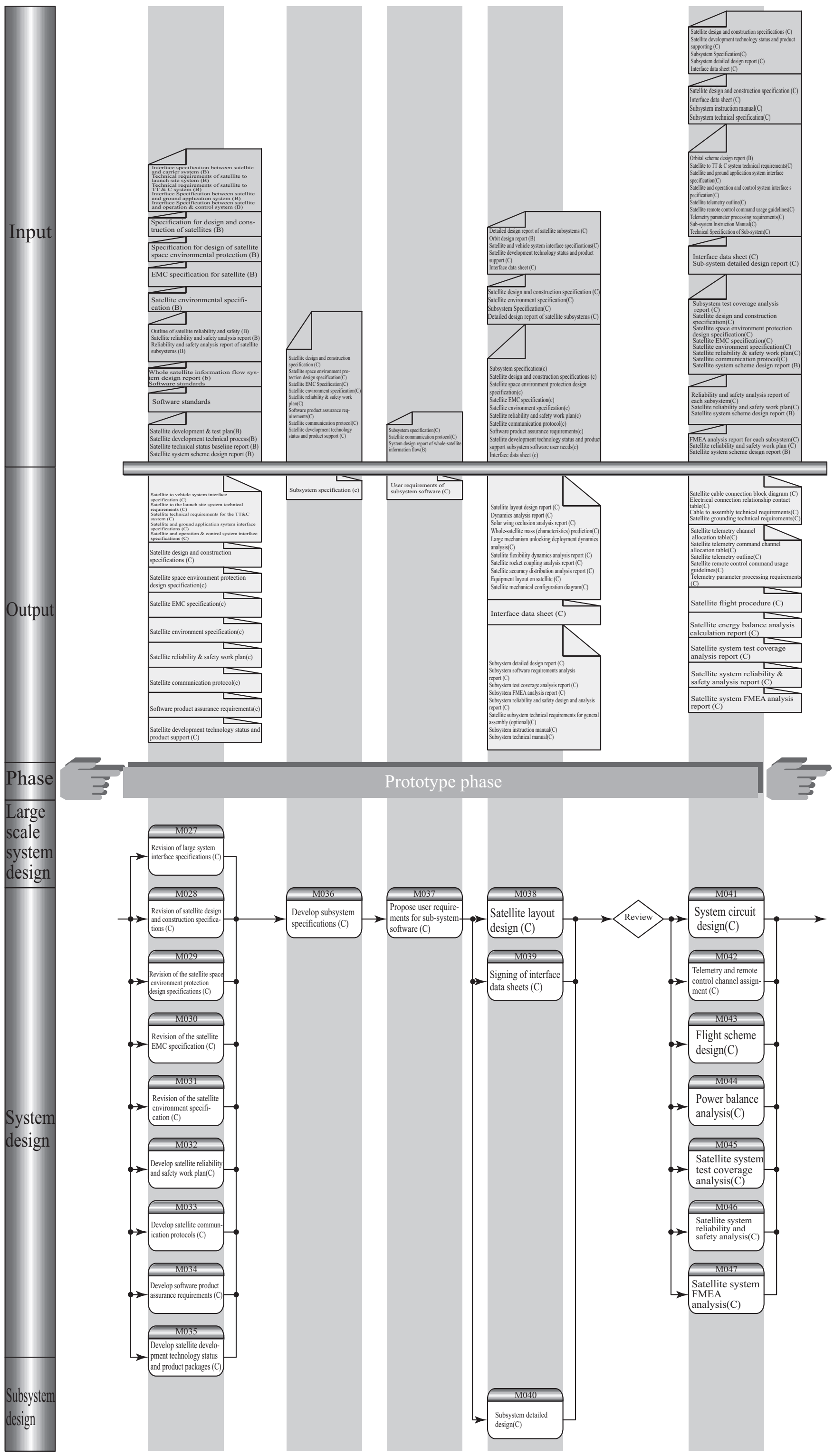


FIGURE 2.7 General flow of spacecraft system design in the prototype phase.

correspondingly clarify the space environment protection design requirements and measures for satellites, so as to effectively guide the design and analysis of space environment effect protection of satellite systems, selection, and testing of components and raw materials in the prototype phase.

Adaptive revision of the system EMC specification makes it applicable to prototype phase spacecraft development and testing. Through testing and analysis of equipment, subsystems, and system EMC, also the electromagnetic interface characteristics of the satellite during the launch phase are verified when docking with the launch vehicle and other ground support equipment, and it verifies whether the satellite achieves electromagnetic compatibility characteristics with each subsystem during the launch phase.

In the system environmental specification, the environmental test items, test conditions, test sequence, and other test requirements for the whole satellite and stand-alone unit in the prototype and flight model development phases are specified, and the environmental test matrix for the whole satellite and stand-alone unit is given to provide the basis for the whole satellite and stand-alone unit tests in the prototype and flight model phases.

2.3.3.3.3 Develop a Spacecraft Reliability Safety Work Plan Based on the reliability work items and requirements specified in the spacecraft reliability and safety outline, the timing and product levels for carrying out these work items are formulated in conjunction with the model development plan. Compared with the reliability and safety outline, the reliability and safety work plan only adds the timing and specific planning of reliability work (including input and output, starting and completion time, responsible person, and responsible unit), and the work items and their requirements are the same as the outline.

2.3.3.3.4 Development of Spacecraft Communication Protocols The development of spacecraft communication protocols in the usual sense refers only to the bus communication protocols of the whole satellite system. The purpose of developing communication protocols is to serve as the relevant subsystem agreements on the one hand, and as a basis for software design by software developers on the other hand, and also as one of the bases for software testing. In addition to introducing the basic components and physical characteristics involved in spacecraft communication, the communication protocol focuses on the description and agreement of bus operation, data format, communication terminal address, data sending and receiving mode, mode code definition, and each type of data, in order to achieve consistency and coordination in the execution and understanding of communication among subsystems and/or between the satellite and ground.

2.3.3.3.5 Develop Software Product Warranty Requirements and FPGA Product Warranty Requirements Software product assurance is to ensure that the delivered software products meet the quality standards and specifications requirements related to user activities in the software life cycle, and software product assurance requirements, the regulations and guidance on how to carry out the relevant activities, generally specify the composition and main responsibilities of the parties involved in software product development; explain the definition of software life cycle, criticality level, and scale; propose the requirements on software reliability, safety analysis and design, and software replication, solidification, reuse, procurement, and outsourcing; stipulate the technical process of software development, configuration management, handling after change, software testing, software inherited, acceptance and delivery, product maintenance, and so on; and give the reference template for the preparation of relevant development documents to standardize the relevant work of software developers.

2.3.3.3.6 Develop Subsystem Specifications and Propose Subsystem Software Requirements The purpose of developing a subsystem specification is to regulate the design activities of the subsystem. The general design principles and interface requirements common to each subsystem are defined in the system specification. In the subsystem specification, the design principles and interface requirements applicable to this subsystem are trimmed, on the one hand, and the design principles and interface requirements specific to this subsystem are clarified and defined, on the other hand.

According to the functional requirements, index decomposition, and specification requirements of the spacecraft to each subsystem, and the requirements of each subsystem to the system, the overall requirements of each subsystem software module are proposed as the design basis for software development and operation. The main contents of subsystem software user's needs are function, performance, operation environment, reliability design, interface design, quality assurance, acceptance test and delivery items, development plan and assessment nodes, etc.

2.3.3.3.7 Signed Product Interface Data Sheets and Completed Spacecraft Layout Design The interface data sheet is an important basement document for the system control of the whole satellite interface and is also an important technical input for the system design. The purpose of signing the interface data sheet is to make the system and subsystem in a unified interface state for carrying out their respective design work, so as to coordinate and match the system and subsystem design interface. The signing activities are divided into three groups, that is, mechanical, electrical, and thermal interface control groups. The final design state of each equipment and each component of the subsystem can be confirmed only after the simultaneous approval of the three interface control groups. The interface control groups should strictly review the applicable design principles and interface requirements for each equipment according to the design and construction specification requirements, subsystem specifications, etc. to ensure the coordination and matching of the whole satellite interface.

The layout design of the prototype phase is to deepen the configuration layout design of the scheme phase, to re-evaluate and analyze and design the configuration in the scheme phase according to the requirements of the spacecraft mission, the orbit situation, the large system constraints, and the equipment support of the spacecraft subsystems, the equipment status and the installation requirements of the spacecraft, etc., and to complete the spacecraft layout report, the quality (characteristics) budget, the equipment location diagram, the spacecraft layout report, mass (characteristics) budget, equipment location diagram, mechanical configuration diagram, etc., so as to provide the basis for the detailed design of the system and each subsystem.

2.3.3.3.8 Detailed Design of Subsystems The subsystem detailed design activities in the prototype phase are the deeper design activities carried out under the condition that the input conditions have been clarified. The subsystem level and the detailed design of each individual machine are carried out under the constraints of the system top-level specification, subsystem specification, signed interface data sheet, etc. During the detailed design process, more attention should be paid to the details of technical implementation, such as matching of interfaces, reliability and safety design of subsystems and equipment, environmental resistant design of devices, repairability design, derating design, and EMC design.

2.3.3.3.9 System Circuit Design and Remote Telemetry Channel Assignment The purpose of telemetry and remote control channel assignment is to ensure the telemetry and remote control resources for important functions according to the top-level task needs of the system and the demand for telemetry and remote control resources made by each subsystem while completing the functions given by the system, and the satellite system is traded off among the subsystems according to the quantity of telemetry and remote control resources and the necessity and importance level of subsystem demand, and when assigning telemetry and remote control channel, especially for important control commands, there should be targeted consideration of design redundancy to avoid the occurrence of single-point failure.

The system circuit design is the final design link of the whole-satellite electrical connection, through which the interconnection between equipment and equipment, subsystem and subsystem is realized, and the safe and reliable transfer of the whole-satellite energy and data flow is ensured. The most direct input file of the system circuit design is the whole-satellite interface data sheet. On the one hand, the system circuit design realizes the electrical interconnection between equipment and equipment, and between subsystems and subsystems on satellite; on the other hand, it solves the electromagnetic compatibility problem of the whole-satellite system, establishes the whole-satellite zero-potential reference through grounding and lap connection, ensures the reliable transmission of all kinds of signals of the whole-satellite through reasonable wiring, and realizes the scientific distribution of telemetry and remote control channels through reasonable physical mapping.

2.3.3.3.10 Flight Scheme Design The flight procedure is an important interface document between the satellite system and the ground TT&C system, which specifies all the work items of the spacecraft from the pre-launch status setting to the completion of the on-orbit test, and is the basis document for the system design work in this phase, and also the guiding document for the ground TT&C system to prepare the TT&C work plan. The flight procedures include the composition and layout of the TT&C stations and the data receiving stations, the division of the flight phase, the working mode of each subsystem, the on-orbit test mode of each subsystem, and the design of the flight procedures of the whole satellite.

2.3.3.3.11 System Design Analysis In the system design analysis, such as power balance analysis, channel link margin analysis, and test coverage analysis, the correctness of the system design is further verified.

In addition, it is necessary to carry out spacecraft system reliability and safety analysis and satellite system failure mode analysis to identify all possible failure modes at the system level, analyze the effects and causes of each failure mode, identify potential weak links, and propose possible preventive/corrective measures and on-orbit compensation measures, so as to reduce the severity of failure and/or the likelihood of failure and ensure product reliability.

2.3.3.3.12 Final Assembly Design According to the development process and technical state of the whole satellite, the integration design technology of assembling each subsystem equipment into a complete spacecraft according to the assembly plan provides the technical basis for the assembly process design and related tests of the spacecraft. The assembly design mainly includes assembly technology state and process design, instrumentation and equipment installation design, cable installation design, piping installation design, assembly technology requirements, various measurement technology requirements, and the requirements of ground support equipment and supporting content.

2.3.3.3.13 Large System Interface Test Plan Design and Verification Test Coordinating the large system interface test in the whole prototype phase, the main work includes the preparation of large system interface test outline and test details, and large system interface verification test. Verify the matching of satellite-to-rocket interface and satellite-to-ground interface, test whether each stand-alone equipment meets the design requirements, and verify whether the working modes of large-scale systems can connect normally.

2.3.3.3.14 Test and Verification of the Electric-Model Satellite Through the test and verification of the electric-model satellite, the matching of the electrical interface of the whole satellite, the matching of the satellite to ground interface, whether the electrical performance of all the equipment of the whole satellite meets the design requirements, and whether the working mode of the whole satellite design can be connected and operated normally are verified. Electric-model satellite test and verification is generally divided into different test states. For different models, different sub-test states are set up in the larger state according to the different settings of the whole satellite assembly state, TT&C state, and digital transmission state, in order to implement more targeted test work.

2.3.3.3.15 Structural-Model Satellite Test and Verification The purpose of the structural-model satellite development is to verify the mechanical properties of the satellite structure through targeted mechanical tests, so as to evaluate the rationality of the satellite structural design and provide a basis for the improvement of the prototype design. Develop a separate structural-model satellite and carry out structural-model satellite verification. Through the analysis of the test data, the dynamic characteristics and design rationality of the structural-model satellite (i.e., the satellite structure) are evaluated to provide a basis for the design improvement of the satellite structure.

In addition, after the structural-model satellite mechanics test and verification, the unlocking and unfolding test and verification is often carried out on the mechanism part to test the reasonableness of the mechanism.

2.3.3.3.16 Thermal Control-Model Satellite Test and Verification Verify the adaptability of thermal analysis models and thermal designs of spacecraft, payloads and components, check the functions and performance of thermal control products, and evaluate the problems caused by thermal deformation of structures through thermal balance tests. Through a comprehensive analysis of the test data, an evaluation of the thermal design rationality and the correctness of the thermal mathematical model can be given to provide a basis for the satellite thermal design improvement and mathematical model correction.

2.3.3.3.17 Radiation Model Satellite Test and Verification Radiation model satellite test is optional in the model development process, and mature models can work without this activity. For the first model, if there is a new antenna form application and its satellite application performance has uncertainty, the radiation model satellite verification is generally carried out. Through the radiation model star test, it is verified whether the radio frequency performance of the antenna still meets the specifications in the satellite environment, whether it meets the electromagnetic compatibility design and provides necessary data for the electromagnetic compatibility of the entire satellite system, and finally determines the star installation and layout of the antenna.

2.3.3.3.18 Summary of Prototype Development A summary of all aspects of the prototype development includes all aspects of the development of the prototype, such as the design and implementation of the development process in the prototype phase; product design and technical status change control at all levels; structural-model satellite design and verification; electric-model satellite design and verification; thermal control-model satellite design and verification; qualification parts design and verification; software design and verification; key items and critical parts design and verification; design and verification of the interface of the large system; related review; quality issues and resolution; reliability and safety verification; test coverage; and functional performance index satisfaction and verification. The prototype development summary may also include quality-related content such as quality issues and handling, which is the prototype development and quality summary.

2.3.4 Flight Model Development Phase Process

2.3.4.1 Main Content

The flight model development phase begins at the end of the prototype phase and continues until the spacecraft is shipped. The main purpose of the flight model phase work is to develop a spacecraft that can be used for launch that meets the user's mission requirements through revised design and acceptance tests.

In the flight model phase, the work is mainly carried out on the basis of fully verifying and perfecting the design work in the prototype phase or on the basis of the previous technical modification requirements and demonstration. Through the summary of the work of the prototype phase and technical improvement, the technical status of the flight model satellite is determined, which will serve as the basis for the development of the system and subsystems. With the activities of the revision of the top-level documents of the system (including large-scale system specifications, environment, and reliability) and subsystem technical requirements (or specifications), system and subsystem flight model scheme design, and design interface table signing, the technical status of the system is reflected in the design of the system and subsystems and the mutual interface, which serves as the basis for the correct adaptive modification of the design. On this basis, first, system-level adaptive modification and preparation of subsystem acceptance specifications are carried out. The system-level adaptive modification mainly includes the system circuit, remote control and telemetry, flight procedures, test coverage, power balance, reliability and safety, and final assembly design. Second, the verification test is designed for the flight model satellite; the test outline and test rules are prepared for each test; the spacecraft assembly, test, and large-scale test are carried out; the subsystem and the whole satellite development and quality summary reports are prepared; the special items of the factory review report, such as component quality summary, software summary, technical status control summary, quality return to zero summary, reliability and safety work summary, and flight control plan, are prepared, and the factory review is carried out as required.

2.3.4.2 Universal Design Process

The general system design process of the spacecraft in the flight model phase is shown in Figure 2.8. Milestone at this phase include (1) special reviews of components, software, technical status changes, reliability and safety, and quality issues reset; and (2) the spacecraft factory review.

2.3.4.3 Key Links Description

2.3.4.3.1 Revise the Interface Specification of Large System and the Top-Level Design Specification of Satellite The interface control documents between the spacecraft and other large-scale systems (launch system, TT&C system, launch site system, ground application system, etc.) are revised according to the results of the prototype development of the spacecraft, the problems in the prototype phase, and the top-level design specifications of spacecraft, such as design and construction specifications, electromagnetic compatibility specifications and environmental test specifications, are revised. The reliability and safety work plan of spacecraft are further improved and revised according to the characteristics of prototype development.

2.3.4.3.2 Revision of Spacecraft Communication Protocol and Software Product Development Requirements The communication protocol of spacecraft in the flight model phase I is mainly based on the prototype design state, and then the system-level communication protocol is agreed. In the flight model development phase, the software product assurance requirements are modified and improved mainly in response to the changes in the prototype status relative to the initial sample status to ensure the consistency of the software product assurance requirements and the satellite software design status.

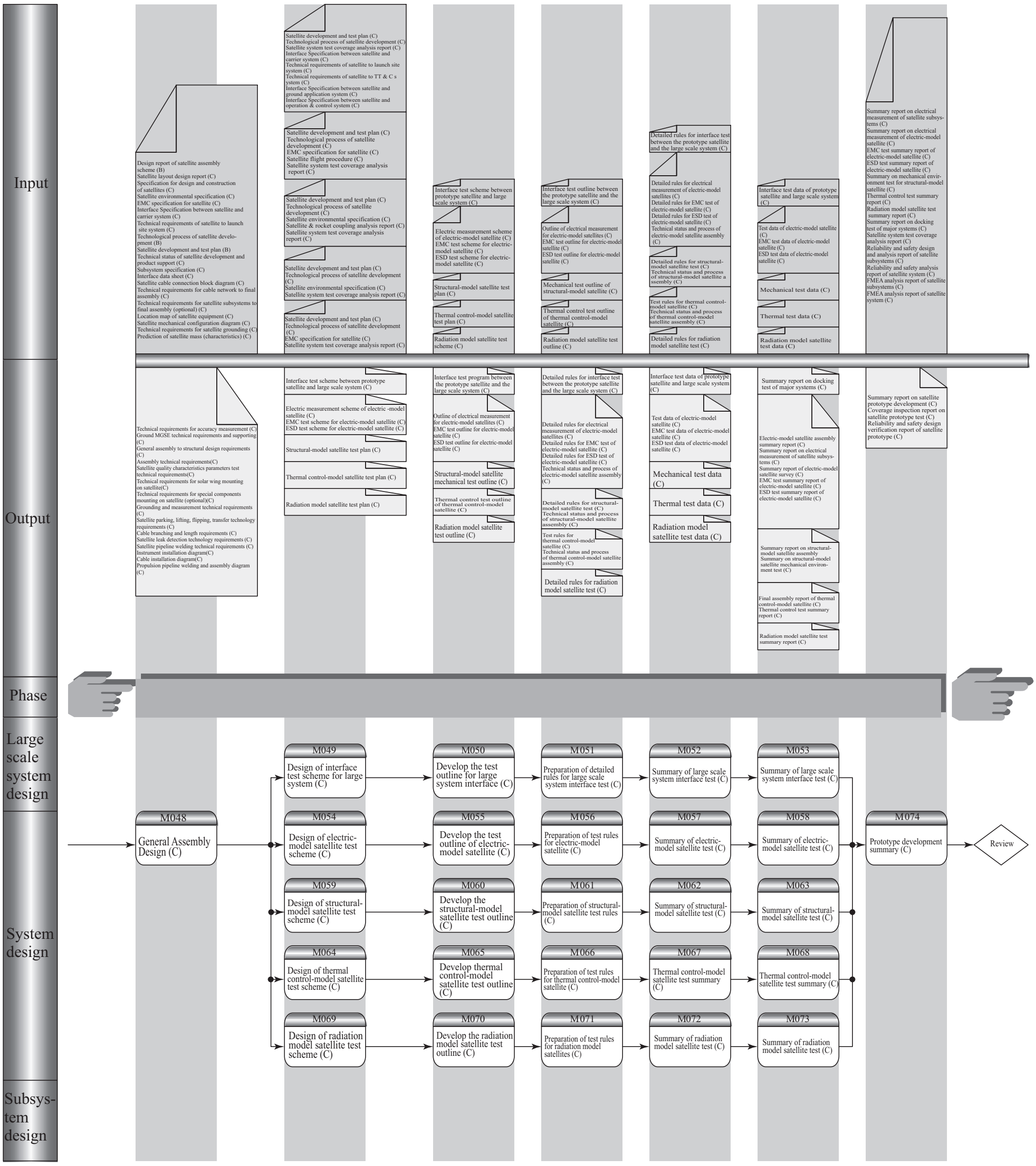


FIGURE 2.8 General flow chart of spacecraft system design in the flight model phase.

2.3.4.3.3 Develop the Technical Process of the Development of the Flight Model

The main purpose of developing the technological process of flight model development is to effectively plan, organize, guide, and control the technological state in the process of spacecraft flight model development, so as to successfully complete the development project. The actual development technology process should be planned and compiled at different levels and phases. The top-level R&D technological process is the technological planning of the whole process of system-level R&D, from which the R&D technological process of each level of subsystem is derived. The top-level R&D technological process and the derived technological process constitute the spacecraft R&D process system. It is necessary to determine the nodes of each development phase in the development technology process, and take the key nodes as the milestones of the technology process.

2.3.4.3.4 Develop Subsystem Specifications and Improve Software Requirements After the prototype test and verification, it is necessary to refine and standardize some problems existing in the prototype development, and reflect them into the flight model subsystem specification. According to the function and performance verification of the system spacecraft and each subsystem in the prototype development phase, as well as the possible changes of user's needs, the system requirements of each subsystem software are revised as the basis of the flight model software development.

2.3.4.3.5 Sign the Flight Model Product Interface Data Sheet and Modify the Satellite Layout Adaptively The flight model interface data sheet is based on the prototype interface data sheet. After the improvement of the prototype verification, the equipment interface control document is re-signed in the flight model phase. The flight model interface data sheet is the basis for the production of the equipment of the flight model. Based on the development of the prototype, the changes and corresponding analysis during the development of the prototype are reflected in the layout of the satellite, and confirmed by each subsystem and large-scale system department as the basis for the design of the structure, general assembly, and other related subsystems.

2.3.4.3.6 Flight Model Design of Subsystem In the flight model phase I, the subsystem department summarizes the development situation and quality control situation of the prototype, completely solves the quality problems in the development process of the subsystem, so as to provide support for the subsystem to turn into flight model phase. According to the system technical requirements for the subsystem in the flight model phase I, the development of the flight model and the related test and verification are carried out.

2.3.4.3.7 The System Circuit and the Adaptability of Remote Control and Telemetry Are Perfect According to the development situation of the prototype phase, the system circuit and remote control and telemetry channel should be improved, but the adjustment should be individual phenomenon.

2.3.4.3.8 Adaptive Modification of Flight Scheme Design According to the flight procedure specified in the design report of the spacecraft's flight model scheme, the prototype flight procedure is modified and refined in combination with the change of orbit change strategy, launch window, orbit entry attitude, orbiting arc, and launch time.

2.3.4.3.9 Perfect System Design Analysis According to the detailed design results of the flight model, the system design is analyzed focusing on the changes with the prototype and improving the system analysis and verification. System design analysis work includes power balance analysis, test coverage analysis, system reliability and safety analysis, and system-level failure mode analysis.

2.3.4.3.10 Adaptability Modification of General Assembly Design According to the general assembly design, implementation and test in the prototype phase, combined with the technical requirements of the system and subsystem on general assembly in the flight model phase I, the adaptability of the general assembly design is modified, which provides the technical basis for the general assembly process design, structure, and subsystem design and related tests of the flight model satellite.

2.3.4.3.11 Verification Test of Flight Model Satellite In general, the verification tests of sample satellites include the whole satellite electrical test, EMC test, mechanical test (including sinusoidal vibration test and noise test), and thermal test (including thermal balance test and thermal vacuum test). The purpose of the whole satellite electrical measurement is to verify the correctness of the electrical performance index and the matching of the electrical interface. The purpose of EMC test is to obtain the EMC test data of the satellite in the active phase, so as to judge whether the spacecraft is compatible with the carrier; to verify the electromagnetic compatibility of each subsystem on board, so as to judge whether there is electromagnetic interference in each subsystem. The purpose of the mechanics test is to check whether the performance of the spacecraft meets the requirements under the quasi-qualification or acceptance-level sinusoidal vibration environment and noise environment, and to expose the defects of the craft and quality. The purpose of the thermal balance test is to verify the correctness of the entire satellite's flight model thermal design; to provide a reference benchmark for the deflection temperature for the thermal vacuum test; to evaluate the operational performance of the spacecraft under the conditions of simulating the orbital thermal environment, especially the performance of thermal control products. The purpose of thermal vacuum test is to evaluate the working performance of spacecraft under vacuum thermal cycle temperature conditions, especially whether the performance indexes of active instruments and equipment on satellite meet the design requirements under more stringent temperature conditions than extreme orbit conditions during the whole working life of satellite, to expose the potential quality defects of instruments and equipment, so as to further improve the working reliability of instruments and equipment.

2.3.4.3.12 Large-Scale System Interface Verification Test The purpose of large-scale system interface verification test is to verify the matching of satellite and rocket interface and satellite and ground interface, to test whether each stand-alone equipment meets the interface design requirements, and whether the working mode between large-scale systems can connect normally. The large-scale system interface test of the flight model satellite includes satellite and rocket matching test, TT&C docking test, and application docking test. For different types of large-scale system interface test, it is necessary to adjust the specific large-scale system interface test content according to the different factors such as carrier type, TT&C mode, data transmission status, and whether the first satellite is launched or not, so as to implement specific test and verification work.

2.3.4.3.13 Summary of Sample Development The summary of the prototype development is to generalize and summarize the completion of the technical work, the control of the technical status, and the completely resolving of quality problems during the development of the flight model spacecraft. Its purpose is to grasp the technical status of spacecraft development process, to evaluate the quality of spacecraft development, and to provide technical support for the smooth delivery and successful launch of spacecraft.

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CHAPTER 3

Spacecraft Environment Impact Analysis

Qu Shaojie, Zhu Jiantao, Geng Liyin, and Zheng Shigui

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A spacecraft usually needs a carrier rocket to send itself into space for operation in space, so it has to experience special environments that other engineering system projects do not experience. The special environments include the carrier rocket environment, in which the spacecraft is launched, the dynamic environment, the external heat flow environment, and the space environment. As part of the environmental system, these special environments should be taken as constraints and inputs for spacecraft adaptability design (Figure 3.1).

This chapter introduces the environments experienced by a spacecraft during development and in-orbit operation, as well as their impact on the spacecraft.

3.1 LAUNCH ENVIRONMENTS AND THEIR IMPACT ON SPACECRAFT

The spacecraft is launched by a carrier rocket. Generally, the spacecraft is in the fairing of the carrier rocket and is connected to the rocket load holder. During the launch process, the spacecraft will experience a variety of environments such as force, heat, and magnetism.

3.1.1 Mechanical Environment in the Launching Process

As the mechanical environment is harsh during the launch process, the mechanical environment effect is also a unique feature of the spacecraft in comparison to general products. The mechanical environment effect is mainly manifested in structure-vibration response, which may lead to structural deformation, instability, and cracking, resulting in

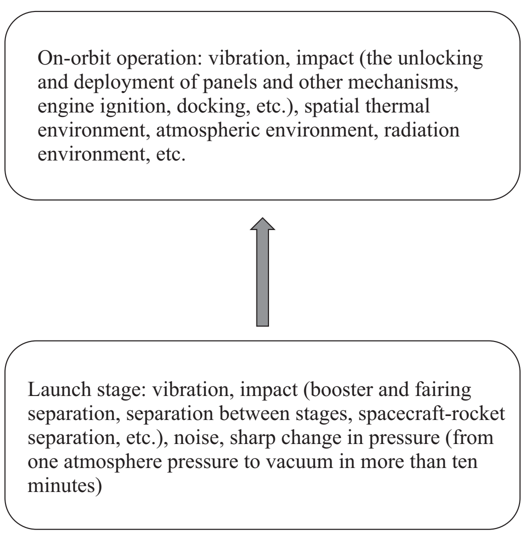

FIGURE 3.1 Environments during the launch and in-orbit operation of a spacecraft.

the loosening and fall-off of the installed devices, pipeline, and cables, the drift and out-of-tolerance of performance parameters of electronic devices and the damage and fracture of mounts and fixtures. The spacecraft structure design is mainly to overcome the influence of the mechanical environment during the launch process.

3.1.1.1 Ground Noise Environment

The spacecraft takeoff creates a complex and severe dynamic environment. As a rocket engine starts, its exhaust velocity changes dramatically in a short time. The pressure in the exhaust slot of the launch pad and the surrounding air increases swiftly, leading to asymmetric transient pulsed air pressure on the carrier rocket and tremendous noise to the carrier rocket and the spacecraft. If two or more booster rockets get involved in takeoff but ignited asynchronously, the carrier rocket and the spacecraft will vibrate laterally. If the constraints on the carrier rocket in the launching pad are not synchronous, the release will cause greater lateral load, resulting in rocket instability.

3.1.1.2 Maximum Aerodynamic Load Environment

When the speed of a carrier rocket approaches and exceeds that of sound (during transonic flight), the air around the carrier rocket will form a shock wave due to compression, and the airflow disturbance on the outer surface of the rocket will produce pressure pulsations, which will result in a severe noise environment. Furthermore, the pressure pulsations, in combination of static air pressure, steady-state wind, shearing wind and gusts, as well as steady-state acceleration overload and the force steering the boosters, will create a complex load environment.

Wind gusts and buffetings can cause low-frequency bending vibrations in the spacecraft/carrier rocket system, so that the carrier rocket will bend and vibrate like a beam. It means that the spacecraft will bear lateral inertial loads.

3.1.1.3 Steady-State Flight Environment

Except for the above-mentioned dynamic flight events, the combination of the carrier rocket and the spacecraft performs an accelerated flight under the thrust of the rocket engine in most of the launch time. During the flight, the thrust of the rocket engine is basically unchanged. However, with the continuous consumption of rocket fuel, the mass of the combination is gradually decreasing, while the acceleration of the combination is increasing. Therefore, the maximum steady-state acceleration of each stage occurs at the end of normal combustion of the rocket engine of that stage. Generally, the maximum steady-state acceleration of the entire carrier rocket occurs at the end of the flight of the first or second stage. The longitudinal acceleration of some carrier rockets can be controlled at a predetermined value by adjusting the engine thrust. The longitudinal acceleration of US space shuttles, for example, is controlled at 3 g through the adjustment of rocket engine thrust.

3.1.1.4 Stage Separation Environment

The stage separation events include the switch-off of the upper stage engine, the ignition of the next stage engine, and the separation of the two stages. When the rocket engine is switched off, the incompletely burned fuels may produce a large transient load. In addition, the release of elastic potential energy of the rocket structure may cause transient vibration during the stage separation.

The ignition shock caused by the initiating explosive device (IED) during the stage separation has practically no effect on spacecraft structure because the spacecraft is far away from the separation interface of the carrier rocket.

3.1.1.5 Fairing Separation Environment

When a carrier rocket reaches enough altitude, the atmosphere will be quite thin and the aerodynamic force and aerodynamic heat generated during flight will be very small. In this case, the carrier rocket fairing is not a necessary protection for the spacecraft but a burden. Therefore, it should be separated from the carrier rocket and discarded. The fairing will be separated by an IED, and the ignition impact on the spacecraft structure will be too small to be considered.

3.1.1.6 Environment for Spacecraft-Rocket Separation

When the spacecraft-rocket combination flies into a predetermined orbit, the connecting mechanism between the spacecraft and the carrier rocket will be released first, and then the spacecraft will be separated from the carrier rocket with the help of springs, ignition actuator rod, or small rockets. Since the release device is generally an IED, its impact on the nearby structural parts of the spacecraft should be considered.

3.1.2 Other Environments During Launch

3.1.2.1 Thermal Environment

Due to the thermal radiation on the satellite exerted by the inner surface of each section of the fairing during the rocket flight, the influence of radiative heat flux caused by the fairing should be considered while conducting the thermal design of the satellite. The typical radiative heat flux and emissivity of the fairing are shown in Figure 3.2.

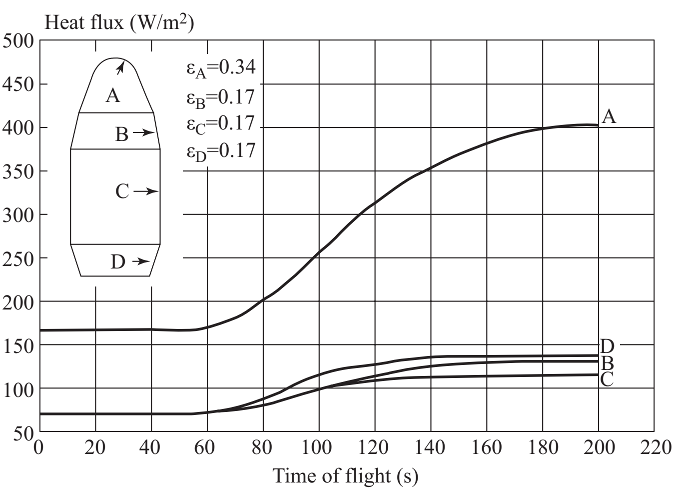

FIGURE 3.2 Typical thermal environment inside the fairing during the launch of a carrier rocket.

3.1.2.2 Pressure Environment

As a carrier rocket flies in the atmosphere, the air pressure inside the fairing is reduced through the hole on the column section of the fairing. A typical air pressure change inside the fairing is shown in Figure 3.3, where the maximum drop rate of the air pressure inside the fairing during the flight does not exceed 6.9 kPa/s.

3.1.2.3 Electromagnetic Environment

Some wireless devices on the carrier rocket and the spacecraft are working during the spacecraft launch, so the requirements on the electromagnetic compatibility between the wireless devices of the spacecraft and those of the carrier rocket should be satisfied in this process to ensure the safety of the flight.

During the process of satellite design, the sensitiveness test should be completed in accordance with the requirements on radiation sensitiveness given by the carrier rocket. Meanwhile, the emitted radiation of wireless devices of the spacecraft should meet the constraints exercised by the carrier rocket.

3.2 IN-ORBIT OPERATION ENVIRONMENTS AND THEIR IMPACT

A spacecraft in orbit is exposed to thin atmosphere, charged particle radiation, solar electromagnetic radiation, and other environments, which will have great influence on its stable operation. Therefore, it is necessary to fully analyze the in-orbit environments and take protective design measures while designing a spacecraft system.

3.2.1 In-orbit Space Environment

3.2.1.1 Sun and Its Activity

Since the sun is the source of change in the space environment of solar system, solar activities have important influences on the spacecraft. The common solar activities include sunspots, solar flares, and coronal mass ejections (CME). Sunspots are the dark areas in groups seen on solar photosphere and mainly appear at the 5°–30° solar latitudes. Solar flares are the sudden flashes that appear on a small area of solar photosphere and remain visible just from a few

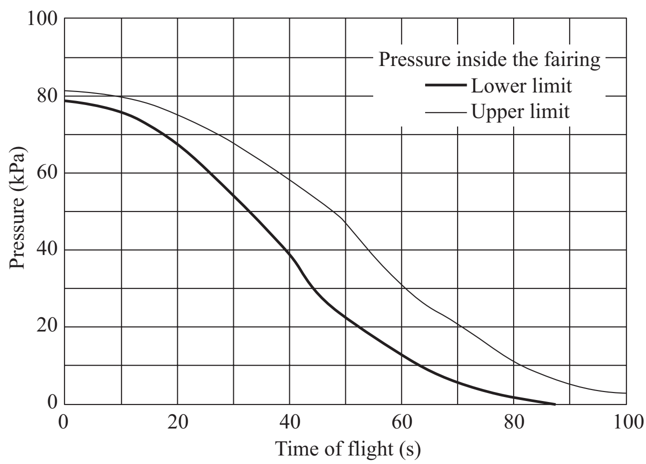

FIGURE 3.3 Typical pressure environment inside the fairing during the launch of a carrier rocket.

minutes to a few hours. CME is the phenomenon that the sun ejects a large amount of plasma cloud with relatively low energy. Solar activities have a variation cycle of 11 years on average, including about 7 years in solar maximum and 4 years in solar minimum. Solar activities also show 27-day and 13-month cyclic variations (or 15 solar rotation cycles).[1]

3.2.1.2 Near-Earth Space Environment Elements

The space environment elements that affect the spacecraft in solar-terrestrial space mainly include the Earth's atmosphere and vacuum environment, solar electromagnetic radiation environment, charged particle radiation environment, and geomagnetic field environment.[2]

3.2.1.2.1 Earth's Atmosphere, Vacuum Environment, and Their Influences The Earth's atmosphere is a unique environment faced by the Low Earth Orbit (LEO) spacecraft. The altitude range from 100 to 1000 km lies between the thermal layer and outer layer of the atmosphere. The vertical distribution of atmospheric layers at different temperatures is shown in Figure 3.4.

The atmosphere 0–50 km above the ground accounts for 99.9% of the total amount of atmosphere, and that higher than 100 km accounts for 0.0001%. The contents of N_2, O_2, Ar, and CO_2 account for about 99.997% of the total. The maximum value of atmospheric temperature appears at 15 o'clock in local time, and the minimum value at 3 o'clock. The maximum value of atmospheric density appears at 14 o'clock in local time, and the minimum value at 4 o'clock. The average lag time for upper atmosphere to respond to a magnetic storm is 6.7 hours. The atmosphere's temperature and density are the maximum in October, the second maximum in April, the minimum in July, and the second minimum in January. At the same altitude, the oxygen atom density in a solar maximum year is higher than that in a solar minimum year.

The influence of the Earth's atmosphere on a spacecraft is mainly reflected in two aspects. First, the atmospheric resistance to the spacecraft will lead to the variations in

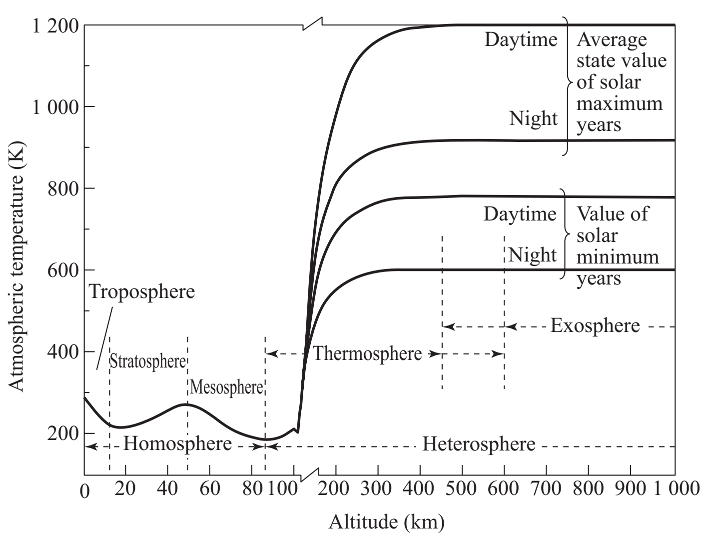

FIGURE 3.4 Schematic diagram of vertical atmospheric stratification by temperature.

the spacecraft's service life, orbital decay rate, and attitude. Second, the atom oxygen in high-altitude atmosphere will act as a strong oxidant to react with the spacecraft's surface material (for example, oxidation, sputtering, corrosion, and hollowing), thus causing mass loss, surface denudation as well as physical and chemical property variation.

Besides, the in-orbit spacecraft is also exposed to low-pressure environment (in which the pressure is much lower than that on the ground), whose impacts include low-pressure discharge, material outgassing and contamination, as well as vacuum cold welding.

Low-pressure discharge refers to the possible discharge between the two high-voltage electrodes of an on-board active device when the external atmospheric pressure is within a low vacuum range of 10^3 to 10^{-1} Pa. Low-pressure discharge is most likely to occur at an altitude of about 50 km under the pressure of about 5.7 Torr (758 Pa). The spacecraft devices that suffer slow air leakage after being powered on at the launch phrase or after entering the orbit must be carefully protected.

Micro-discharge is the phenomenon in which secondary electrons are multiplied between the two electrodes to which microwave power signals have been applied. Protective measures should be taken to avoid micro-discharges when designing the passive devices for a high-power microwave system.

When the atmospheric pressure is lower than 10^{-2} Pa, the surface of spacecraft material will release gases, including: the gas adsorbed by the material surface, the gas dissolved inside the material, and the gas penetrated into the surface of solid material.

Cold welding generally occurs under the pressure of 10^{-7} Pa or lower. On the ground, gas films and polluted films are always adsorbed on solid surfaces in contact with each other and become boundary lubricants. In a vacuum, the adsorbed film on a solid surface will evaporate and disappear, so that a clean solid surface is formed and different bonding degrees are found between solid surfaces. This phenomenon is referred to as adhesion. If the surfaces are atomically clean without oxidation film, the overall adhesion may further occur under certain pressure and temperature conditions, that is, the cold welding effect may be caused.

3.2.1.2.2 Solar Ultraviolet Radiation and Its Effects The electromagnetic radiation from the sun with a wavelength between 0.01 and 0.4 μm is known as solar ultraviolet radiation. Its energy accounts for about 8.7% of solar radiation. The relationship between solar spectral irradiance and wavelength is shown in Figure 3.5.

Solar ultraviolet radiation can cause damage to materials. The energy of ultraviolet photons with a wavelength below 0.3 μm is higher than 376.6 kJ/mol, which is strong enough to cause the break of some organic chemical bonds because the bond energy of organic polymer molecules is generally between 250 and 418 kJ/mol. As a result, the material becomes brittle, with cracks and, shrinkage appearing on its surface and mechanical properties going down. Ultraviolet radiation also severely discolors the polymer matrix and affects its optical properties. In some cases, the presence of ultraviolet radiation may further exacerbate the material erosion caused by atomic oxygen and significantly increase the mass loss. The solar ultraviolet radiation with a wavelength range of 0.01–0.4 μm is strong enough to break covalent bonds such as C-H bond, thereby affecting the performance of organics.

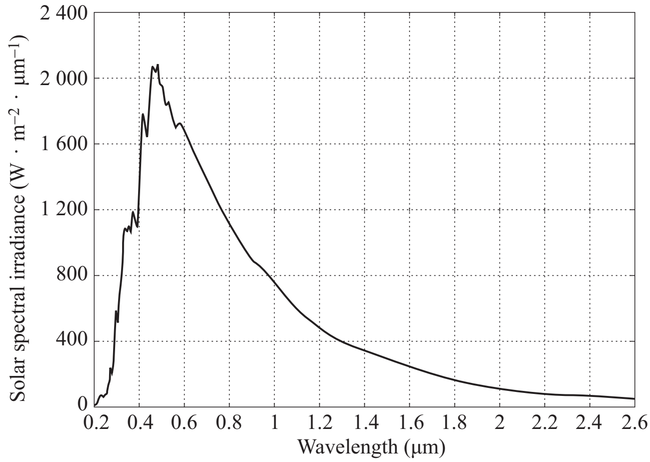

FIGURE 3.5 Relationship between solar spectral irradiance and wavelength.

3.2.1.2.3 Charged Particle Radiation and Its Effects

3.2.1.2.3.1 Earth's Radiation Belt The Earth's radiation belt refers to an area of high-intensity charged particles captured by the geomagnetic field within the near-Earth space. It is often called geomagnetically trapped radiation belt. It was first discovered by the American scientist, Van Allen, so it is also known as Van Allen radiation belt.

The formation of the Earth's radiation belt is closely related to the Earth's magnetic field. The Earth's magnetic field is similar to an eccentric dipole magnetic field. Under the compression force of the solar wind, the shape of the Earth's magnetosphere is deformed from an approximately symmetric shape into a shape whose sunlight side and shadow side are obviously asymmetric. After entering the Earth's magnetosphere, the charged particles from the sun are influenced by the Lorentz force in the Earth's magnetic field and move in three ways: spiraling along the magnetic lines of force, oscillating back and forth between the mirror points along the magnetic lines of force, and drifting caused by the Earth's rotation. In this case, the charged particles with different energies are also stably captured by the geomagnetic field in the corresponding areas around the Earth, forming the Earth's radiation belt.

The captured charged particles in space form the Earth's radiation belt structure shown in Figure 3.6. It is similar to the ring structure surrounding the Earth above the equator. Its intensity is obviously concentrated on two space areas, namely the inner radiation belt and the outer radiation belt.

The inner radiation belt is the trapped particle zone closest to the Earth, mainly composed of the trapped protons with the energy of 0.1–400 MeV, the trapped electrons with the energy of 0.04–7 MeV, and a small number of heavy nuclear ions. Its latitude boundary on the Earth's meridian plane is about ±40°, its space range is roughly $L = 1.2$–2.5 (where L represents the ratio of the distance from the intersection of the magnetic line of force in which the space point lies and the Earth's equatorial plane to the center of the Earth to the Earth's radius), and its altitude range is 600–10,000 km above the equatorial plane. Its center position varies with the energy of the particles. Generally, the center position of a

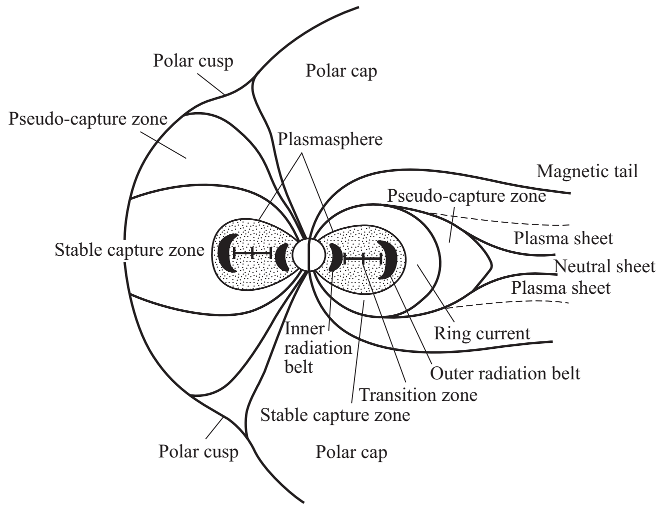

FIGURE 3.6 Schematic diagram of the structure of the earth's radiation belt.

low-energy particle zone is far away from the Earth, and that of a high-energy particle zone is close to the Earth.

The outer radiation belt is farther from the Earth than the inner one. Its latitude range on the Earth's meridian plane is about ±55° to ±70°, its spatial range is roughly $L = 3.0$–8.0, and its altitude range on the equatorial plane is about 10,000–60,000 km. In this belt, the altitude of the center intensity position of charged particles from the ground is about 20,000–25,000 km. The outer radiation belt is mainly composed of electrons and protons. However, the proton energy is usually below a few MeV, and its intensity decreases rapidly with the increase of energy. Therefore, the outer radiation belt is mainly the trapped electron belt, whose electron energy ranges from about 0.04 to 4 MeV.

As the actual geomagnetic field deviates from the dipole magnetic field, the intensity of the geomagnetic field above the South Atlantic is lower than that of the dipole magnetic field, forming a negative magnetic anomaly area with the longitude range of 20° east to 100° west, the latitude range of 10° north to 60° south and the center position of about 40° west longitude. This area is known as South Atlantic negative magnetic anomaly area. Due to the downward bending of the magnetic lines of force in this area, the altitude of the lower boundary of the inner radiation belt drops to the lowest point, and high-flux and high-energy particles can be encountered at an altitude of 200 km from the ground. Therefore, the radiation belt in this area is often called South Atlantic anomaly area. As long as a spacecraft's orbital inclination exceeds 40°, the spacecraft orbiting the Earth will continuously traverse the South Atlantic anomaly area.

At the north and south poles of the geomagnetic field, the magnetic lines of force gradually gather to form a funnel-like shape, so that the low-energy charged particles from outer space can move along the magnetic lines of force and then enter the polar region directly. The charged radiation particles with lower energy and larger flux may appear at very low altitudes in the polar regions of the Earth, which will affect the spacecraft traveling in the Earth's polar orbits.

3.2.1.2.3.2 Solar Cosmic Ray Solar cosmic ray (SCR) refers to a stream of high-energy and high-flux charged particles ejected from the sun's surface when the photosphere suddenly bursts and releases huge energy (solar flares). Since most of its charged particles are composed of protons, it is also called solar proton event. No solar cosmic ray will be emitted when the sun's surface is tranquil.

Solar cosmic ray particles, whose energy generally ranges from 10 MeV to dozens of GeV, are mainly protons. In addition, the helium nuclei account for about 3%~15%. There also exist heavy nuclei with atomic number $Z > 2$, among which the flux of the heavy nuclei with $Z = 6$, 7, and 8 accounts for 0.05% of the total particle flux.

The intensities and energy spectra of solar cosmic ray events following solar eruptions are not exactly the same. An entire solar cosmic ray event lasts from about a few hours to dozens of hours. The occurrence of solar proton events is very random, and the source of high-energy particles is limited to a local area on the Sun's surface. Besides, the particle propagation process from the sun to the Earth is strongly modulated by the solar wind and the interplanetary magnetic field. Therefore, the spatial distribution of high-energy particles is uneven and paroxysmal.

The appearance of solar flares (i.e., the solar cosmic rays) is random. Generally, more flares will appear in a solar maximum year than in a solar minimum year. Moreover, the energy spectrum and particle flux of the solar cosmic rays will vary whenever the solar flares appear. The statistics show that more proton events occur during a solar maximum year. Specifically, there are about ten events of great intensity each year, once a month on average. However, due to their sporadic nature, sometimes there is no event in a few months, and sometimes there are multiple events in a month. During a solar minimum year, fewer solar proton events – generally only three to four events a year, or even fewer – will occur.

The high-energy particles of solar cosmic rays can trigger single event effects (SEEs) in spacecraft microelectronic devices and contribute a part of the total dose to electronic components and materials. Solar flare protons are one of the major contributing factors to not only the displacement damage of solar cells but also the radiation damage of temperature control coatings and other materials.[3]

3.2.1.2.3.3 Galactic Cosmic Rays (GCR) GCR comes from the interplanetary space of the Galaxy outside the solar system and is mainly composed of the protons and heavy ions with very low flux and extremely high energy. Due to the shielding effect of the interplanetary magnetic field in the propagation of GCR particles and the close relation between the energy spectrum of the GCRs arriving at the spacecraft orbit and the strength of the interplanetary magnetic field (which is strongly influenced by solar activities), the GCRs are also affected by solar activities. Generally speaking, there is a negative correlation between the intensity of GCRs and the solar activity. In other words, during a solar maximum year, the intensity of GCRs reaches the minimum value; during a solar minimum year, the intensity of GCRs reaches the maximum value.

The GCRs contain almost all the element particles in the periodic table, but their energy spectra are different. The GCRs are composed of the charged particles with extremely low flux but extremely high energy. The energy of the particles generally ranges from 10^2 MeV to 10^9 GeV, and the energy of most of the particles is concentrated in 10^3–10^7 MeV, but the flux in free space is generally from 0.2 to 0.4 $(cm^2.sr.s)^{-1}$. The GCR particles are mainly protons, which account for about 84.3% of the total, followed by α particles (about 14.4%) and other heavy nuclei (about 1.3%).[4]

Because of low flux, galactic cosmic-ray particles contribute not much to the total dose effect of the spacecraft's electronic components and materials. Yet they may be capable of triggering SEEs in microelectronic devices due to their high energy and high linear energy transfer value.

3.2.1.2.4 Earth's Magnetic Field and Its Effects The Earth's basic magnetic field is mainly composed of the inherent eccentric dipole magnetic field (accounting for about 90%), which originated from the current system of the Earth's core. It is very stable, changing slowly during a long time at an annual rate of less than one thousandth. The basic configuration of the geomagnetic field is shown in Figure 3.7. The International Association of Geomagnetism and Aeronomy has published the International Geomagnetic Reference Field models, and released the Gaussian coefficient once every 5 years.

Geomagnetic activities include quite variations and disturbed variations. Quite variations refer to the variations of magnetically quiet day, solar day, and lunar day, and the annual variation. Disturbed variations include magnetic storm, substorm, solar disturbed daily variation, and magnetic pulsation. The geomagnetic disturbance is an important symbol of the disturbance state of the space environment.[5]

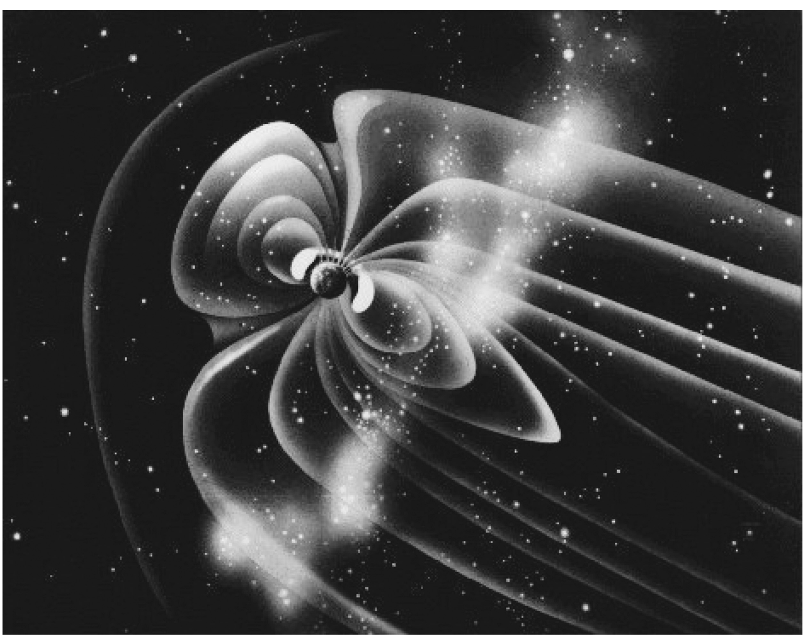

FIGURE 3.7 Configuration diagram for the earth's magnetic field.

The geomagnetic field variation has an important impact on the Earth's space radiation environment, thereby affecting the spacecraft's magnetic torque and electronic devices to different degrees.

3.2.1.3 Deep Space Environment

Deep space usually refers to the moon and the vast airspace beyond the moon. Different from the near-Earth satellites orbiting the Earth, the deep space probes performing the missions will be affected by not only the Earth's space environment, but also the deep space environments, including the interplanetary space environment and the environment surrounding or on the surface of a celestial body. For example, planetary atmospheres (of Mars, Venus, etc.) may produce resistance or aerodynamic heat when the probe is orbiting or landing; the strong radiation from Jupiter will challenge the radiation resistance of the probe; and the dust of Mars and the Moon will contaminate the landing rover's optic system; the probe may suffer extremely cold or hot environments due to the complex thermal environment on the surface of a celestial body; the soil characteristics on the surface of a celestial body will affect the soft and hard landing; and the topographic features of a celestial body's surface has an impact on the capability of the rover to travel and avoid obstacles.[6]

The space environments experienced by a deep space probe during its mission are generally divided into three phases:

3.2.1.3.1 Launch and Escape from Earth In this phase, the deep space probe is still operating within the Earth's magnetosphere (the altitude of the magnetopause is about 10 Re, where Re is the radius of the Earth), and the operating time usually lasts 5–6 hours. The space environment the probe faces in this phase is near-Earth space environment, mainly including the Earth's radiation belt, GCRs, solar cosmic rays, plasma, the Earth's atmosphere, atomic oxygen, solar ultraviolet radiation, and thermal radiation.

3.2.1.3.2 Transition Phase from the Earth to Celestial Body This phase starts from crossing the Earth's magnetopause to reaching the target celestial body. This phase lasts about several days for lunar exploration, several months for Mars exploration, and several years for other planet exploration. The space environment experienced by probes at this phase is the interplanetary environment, which mainly includes solar wind, solar cosmic rays, GCRs, thermal radiation, and solar ultraviolet radiation.

3.2.1.3.3 Celestial Body Orbiting and Surface Operation At this phase, the probe performs its target mission, including orbiting and landing on the target celestial body, and patrolling the surface of the celestial body. The operating time ranges from several days to several years. In addition to facing the conventional environment of the Sun, the probe will also face unique space environment of the celestial body, such as the atmosphere and dust of Mars, the strong magnetic field and strong radiation of Jupiter, the dense atmosphere and sulfuric acid cloud of Venus, and the dust of the moon, the topographic features of the planet surface, etc.[6]

To ensure enough adaptability of a deep space probe to the space environment, the environmental adaptability should be fully considered during the process of engineering design, R&D and manufacturing of the space probe, thereby improving its survival capability in deep space to accomplish the scheduled missions.

3.2.1.4 Micro Meteor and Debris

Although the volume and mass of most of the solid objects such as micrometeoroids and space debris are very small, they can cause various hazards to human space activities due to high speed. Mostly, small-mass micrometeoroids and space debris will erode and roughen the spacecraft's surface, causing surface material to melt and vaporize. Consequently, the thermal physical properties of the thermal control coating on surface will deteriorate (affecting the thermal balance of the spacecraft), and light transmittance of the optical surface will decrease, which can lower the efficiency of solar batteries. Large-mass micrometeoroids and space debris can cause cracks or penetration on the surface of the spacecraft, and damage the mechanical structure and seals due to their large energy. The actual measurement results show that the main hazards come from micrometeoroids and space debris with a mass less than 10^{-7} g and a diameter less than 100 μm. Because of their large quantity, their collision probability is high. Instead, the micrometeoroids and space debris with relatively large masses have a small collision probability because of their small quantity.

Because there are a small number of large-mass micrometeoroids and space debris, their collision probability is very low. Generally, their hazards can be neglected by an unmanned spacecraft. However, safe and reliable protective measures should be taken for a manned spacecraft, especially for a permanent manned space station that will keep operating in orbit for a long time in the future. In short, micrometeoroids and space debris are also one of the environmental factors that must be considered in the design of all kinds of spacecrafts. Table 3.1 lists some collisions encountered by foreign spacecrafts.

3.2.2 In-orbit Thermal Environment

In a near-vacuum environment, the heat exchange between the spacecraft and the outer space is almost entirely in the form of radiation. Meanwhile, because of the extremely small (about 10^{-5} W/m^2) radiant energy of the outer space, equivalent to a 4 K (–269°C) blackbody, the energy radiated by the spacecraft will be completely absorbed by the infinite universe. In this way, the outer space will become a heatsink (black background). The spatial thermal environments mainly include solar radiation, Earth albedo, Earth infrared radiation, etc.

3.2.2.1 Solar Radiation

Solar radiation is the strongest thermal radiation received by a spacecraft orbiting the Earth. The spacecraft thermal control mainly focuses on solar spectrum, solar intensity, and optical parallelism.

The solar spectrum involved in thermal physics mainly refers to the spectrum that has been converted into thermal energy, ranging from 0.1 to 1000 μm, which accounts for 99.99% of the total radiant energy. It is equivalent to a blackbody of 5760 K. The range of visible light wave is approximately 0.38–0.76 μm, and that of infrared rays is 0.76–1000 μm. The band

TABLE 3.1 Collisions Encountered by Foreign Spacecrafts

Spacecraft Model	Time of Occurrence	Consequences	Cause Analysis
International Sun-Earth Explorer-1 (ISEE-1)	1977.10.22 October 22, 1977	The low-energy cosmic-ray detector was damaged, and 25% of the data was lost.	The damage was caused by the micrometeoroids penetrating the window of the probe.
Solar Maximum Mission (SMM)	April, 1984	Many craters, with a diameter up to 140 μm and a perforation diameter of 80–500 μm, appear on the surface of the failed electronic circuit box of the SMM that was recovered from the space shuttle.	30% of the craters were caused by meteoroids and 70% by orbital debris.
MIR Space Station (MIRSS)	Launched on February 19, 1986	Power supply was running out.	The solar arrays were hit by micrometeoroids and debris, and oxidized by atomic oxygen.
Hubble Space Telescope	December 1993	The solar arrays were scratched and perforated.	It was hit by more than 5000 micrometeoroids in orbit over the last 4 years.
Long Duration Exposure Facility	April 1984 to January 1990	After recovery, it was found having 606 small pits with a diameter of ≥0.5 mm.	It was hit by micrometeoroids and debris.
Space Transportation System	April 1981 to May 1991	A total of 25 portholes were replaced for 40 times	It was hit by micrometeoroids and debris for 50 times.
Small Expendable Deployer System for a tethered satellite	March 10, 1994	It failed on the fourth day after launch.	A 20 km-long tether was broken by micrometeoroids and debris.

between 0.76 and 2 μm is for near infrared, the band less than 0.38 μm is for ultraviolet and Roentgen rays, and the band greater than 1000 μm is for radio waves. Most of the energy of solar radiation is concentrated in the visible and infrared bands, in which visible light (approximately 46%) and near infrared light (approximately 47%) have the biggest share.

The distance from the Earth to the sun is recorded as 1 astronomical unit (1 AU). The intensity of solar radiation, 1 AU away from the sun outside the Earth's atmosphere, is defined as the solar constant S. The solar radiation intensity outside the Earth's atmosphere is 1367 W/m^2 on average, 1322 W/m^2 at the summer solstice and 1414 W/m^2 at the winter solstice.

3.2.2.2 Earth's Albedo

The Earth's albedo is formed when the sunlight is reflected by the Earth. After entering the Earth-atmosphere system, part of the solar radiation is absorbed and part of it reflected. The percentage of the reflected energy is called the Earth's albedo. During the spacecraft thermal design, the Earth's albedo generally follows solar spectral distribution and is assumed to be diffuse reflection. The Earth's albedo is an important parameter in spacecraft thermal calculation, and the global average albedo $\alpha = 0.30–0.35$ is generally adopted for the entire spacecraft.

3.2.2.3 Earth Infrared Radiation

After the solar radiation enters the Earth-atmosphere system, the absorbed energy is converted into heat energy of the system and then is radiated to space in the form of infrared wavelength radiation. This part of the energy is called Earth's infrared radiation.

The Earth's infrared radiation is depended on the Earth, land or ocean, seasons, day and night, etc. The wavelengths of the Earth's radiation are within the 2–50 μm infrared range, peaking at 10 μm. The atmosphere is basically opaque. What can be seen from the spacecraft is the combined radiation above the atmosphere, equivalent to the blackbody radiation around 250 K. During thermal design, the average value of the Earth's albedo $\alpha = 0.30$ is adopted for calculation, so the Earth's infrared radiation is:

$$E = \frac{1-a}{4} S = 237 \text{ W/m}^2 \tag{3.1}$$

3.2.3 In-orbit Mechanical Environment

A spacecraft moving in space will inevitably be disturbed by the forces and torques exerted by the space environment. Even if the disturbance is relatively small, the orbit and attitude of the spacecraft will gradually deviate from the required nominal motion during the long-term operation. Therefore, measures should be taken to eliminate the deviation caused by disturbance. Moreover, with the development of the spacecrafts toward long service life, high reliability, large power, and high precision, space environment disturbance has become a factor that cannot be ignored in spacecraft design.

The in-orbit mechanical environments of a spacecraft include external mechanical environment and internal mechanical environment.

3.2.3.1 External Mechanical Environment

The external mechanical environment, coming from the disturbance of the space environment, includes atmospheric drag torque, sunlight pressure torque, gravity gradient torque, and geomagnetic torque. The relationship between the main magnitude of disturbance moment of space environment and the orbital altitude is shown in Figure 3.8.

3.2.3.1.1 Atmospheric Drag Torque For a LEO spacecraft, its atmospheric drag torque can be written as

$$F_d = \frac{1}{2} \rho V^2 \sum C_{Di} A_i \tag{3.2),}$$

where ρ represents the atmospheric density, kg/m^3; V is the orbital velocity, m/s; C_{Di}, is the drag coefficient of the i-th substructure; and A_i is the windward area of the i-th substructure, m^2.

The atmospheric characteristics and the interaction between the atmosphere and the object surface have been comprehensively considered in the drag coefficient, whose value

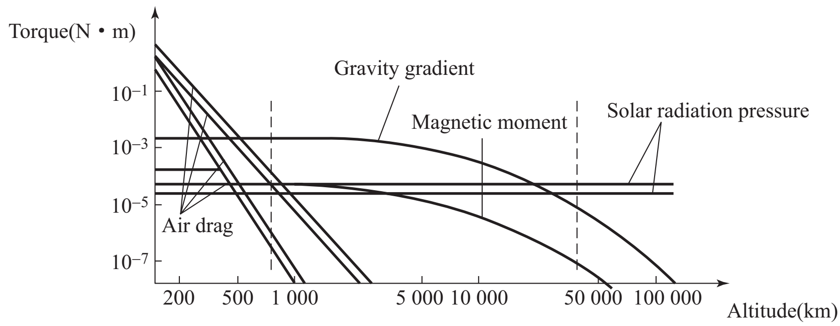

FIGURE 3.8 Spacecraft's in-orbit mechanical environment.

is obtained from wind tunnel tests and experiences, generally ranging from 2.0 to 2.5. The orbital velocity is the relative velocity of the spacecraft considering the atmospheric velocity, which can also be neglected.

3.2.3.1.2 Sunlight Pressure Torque Sunlight pressure is the result of transferring the photon momentum to objects. The energy of each photon, according to photon theory, is Planck's constant $h \times v$, where h represents Planck's constant and v represents light wave frequency. The momentum of each photon is $h \times v/C$, where h represents Planck's constant, v represents light wave frequency, and C represents the speed of light. If N photons hit the object vertically in every second and are completely absorbed by the object, then the object will gain the increment of momentum, that is, the pressure applied on it will be $N \times hv/C$. If the photons are completely reflected back, the pressure will be $2N \times hv/C$.

3.2.3.1.2.1 Solar Radiation Intensity If the intensity of the sun light perpendicularly incident on a unit area per second, that is, the intensity of solar radiation, is $E_{0,}$ then the number of the photons perpendicularly incident on a unit area per second will be E_0/hv. If the photons are completely absorbed by the object, the pressure acting on per unit area of the object will be $E_{0/}C$.

The calculation formula of solar radiation intensity is

$$E_0 = \sigma T^4 \left(\frac{R_s}{D_s} \right)^2 \tag{3.3}$$,

where

T is the temperature of black body, 5780 K;
σ is the Stefan-Boltzmann constant, 5.6704×10^{-8} W/m²/K⁴;
R_s is the radius of the sun, km;
D_s is the distance between the spacecraft and the sun, km.

3.2.3.1.2.2 Surface Optical Properties The behavior of the photons after hitting the surface of the spacecraft is very complex, including absorption, specular reflection, diffuse reflection, specular transmission, and diffuse transmission, each generating a different force.

Generally, surface optical properties are used to characterize different ways of action, which correspond to absorption coefficient, specular reflection coefficient, diffuse reflection coefficient, positive transmission coefficient, and diffuse transmission coefficient, respectively. The sum of these coefficients is 1. It is difficult to accurately obtain the optical properties of a material surface, which will change under the long-term effect of the space environment.

3.2.3.1.2.3 Sunlight Pressure Model The sunlight pressure acting on a unit surface element *ds* is

$$d\vec{F} = \frac{E_0}{C}\cos\theta\left[(1-\rho_s-\tau_s)\vec{d}-2\left(\rho_s\cos\theta+\frac{\rho_d-\tau_d}{3}\right)\vec{n}\right]ds \tag{3.4}$$

where ρ_s is the specular reflection coefficient; ρ_d is the diffuse reflection coefficient; τ_s is the positive transmission coefficient; τ_d is the diffuse transmission coefficient; θ is the incident angle of light; $\vec{d}$ is the incident direction of light; $\vec{n}$ is the unit normal vector to the surface.

The sunlight pressure torque acting on a unit surface element *ds* is

$$d\overline{M} = \frac{E_0}{C}\cos\theta\left[(1-\rho_s-\tau_s)\overline{R}\times\vec{d}-2\left(\rho_s\cos\theta+\frac{\rho_d-\tau_d}{3}\right)\overline{R}\times\vec{n}\right]ds \tag{3.5}$$

where $\overline{R}$ is the distance vector from the center of mass of the spacecraft to the center of pressure of the unit surface element.

3.2.3.1.3 Gravity Gradient Moment Suppose the Earth is the central gravitational field, the mass is M, and $\mu = GM$ is the geocentric gravitational constant. The gravitational force of the Earth acting on a satellite mass element dm will be

$$d\mathbf{F} = -dm\mu R^{-3}\mathbf{R} \tag{3.6}$$

Due to slight difference in the distances between the mass elements of different parts of the satellite and the geocenter, the resultant gravitational force (gravity) sometimes does not pass through the center of mass of the satellite, resulting in the production of a disturbance moment, which is called gravity (or gravitational) gradient torque, or gravity torque for short. Therefore, in the inertial frame, the general vector expression of the gravity gradient torque can be written as

$$\mathbf{T}_g = \int \mathbf{r}\times d\mathbf{F} = -\int \mathbf{r}\times \mu R^{-3}\mathbf{R}\,dm = -\mu\int \mathbf{r}\times\mathbf{R}R^{-3}\,dm \tag{3.7}$$

Because $\mathbf{R} = \mathbf{R}_0 + t$, r $\ll R_0$, the expansion of R^{-3} is approximately

$$R^{-3} = R_0^{-3}\left[1+\left(\frac{r}{R_0}\right)^2+\frac{2\mathbf{R}_0 \bullet \mathbf{r}}{R_0^2}\right]^{-\frac{3}{2}} \approx R_0^{-3}\left(1-\frac{3\mathbf{R}_0 \bullet \mathbf{r}}{R_0^2}\right) \quad (3.8).$$

Substituting Equation (3.8) into Equation (3.7), we can get

$$\begin{aligned}
\mathbf{T}_g &= -\mu R_0^{-3}\int\left(1-3R_0^{-2}\mathbf{R}_0\cdot\mathbf{r}\right)\left(\mathbf{r}\times\mathbf{R}_0\right)dm \\
&= -\mu R_0^{-3}\int \mathbf{r}\times\mathbf{R}_0\,dm + 3\mu R_0^{-5}\int \mathbf{R}_0\cdot\mathbf{rr}\times\mathbf{R}_0\,dm \\
&= 3\mu R_0^{-3}\int \mathbf{K}_0\cdot\mathbf{rr}\times\mathbf{K}_0\,dm \\
&= 3\mu R_0^{-3}\mathbf{K}_0\cdot\int \mathbf{rr}\,dm\times\mathbf{K}_0 \\
&= -3\mu R_0^{-3}\mathbf{K}_0\times\int \mathbf{rr}\,dm\cdot\mathbf{K}_0
\end{aligned} \quad (3.9),$$

where $\mathbf{K}_0 = -\mathbf{R}_0 / R_0$. For local orbital reference frame, $\{r\} = \left(\mathbf{I}_0 \;\; \mathbf{J}_0 \;\; \mathbf{K}_0\right)^T$, $K_0 = \left(0 \;\; 0 \;\; 1\right)^T$. The satellite's inertia tensor is assumed to be I, $\mathbf{I} = \int\left(\mathbf{r}\cdot\mathbf{r}\mathbf{E}-\mathbf{rr}\right)dm$, where $\mathbf{E}$ is the unit dyad. Then, we obtain $\int \mathbf{rr}\,dm = \int \mathbf{r}\cdot\mathbf{r}\mathbf{E}\,dm - \mathbf{I}$. Therefore,

$$\begin{aligned}
\mathbf{T}_g &= -3\mu R_0^{-3}\mathbf{K}_0\times\left(\int \mathbf{r}\cdot\mathbf{r}\mathbf{E}dm-\mathbf{I}\right)\cdot\mathbf{K}_0 \\
&= -3\mu R_0^{-3}\int \mathbf{K}_0\times\mathbf{r}\cdot\mathbf{r}\mathbf{E}\cdot\mathbf{K}_0 dm + 3\mu R_0^{-3}\mathbf{K}_0\times\mathbf{I}\cdot\mathbf{K}_0 \\
&= 3\mu R_0^{-3}\mathbf{K}_0\times\mathbf{I}\cdot\mathbf{K}_0
\end{aligned} \quad (3.10),$$

where $\mathbf{K}_0\times\mathbf{r}\cdot\mathbf{r}\mathbf{E}\cdot\mathbf{K}_0 = -\mathbf{r}\cdot\mathbf{r}\mathbf{E}\mathbf{K}_0\times\mathbf{K}_0 = 0$. For a circular orbit, $\mu R_0^{-3} = \omega_0^2$, where ω_0 is the orbital angular velocity of the spacecraft as it orbits the Earth. For an elliptical orbit, the average value is $\mu R_0^{-3} \approx \left(1-e^2\right)^{-3/2}\omega_0$, where is the eccentricity of the orbit. Therefore, relative to the local orbital coordinate system, the vector and matrix forms of the above equation are respectively

$$\mathbf{T}_g = 3\omega_0^2\mathbf{K}_0\times\mathbf{I}\cdot\mathbf{K}_0, \mathbf{T}_g = 3\omega_0^2\tilde{\mathbf{K}}_0\mathbf{I}\mathbf{K}_0 \quad (3.11),$$

where I is the inertia matrix of the spacecraft. If we introduce

$$K_0 = \begin{pmatrix} 0 & 0 & 1 \end{pmatrix}^T, \tilde{\mathbf{K}}_0 \begin{bmatrix} 0 & -1 & 0 \\ 1 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix}, \mathbf{I} = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix} \quad (3.12),$$

then Equation (3.11) will be

$$\mathbf{T}_g = 3\omega_0^2 \left(-I_{yz}\ I_{xz}\ 0\right)^T \quad (3.13).$$

This shows that when the satellite coordinate system is consistent with the local orbital reference frame, the gravity gradient torque acting on the spacecraft is only related to the values of its products of inertia I_{yz} and I_{xz}. In fact, the satellite system {b} always deviates from its reference frame {r}. For small attitude angles φ, θ, and ψ, we obtain

$$\begin{aligned} \mathbf{K}_0 &= -\cos\phi\sin\theta\, \mathbf{i} + \sin\phi \qquad \mathbf{j} + \cos\phi\cos\theta\, \mathbf{k} \\ &= \begin{pmatrix} -\theta & \phi & 1 \end{pmatrix}\begin{pmatrix} \mathbf{i} & \mathbf{j} & \mathbf{k} \end{pmatrix}^T \end{aligned} \quad (3.14).$$

By converting Equation (3.14) into the form of Equation (3.12) and substituting it into Equation (3.9), the matrix expression of the gravity gradient torque of the spacecraft relative to the satellite coordinate system can be obtained as

$$\mathbf{T}_{gb} = 3\omega_0^2 \begin{Bmatrix} \left(I_z - I_y\right)\phi - I_{xy}\theta - I_{yz} \\ \left(I_z - I_x\right)\theta + I_{xy}\phi + I_{xz} \\ -I_{xz}\phi + I_{yz}\theta \end{Bmatrix} \quad (3.15).$$

It can be seen from Equation (3.15) that the gravity gradient torque acting on the spacecraft is related to not only its product of inertia, but also the difference between the attitude angle and the moment of inertia. For the principal axis system of a satellite, the product of inertia is zero, so

$$\mathbf{T}_{gb} = \frac{3}{2}\omega_0^2 \begin{Bmatrix} \left(I_z - I_y\right)\sin 2\phi \cos\theta \\ \left(I_z - I_x\right)\sin 2\theta \cos^2\phi \\ \left(I_x - I_y\right)\sin 2\phi \sin\theta \end{Bmatrix} \approx 3\omega_0^2 \begin{Bmatrix} \left(I_z - I_y\right)\phi \\ \left(I_z - I_x\right)\theta \\ 0 \end{Bmatrix} \quad (3.16).$$

The average gravity gradient torque acting on the satellite within an orbital period should be determined in engineering applications. For an elliptical orbit with the eccentricity e and the angular velocity ω, the square of its average angular velocity is $\omega_{\sim}^{2}=\left(1-e^{2}\right)^{-3/2}\omega_0^2$ because $e>0$. Then, the above Equations (3.15) and (3.16) will become

$$\mathbf{T}_{gb\sim}=3\left(1-e^{2}\right)^{-3/2}\omega_0^2\left\{\begin{array}{c}\left(I_z-I_y\right)\varphi-I_{xy}\theta-I_{yz}\\ \left(I_z-I_x\right)\theta+I_{xy}\varphi+I_{xz}\\ -I_{xz}\varphi+I_{yz}\theta\end{array}\right\} \tag{3.17}$$

$$\mathbf{T}_{gb\sim}=3\left(1\text{-e}^{2}\right)^{-3/2}\ \omega_0^2\left\{\left(I_z-I_y\right)\phi\quad\left(I_z-I_x\right)\theta\quad 0\right\}^T \tag{3.18}$$.

3.2.3.1.4 Geomagnetic Moment

For a low-Earth orbit spacecraft, the geomagnetic torque is an important source of disturbance. Generally, the Earth's magnetic field can be approximated as a magnetic dipole. Suppose $\mathbf{B}_m$ is the magnetic field intensity vector at a certain position in the orbit and $\mathbf{m}_s$ is the residual magnetic torque vector of the spacecraft, then the magnetic torque acting on the spacecraft will be

$$\mathbf{T}_m=\mathbf{m}_s\times\mathbf{B}_m,\ \mathbf{T}_m=\tilde{\mathbf{m}}_s\mathbf{B}_m \tag{3.19}$$,

where $\mathbf{T}_m=\left(T_{mx}\quad T_{my}\quad T_{mz}\right)^T$, $\mathbf{B}_m=\left(B_{mx}\quad B_{my}\quad B_{mz}\right)^T$, $\mathbf{m}_s=\left(m_{sx}\quad m_{sy}\quad m_{sz}\right)^T$, and $\tilde{\mathbf{m}}_s$ is the antisymmetric matrix of $\mathbf{m}_s$.

The residual magnetic torque $\mathbf{m}_s$ of a spacecraft is usually measured by a zero-magnetic field remanence measurement device. For a spacecraft in a highly elliptical orbit, $\mathbf{m}_s$ can be measured in orbit by making use of the apogee region where the space magnetic field is almost zero. To reduce residual magnetic torque, the satellite remanence should be limited and treated during the processes of satellite design and manufacturing.

It is very difficult to accurately determine the Earth's magnetic field intensity $\mathbf{B}_m$. The observations show that the geomagnetic axis deviates from the Earth's spin axis by approximately 11.5°–17°. In 1975, the geomagnetic dipole deviated from the Earth's polar axis by 474.2 km on the Earth's surface. The vast majority of the Earth's magnetic field comes from the inside of the Earth, accounting for about 90% of the total magnetic field intensity $\mathbf{B}_m$ and being regarded as a uniform magnetic field $\mathbf{B}_{m0}$. The remaining magnetic field is the abnormal magnetic field generated by the ferromagnetic material of the Earth and the changing magnetic field caused by the interaction of solar wind and the upper atmosphere. $\mathbf{B}_{m0}$ is always along the tangent direction of the local magnetic field line. If the angle between the geocentric radius vector of spacecraft's location and the magnetic equator is θ_m, then the magnitude of the geomagnetic field intensity $\mathbf{B}_{m0}$ can be expressed as

$$B_{m0}=\frac{u_E}{R_0^3}\left(1+3\sin^2\theta_m\right)^{1/2} \tag{3.20}$$,

where $u_E = 8.1\times10^{25}\,\mathrm{Gs\cdot cm^3}$ (1 unit of electromagnetic system=1 Gauss·cm^3) is the component of the magnetic torque of the geomagnetic field along the magnetic axis, and θ_m is the geomagnetic latitude. The geomagnetic field intensity is $B_{m_0} = 0.311\,\mathrm{Gs}$ at the equator of the Earth and $B_{m_0} = 0.622\,\mathrm{Gs}$ at the magnetic pole. A magnetic field with a magnetic field intensity of 1 Gs acts on 1 unit magnetic pole to generate a force of 1 dyne. The relationship between different units of magnetic field intensity is $1\mathrm{T} = 1\ \mathrm{Wb/m^2} = 10^4\,\mathrm{Gs} = 10^9\,r$. The unit of magnetic torque can be (Gauss·cm^3 [$\mathrm{Gs\cdot cm^3}$]), (Ampere·m; [$\mathrm{A\cdot m^2}$]), (Dipole torque·cm [$\mathrm{Pole\cdot cm}$]). When the unit of magnetic torque is $\mathrm{A\cdot m^2}$, the unit of B_{m_0} is T or $\mathrm{Wb/m^2}$ and that of T_m is $\mathrm{N\cdot m}$; when the unit of magnetic torque is $\mathrm{Pole\cdot cm}$, the unit of B_{m_0} is Gs and that of T_m is $\mathrm{dyn\cdot cm}$. 1 ampere·m^2 = 1000 electromagnetic unit of current·cm^2 = 1000 dyne·cm/Gauss electromagnetic unit of current(emu) 1 ampere = 0.1emu = 0.1dyn1/2(current of Gaussian units) 1 Gauss = 1dyn1/2/cm 1 ampere·m^2 = 0.1dyn1/2·$(100cm)^2$ = 1000dyn1/2·cm^2 = 1000dyn·cm/(1000dyn1/2·cm) = 1000dyn·cm/Gauss Magnetic moment of Gaussian units: 1A.m^2 = 1000dyn1/2·cm^2 = 1000emu dyn1/2·cm^2 = emu, dyn1/2/cm = emu/cm^3, 1 Guass.cm^3 = 1emu. The above-mentioned units and their conversions are vital while adopting $\mathrm{cm\cdot g\cdot s}$ or $\mathrm{N\cdot Kg\cdot m}$ systems for calculation.

3.2.3.2 Internal Mechanical Environment

The internal mechanical environment of an in-orbit spacecraft includes the impact load of the spacecraft accessories (such as the solar wing and antenna) under deployment and locking, the load generated when the spacecraft's attitude and orbit control engine is in operation, and the microvibration produced by the operating moving parts on the spacecraft (such as momentum wheel), and the mechanical interference caused by liquid sloshing inside tanks.

3.2.3.2.1 Impact Load due to Deployment and Locking of the Accessories During the launch phase, the large components to be deployed are in a folded state. After the spacecraft enters the predetermined orbit, these components need to be deployed and locked. At this time, a locking impact load will be generated. If the deployed components are particularly sensitive to impact loads, a dynamic analysis is required to estimate whether the impact load of deployment and locking can satisfy the requirements so as to avoid subversive problems.

3.2.3.2.2 Operating Load Generated by Attitude and Orbit Control Engines All pulse thrusts generated by orbit control engines (such as 490 N engine) in trial injection and orbit change will lead to an acceleration of the spacecraft. They, coupled with the power amplification effect, often result in severe deformation and stress of the spacecraft. Great importance should be attached to this impact, especially for large flexible accessories. For example, if the solar arrays have been deployed during the flight, large bending torques and shear forces will be produced by solar array drive assemblies (SADA) (connecting solar arrays to the spacecraft body) under the action of orbit control engine. In severe cases, the root hinge of the solar arrays or SADA may be damaged.

3.2.3.2.3 Microvibration In-orbit microvibration or disturbance mainly affects the payload of remote sensing satellites, laser communication satellites, and microgravity experiment satellites. Such satellites generally have a variety of movable parts such as momentum wheels, gyroscopes, solar array driving mechanisms, and antenna driving pointing mechanisms. The high-speed movement of these movable parts and any microvibration of large flexible accessories will cause the satellite and its remote sensor to vibrate, which degrades the imaging quality of remote sensors (especially high-resolution remote sensor).

3.2.3.2.4 Liquid Sloshing The sloshing characteristics of the liquid fuel in the on-board storage tank have a powerful influence on the spacecraft's dynamic characteristics and the control-system's stability. The engine's on/off switch during orbit change maneuver, the maneuver at large attitude angles, and the impact during rendezvous and docking will cause the liquid fuel in the storage tank to slosh gently or violently.

3.3 SPACE ENVIRONMENTAL EFFECTS ON SPACECRAFT AND ITS PROTECTION DESIGN

3.3.1 Space Environmental Effects

A variety of environmental effects will arise due to the interaction between space environmental elements and the spacecraft and will affect the function and performance of a space mission.

The resistance of the atmosphere to the spacecraft will lead to changes in the service life, the orbital decay rate, and the attitude of a spacecraft. The atom oxygen in high-altitude atmosphere will act as a strong oxidant to react with the spacecraft's surface material (for example, oxidation, sputtering, corrosion, and hollowing), thus causing mass loss, surface denudation as well as physical and chemical property variation. Solar ultraviolet radiation may cause the outer surface material degradation of the spacecraft system. The space-charged particles, such as electrons, protons, and heavy ions, may cause a variety of radiation effects on on-board electronic devices and materials, which typically includes total ionizing dose effect, SEE, displacement damage effect, surface charging and discharging effect, and internal charging effect.[7]

Because of the different spatial distributions of various environmental elements, the difference among space environmental elements and space environmental effects should be considered during the development of the spacecraft to be sent in different orbits. Spatial environmental elements and their impacts are shown in Table 3.2.

3.3.1.1 Atomic Oxygen Erosion Effect

In a low-Earth orbit, oxygen molecules (O_2) in the neutral atmosphere are irradiated by the sun's ultraviolet rays and then produce photoionization, generating oxygen (O) in an atomic state, i.e., atomic oxygen, which has strong oxidizing properties. When the atomic oxygen interacts with the materials on outer spacecraft surface, a strong denudation effect will take place on the material. The consequences include physical/chemical property deterioration of the material surface, surface denudation, pollution, glow, and even loss of material function.

3.3.1.2 Solar Ultraviolet Radiation Effect

The solar electromagnetic radiation environment that a spacecraft faces includes the X-rays, ultraviolet radiation, visible light, infrared radiation, and radio waves originated

TABLE 3.2 Spatial Environmental Elements Involved in Aerospace Engineering

Spatial Environmental Elements	Objects Involved	Impacts
Thermal environment	Thermal control subsystem and the whole spacecraft	Thermal design
Ultraviolet radiation	Surface functional materials	Performance degradation
Atomic oxygen	Functional material of LEO (300–500 km) spacecraft surface	Material denudation
Neutral atmosphere	Attitude and orbit control subsystem and propulsion subsystem (90–2500 km)	Atmospheric drag
Vacuum	The whole spacecraft	Vacuum discharge, material outgassing, pollution, volatilization, denaturation, cold welding
Micrometeoroids/orbital debris	The whole spacecraft and structural subsystem (manned spacecraft)	High-speed impact and structural damage
Geomagnetic field	Attitude control subsystem and the whole spacecraft	Magnetic torque, influential radiation environment
Plume contamination	Optical system, solar cells, attitude control, communication subsystem	Pollution, mechanics, electrics
Ionosphere	Communication subsystem and navigation function	Radio wave delay, flicker, and interruption
Trapped protons and electrons	All components and materials	Total ionizing dose effect
Trapped protons and solar protons	Optoelectronic devices, solar cells, bipolar devices	Displacement damage effect
High-energy protons and heavy ions	Logic device, CMOS device, and MOSFET	SEE
Low-energy plasma	Electronic devices and optical lens	Surface charging/discharging effect
High-energy electrons	Electronic devices	Internal charging effect

from the sun. Among them, solar ultraviolet radiation may cause outer surface material degradation; other electromagnetic environments may produce background noises and stray light interferences in the detection system with wireless communication, optical sensors, and optical cameras.

As the spacecraft moves in and out of shadows repeatedly, the change of the spacecraft attitude also causes uneven or changing outer-surface sunlight. Since the spacecraft's surface is intermittently irradiated by sunlight, the influence of the solar ultraviolet radiation on the spacecraft's outer-surface material should be considered.

3.3.1.3 Charged Particle Radiation Effect

3.3.1.3.1 Total Ionizing Dose Effect The total ionizing dose effect will affect all electronic components and nonmetallic materials in a very complicated mechanism. Enormous difference lies in the effect mechanisms of total ionizing dose for various components and materials. Take the Metal-Oxide-Semiconductor (MOS) microelectronic devices currently widely used in spacecraft as an example. The damage caused by charged particle radiation is generally believed to be the result of the generation of interface states and of fixed positive charges in the oxide layer.

The incidence of charged particles will generate a certain number of new interface states on the Si-SiO_2 interface of the MOS device. At the same time, due to the radiation annealing effect, the new interface states will experience a disappearing process. The final interface state formed after radiation will affect important electrical parameters such as the mobility and service life of charge carriers in electronic devices, and then the electrical performance of the devices.

Meanwhile, the incidence of charged particles can ionize the gate oxide layer of the MOS device and generate a certain number of electrons and positive ions in the gate oxide layer. While the electrons drift under the gate electric field, fixed positive charges are left in the gate oxide layer. These fixed positive charges in oxide layer directly affect the threshold voltage of the MOS device, causing the changes in the turn-on characteristics of the device and the loss of current control of the device in severe cases.

Performance drift, malfunction, even completely failure or damage may happen to electronic components and functional materials of the spacecraft due to total ionizing dose effect. The typical damage to electronic components and materials are as follows:

a. The lowered current amplification coefficient of bipolar transistor, increased leakage current, and decreased reverse breakdown voltage
b. The dropped transconductance of unipolar devices (MOS devices), drifted threshold voltage, and higher leakage current
c. The enlarged input offset of the operational amplifier, decreased open-loop gain, and changed common-mode rejection ratio
d. The increased dark current and background noise of optoelectronic devices and other semiconductor detectors
e. The deviated electrical performance parameters of logic devices such as Central Processing Unit (CPU) and its peripheral chips, and the final errors or even loss of logic functions
f. The darkened glass materials
g. The strength degradation, cracking and crushing of insulating and dielectric materials (such as wire sheaths, polymer materials)
h. The cracking, peeling off, and thermal parameter declining of thermal control coating
i. The viscosity loss of adhesive materials and the falling-off of fixtures

3.3.1.3.2 Single Event Effect Targeting on electronic components, the SEE is the radiation effect generated by the incidence of a single high-energy proton or heavy ion on an electronic device. In terms of effect mechanisms, these effects can be divided into multiple types, such as single event upset (SEU) and single event latch-up (SEL), single event burnout (SEB), single event gate rupture (SEGR), as shown in Table 3.3.

TABLE 3.3 Possible SEEs in the Devices with Different Techniques and Types

Type of Device	Technique	Function	SEU	SEL	SEB	SEGR
Transistor	Power MOS				√	√
Integrated circuit	CMOS	SRAM	√	√[a]		
	BiCMOS	DRAM/SDRAM	√	√[a]		
	SOI	FPGA	√	√[a]		
		EEPROM		√[a]		
		Flash EEPROM				
		A/D converter	√	√[a]		
		D/A converter	√	√[a]		
		Microprocessor/Microcontroller	√	√[a]		
	Dipole		√			
Optoelectronic devices		Optical coupler				
		CCD				

[a] The devices with SOI and SOS technologies are excluded.

SEU is a charged particle radiation effect that occurs in logic devices and logic circuits with monostable or bistable states. The charge accumulation effect (funnel effect) is one of the major theoretical models to explain the SEU. When a single space high-energy charged particle bombards the chip of a large-scale or ultra-large-scale logic microelectronic device, an ionization effect will occur in the area near the Positive-Negative (PN) junction inside the chip along the incident trajectory of the particle, generating a certain number of electron-hole pairs (charge carriers). If the chip is in a power-on state, the carriers generated by radiation will drift and be redistributed under the action of the electric field inside the chip, thereby changing the distribution and motion state of normal carriers inside the chip. When the change is large enough, the electrical performance state of the device will change too, resulting in logic errors in the logic device or circuit, such as the flipping (between "0" and "1") of the data stored in the memory unit, which subsequently causes data processing errors, circuit logic malfunction, computer instruction flow disorder, and program "runaway". The least damage may be errors in all kinds of monitoring data of the satellite; the worst damage may cause the satellite to execute incorrect instructions and behave abnormally and even break down, putting the satellite in a catastrophic situation.

SEL is a very harmful space radiation effect that occurs in bulk silicon Complementary Metal Oxide Semiconductor (CMOS) devices. Limited by the manufacturing techniques, bulk silicon CMOS devices have an inherent four-layer PNPN structure, which forms a parasitic silicon-controlled rectifier. To latch up a CMOS device, the following conditions must be satisfied: a certain trigger signal is needed; the parasitic bipolar junction transistors are forward-biased, and the product of the two current amplification factors is greater than 1, that is, $\beta_{PNP} \times \beta_{NPN} > 1$; the power supply should be able to provide enough current to maintain the latch-up state. When the charged particles bombard a CMOS device, a large number of electron-hole pairs will be ionized out along the particle trajectory. As a large number of these carriers are collected by the sensitive PN junction in the chip through drifting and diffusing, a latch-up trigger signal may be formed. If the other two conditions that can enable the latch-up of CMOS devices are available at the same time, the CMOS devices will be latched up.

SEB is a space radiation effect that mainly occurs in power devices. The charge avalanche multiplication mechanism is one of the key theoretical models to explain SEB effect. If the PN junction of the device is in a forward-biased state under the incidence of high-energy particles, a charge accumulation effect will occur; if the PN junction is reverse biased, a large number of electron-hole pairs will be produced by ionization effect along the particles' incidence trajectory. Under the effect of a strong reverse-biased electric field inside the PN junction, these electron-hole pairs will quickly separate and drift in opposite directions along the direction of electrical wires, respectively. The avalanche multiplication effect of carriers is easily generated when the carriers drift and accelerate in the electric field. Once the carriers are multiplied in an avalanche manner, the reverse breakdown of the PN junction will occur, thereby triggering the SEB effect.

SEGR is also one of the space radiation effects that occurs in power MOS devices. For a MOS device in normal operation, when the incident high-energy charged particles (M) in space penetrate its grid, gate oxide layer (O) and device substrate (S), they will, along the incident trajectory of the particles, generate ionization effects in the gate oxide layer (usually SiO_2) and on the Si-SiO_2 interface to form a large number of electrons and positive ions. Meanwhile, a plasma low-resistance conducting channel from grid to substrate is formed in the originally insulated gate oxide layer along the particles' incident trajectory. In addition, an instantaneous current is generated in the conductive channel under the action of the grid voltage. When this instantaneous current is large enough, breakdown will occur along the current path in the gate oxide layer of the device, forming a permanent conductive channel from the grid to the substrate and causing the device to fail completely.

3.3.1.3.3 Displacement Damage Effect Displacement damage effect (also called non-ionizing dose damage) is a long-term cumulative damage effect caused by energetic particles, which will affect the performance of optoelectronic devices and bipolar devices. The incidence of space-charged particles on spaceborne electronic components and materials will cause not only the total ionizing dose generated through ionization but also displacement effect, that is, the incident high-energy particles bombard the atoms of the absorber and make them move away from their original position in the crystal lattice. Consequently, the lattice defects occur, and the electronic components and materials are damaged. The displacement damage effect will change the service life of minority carriers in semiconductor materials and speed up the discoloration of crystal optical materials and the light absorption.

For a small number of carrier devices (such as optoelectronic devices, bipolar devices, and solar cells), displacement damage will affect their performances. For example:

a. The current gain of bipolar devices decreases, especially at low currents (the PNP devices are more sensitive to displacement damage than NPN devices).

b. The leakage current of diodes increases, and so does the voltage drop in forward conduction.

c. The charge transfer efficiency (CTE) of charge coupled device (CCD) devices decreases, while the dark current and hot spots increase.

d. The output power of the light emitting diode (LED) drops.

e. The short-circuit current, open-circuit voltage, and output power of solar cells (Si, GaAs, etc.) decrease.

3.3.1.3.4 The Charging and Discharging Effect Occurs on the Surface The spacecraft is immersed in space plasma during its in-orbit operating. Under the interaction between the plasma and the spacecraft surface material, the charging and discharging effect will occur on the spacecraft's surface.

The surface of the spacecraft is charged in two ways, i.e., absolutely charged and relatively charged. "Absolutely charged" means that the spacecraft has a certain electric potential relative to the plasma "ground" in space. "Relatively charged" means that due to the different dielectric properties, sunlight conditions, and geometric shapes of the outer surface of the spacecraft, an electric potential difference lie between the spacecraft's adjacent outer surfaces, between the surface and the deep layer, and between the spacecraft's surface and its ground. When the potential difference reaches a certain magnitude, electrostatic discharge (ESD) will occur in the form of corona, arcing, sparkover, etc., and electromagnetic pulses will be radiated. Alternatively, the discharge current will be directly coupled or injected into the electronic system of the spacecraft through the spacecraft's structure and grounding system, which can cause a negative impact on on-board electronic systems and even lead to a circuit failure that directly threatens the safety of the spacecraft.

When a high-power solar array power system is installed on a spacecraft, the coupling probability between the system and the space plasma environment will increase with the operating voltage of the high-voltage solar array, and then the secondary discharge effect will occur due to space ESD. When the potential difference between the wired panels of high-voltage solar arrays is higher than the threshold voltage, a relatively long-lasting current will flow between the adjacent panels after the initial discharge of the triggered static electricity. This phenomenon is called secondary discharge or continuous discharge. The energy of the secondary discharge is provided by the high-voltage solar array power system, which may lead to short-circuit damage to the wired panels of solar array.

A Highly Elliptical Orbit (HEO) spacecraft that experiences an altitude range of 20,000–65,000 km, a LEO spacecraft that flies through polar region, and a LEO spacecraft that uses high-voltage buses will all be affected by surface charging/discharging effect. Even though the surface charging/discharging effect is exerted directly on the spacecraft's outer-surface materials, the electromagnetic signals generated by discharge may affect all spacecraft components.

The ESD caused by space plasma may occur in the following situations:

a. The spacecraft is passing through the shadow region. Without sunlight, the plasma charging potential between the spacecraft surface and the plasma ground may be as high as thousands of volts. It is a dangerous period in which the spacecraft is exposed to high-frequency ESD. When the spacecraft enters in the plasma cavity in the orbit, ESD can occur between the spacecraft surface and the plasma ground. The uneven conductive property of surface material can cause ESD among different parts of the spacecraft surface. ESD can occur between the spacecraft surface and the spacecraft ground when the surface grounding system is not well developed.

b. The spacecraft is moving out of the shadow region. When a spacecraft is in shadow region, the possible charging potential of its surface is as high as thousands of volts. As it moves out, a part of its surface sun-shined by sunlight will be charged positively (generally with a voltage from several volts to tens of volts) due to the effect of photogenerated electrons. As the unshined part of the spacecraft surface still has a negative potential up to thousands of volts, a huge difference of electrostatic potential exists between the shined part and the unshined part. Therefore, ESD may occur to a large extent. The in-orbit monitoring data of spacecrafts all over the world show that the period from local midnight to early morning is the period that ESD occurs frequently on the spacecraft surface.

c. The spacecraft is exposed to sunlight. For a three-axis stabilized spacecraft, even after it completely enters the sunlight area, it still has a sunlight side and a shadow side. On the shadow side, the space plasma can still charge the spacecraft surface to a high-level negative potential; on the sunlight side, the charging potential is very low or even a relatively low positive potential due to the emission of photogenerated electrons. As a result, there may be a high electrostatic potential difference between the sunlight side and the shadow side, thus ESD may occur between these two sides.

d. The surface conductivity is uneven. The uneven electrical conductivity of the spacecraft surface leads to unequal electrostatic charging between different surfaces of a spacecraft. In other words, there is a difference between the charging potentials in different areas of a spacecraft's surface, resulting in ESD between different areas.

e. The spacecraft experiences a geomagnetic substorm. A large amount of hot plasma from the sun is injected into the low-Earth orbit, and the spacecraft surface is more likely to be charged to a higher negative potential, so this period becomes a dangerous period with a high frequency of ESD on the spacecraft.

3.3.1.3.5 Internal Charging Effect In large radiation environment disturbance events (such as solar flares, CME, geomagnetic storms, or geomagnetic substorms), a large number of high-energy electrons can be injected into the geosynchronous orbit or even the low-altitude Sun-synchronous orbit to dramatically increases the electron flux greater than 1 MeV in the Earth's radiation belt. If existing for a long time, these electrons are capable of directly penetrating the satellite skin (including the outer conductive surface and insulating materials), satellite structure, and device housing, as well as deep insulating media (such as the circuit boards embedded in the satellite and the wire insulation layer). In this case, the charges will accumulate in the deep layers of insulating media (such as circuit boards and coaxial cables), resulting in the charged state of deep layers. This is the so-called internal charging effect.

The electron energy that cause charging in deep layers of the media ranges from 100 keV to several MeV. An electron with an energy of 2 MeV can penetrate a 5-mm aluminum plate. As high-energy electrons are continuously injected, embedded in the insulating material, and quickly accumulated, the electric field inside the material will increase as long as the charge accumulation rate exceeds the natural discharge rate of the material. When the field strength is strong enough, the insulating material will be ruptured, resulting in internal ESD (IESD) to directly interfere with the electronic system. In severe cases, dielectric breakdown and short circuit may be caused. Although the internal charging effect generally occurs in dielectric materials and isolated conductors, the IESD generated through internal charging may affect all electronic components of the spacecraft.

3.3.2 Requirements for Atmospheric and Vacuum Environment Protection Design

1. The electronic device that is turned on in the active launch phase should pass the vacuum discharge test to verify its capability of withstanding low-pressure environment during the prototype and flight model phases.
2. For the spacecraft components with relative motion, their ability to move in an ultra-high vacuum environment should be confirmed in the planning and prototype phases. Those that cannot be confirmed should be verified through vacuum dry friction and cold welding tests.
3. High-power microwave devices should be subjected to micro-discharge test.
4. The effects of vacuum environment on materials, such as outgassing, evaporation, sublimation, decomposition, and outgassing-induced surface pollution, should be considered during spacecraft design.
5. As a general requirement, the loss of the total mass of a material should be less than 1%, and the amount of condensed volatile matter less than 0.1%. Besides, the use of low-temperature sublimation materials (such as cadmium plated and zinc plated materials) is avoided.
6. Using the same metal for mating and applying solid lubricants to the contact surface shall be avoided to prevent cold welding.
7. For a LEO spacecraft, atomic oxygen environment analysis should be carried out to calculate the accumulated atomic oxygen flux during the spacecraft's lifecycle.
8. For the stand-alone products in a LEO spacecraft, the outer-surface material with enough resistance to atom oxygen denudation should be chosen according to the result of atomic oxygen flux analysis. Atomic oxygen test should be conducted when it is impossible to determine whether the functions and performance of the materials meet the requirements after atomic oxygen denudation.

3.3.3 Requirements for Solar Ultraviolet Radiation Protection Design

1. During the planning and prototyping phases of spacecraft development, an analysis of ultraviolet radiation effect should be carried out to calculate the accumulated ultraviolet radiation flux during the spacecraft's lifecycle.
2. For stand-alone products, the outer-surface materials with sufficient ultraviolet radiation tolerance, including organic materials, polymer materials, optical materials, film materials, adhesives, and coatings, should be selected according to the analysis results of solar ultraviolet radiation flux during the design and prototyping phases. When the ultraviolet radiation resistance of a material cannot be determined, it should be verified by ultraviolet radiation test, which should cover the spectral range from 10 to 400 nm.

3.3.4 Requirements for Charged Particle Radiation Protection Design

Requirements for the protection design of total ionizing dose effect[8]:

1. Requirements for analysis of total ionizing dose effect
   a. At the planning and prototyping phases of spacecraft development, the total ionizing dose effect analysis of spacecraft orbit should be carried out to draw an in-orbit dose-depth curve based on the one-dimensional solid ball shielding model. In the prototyping phase, an analysis on total ionizing dose effect should be carried out according to the needs of model development through the overview or detailed 3D shielding modeling of the whole spacecraft to obtain a dose-depth curve at the location of typical devices.
   b. The designer of the stand-alone product should, based on the one-dimensional analysis of the total ionizing dose provided by the System Engineering at the planning and prototyping phases, complete the analysis of the total ionizing dose effect related to the design. This task is basically to analyze the total ionizing dose of key parts or components inside devices, and then calculate the radiation design margin (RDM). If the obtained RDM cannot meet the System Engineering's requirements, the analysis on the total ionizing dose effect inside the stand-alone product can be conducted after the System Engineering provides a preliminary or detailed three-dimensional analysis on total ionizing dose, and then the RDM can be given. If the RDM obtained through the above analysis still fails to meet the System Engineering's requirements, protective measures should be taken.
   c. The RDM is defined as follows:

$$\mathrm{RDM} = \frac{D_{\mathrm{failure}}}{D_{\mathrm{environment}}} D_{\mathrm{failure}} / D_{\mathrm{environment}} \tag{3.21},$$

   where D_{failure} represents the ineffective radiation dose of the component or the material itself, and $D_{\mathrm{environment}}$ represents the dose at the actual service position of the component or material. The Model System Engineering needs to clarify the lower limit of RDM for this model. The range of RDM is usually between 1 and 10. The following principles can be used to determine the lower limit of RDM for different model missions. For a spacecraft with experimental properties (such as scientific experiment satellite, technology experiment satellite), the RDM should not be less than 1; for a spacecraft with application properties (such as telecommunication spacecraft, navigation spacecraft), the RDM should not be less than 2; for a special-purpose spacecraft (such as military spacecraft), the RDM should not be less than 2.5.

2. Requirements for the selection of electronic components and materials

When selecting electronic components and materials for a spacecraft, the total ionizing dose resistance should be considered. In the selection and procurement of electronic components and materials, the required total ionizing dose resistance of electronic components and materials should be clarified.

For the electronic components and materials whose total ionizing dose resistance cannot meet the requirements of radiation dose at the actual service position in the spacecraft, appropriate shielding measures need to be taken to ensure that the requirements for the total ionizing dose resistance will be met during the spacecraft's lifecycle.

3. Requirements of total ionizing dose test

At the planning and prototyping phases of spacecraft development, the overall condition of electronic components and materials should be selected according to model to identify a list of electric components and material that need a total ionizing dose irradiation test, and then the test plan should be developed. If the test object is a component or material whose total ionizing dose resistance cannot be determined, its total ionizing dose resistance will be measured by irradiation test as a design basis.

4. Requirements for protective measures

For the components or materials which need protective measures after analysis, the potential protective measures and their additional effects should be controlled. Specifically:

a. The possible protective measures include radiation shielding, cold backup and alternation, optimization of the internal configuration of the devices, overall layout, and tolerance design, etc.

b. Analysis and tests should be conducted to ensure that the potential protective measures will not affect the mechanical properties, electrical properties, thermal properties, and antistatic performances of the components or materials.

c. For the components or materials that have undergone protection design, analysis and simulation should be used for verification. The engineering verification analysis method of total ionizing dose effect is to calculate the final RDM of all electronic components and materials in the final design state, and to complete the analysis verification through determining whether the RDM is greater than the lower limit required by System Engineering.

Requirements for SEE protection design

1. Requirements for the SEE resistance of components

   a. When selecting logic devices (such as CPU, Digital Signal Processing (DSP), Static Random-Access Memory (SRAM), Field Programmable Gate Array (FPGA),) and CMOS devices, the SEE resistance of the components should be considered.

   b. For the power Metal-Oxide-Semiconductor Field-Effect Transistors (MOSFETs) (such as VDMOS transistors) used at high voltages (usually 50 V and above), their performance against SEB and SEGR should be considered. Generally, the selected power MOSFET should have an anti-SEB/SEGR VDS (VDS is the voltage between the drain and source of the field effect transistor) threshold greater than its operating voltage and meet a certain margin (or derating).

2. Requirements for SEE irradiation test

   At the planning and prototyping phases of spacecraft development, the overall condition of the electronic components should be selected in conformity with the model to develop the test plan. When necessary, the test shall be organized and implemented by the System Engineering. The test should be organized and carried out by the personnel with professional knowledge. The plan, program, and rules of the test should be formulated by experts to ensure the correctness of test method.

3. Requirements for protective measures

   a. As for the SEE protection design for stand-alone products, the SEEs to be considered include SEU, SEL, SEB, SEGR, and single event instantaneous interference.

   b. The SEU protection measures for memory devices include: configuring the memories reasonably, using the Error Detection And Correction (EDAC) function and Two-out-of-three voting system. The SEU protection measures for control devices include: adopting multi-level redundancy and fault-tolerant system and watch dog timer, and designing the software reasonably. The SEU protection measures for programming languages include: adopting a majority voting system for data, setting the program paths, use multidigit as a flag, etc. The SEU protection measures for FPGA include: adding a regular refreshment function for on-chip triple modular redundancy, adopting the SRAM FPGA devices, and designing the power-off restart function.

   c. The SEL protection measures include: selecting the SEL-insensitive devices (such as CMOS/SOS or CMOS/SOI); power supply current limiting; powering each stand-alone machine in a multi-machine system; remote power-off protection, and overcurrent power-off protection, etc.

d. For the power MOSFETs that operate under a power supply greater than 50 V or under the working voltage, the threshold voltage of their anti-SEB/SEGR VDS or VGS (VGS is the voltage of the gate relative to the source) should be higher than the VDS or VGS in actual application, and a 50% design margin should be considered.

e. The stand-alone machine must have an automatic or ground-controlled power-off restart function to avoid the constant SEL.

Requirements for the protection design of displacement damage effect

1. Solar cell protection measures

   The corresponding anti-radiation measures can be taken to minimize the solar array power loss and material degradation caused by radiation.

   a. The solar cells with strong resistance to displacement damage should be chosen. The displacement damage effect of solar cells depends on the material and technique. In terms of radiation resistance, the N+P silicon solar cells are better than P+N silicon solar cells, while gallium arsenide solar cells are better than silicon solar cells.

   b. Thinning the single solar cell can also improve the radiation resistance of the cell itself.

   c. The solar cell cover with appropriate thickness is chosen. As the solar cell cover serves as the shield of the solar cell, increasing its surface density can reduce the displacement damage effect.

   d. A power margin is designed for solar batteries. To ensure sufficient output power at the end of solar array lifecycle, the electrical performance attenuation caused by the radiation of charged particles must be accurately predicted, and a certain margin should be considered to analyze the end-of-life power.

2. Protective measures for CCD

   Although CCD devices are very sensitive to displacement damage, the use of reasonable protection design measures can ensure the normal operation of CCD devices and meet the needs of space missions. The current protective design measures mainly include:

   a. Choose the CCD devices with high radiation resistance.

   b. Shield the devices sensitive to displacement effect.

   c. Adopt the cooling measures, select appropriate device architectures and operating status, and process the signals. These measures can all alleviate the displacement damage effect.

3. Protective measures for optoelectronic couplers

   The optoelectronic coupler is usually composed of a LED and a phototransistor. The optical medium lies in the middle. The displacement effect is the main factor to be considered when choosing an optoelectronic coupler.

The displacement damage resistance measures commonly used for optoelectronic couplers include:

a. Increase the driving current of LED so that the LED is capable of emitting a higher optical radiation power.

b. Saturate the collected current of the optoelectronic coupler to obtain a more stable output current.

c. Choose an appropriate optoelectronic coupler to improve the margin of CTE.

Requirements for the protection design of surface charging/discharging effect[9]

1. Requirements for the analysis of surface charging/discharging effect

   a. At the planning phase of spacecraft development, the System Engineering should analyze the hot plasma environment that may be encountered by a spacecraft during its lifecycle and determine the overall condition of surface charging/discharging effect.

   b. For the HEO spacecraft, a detailed analysis of the spacecraft's charging condition should be conducted during the prototyping phase. For the LEO spacecraft passing through a polar region, the above analysis can also be carried out according to the development needs.

2. Test requirements for surface charging/discharging effect

   a. During the development of a HEO spacecraft, the design of the surface charging protection of high-voltage solar arrays should be tested and verified. For the high-voltage solar arrays that have been tested and verified, the design changes of their materials and processes should be verified by evaluating the charging and discharging design of the new solar arrays (for instance, whether the secondary discharge will occur) through the surface charging tests in vacuum. For the LEO spacecrafts passing through polar regions, the above-mentioned tests can also be carried out according to the needs of the development.

   b. During the development of a HEO spacecraft, when the surface charging of its new material cannot be quantitatively evaluated, the surface charging/discharging condition can also be evaluated through the surface charging tests in vacuum. For the LEO spacecrafts passing through polar regions, the above-mentioned tests can also be carried out according to the needs of development.

   c. During the development of a HEO spacecraft, the ability of the spacecraft or its components to withstand the surface-discharging pulse interference should be verified through the component-level or spacecraft-level surface-discharging simulation test. The verification requires to complete the component-level surface-discharging simulation test on the test parts and the spacecraft-level surface-discharging simulation test on the spacecraft electrical model.

3. Design measures for surface charging and discharging effects

During the development of a HEO spacecraft, the following measures can be taken to shield against surface charging and discharging effects:

a. Select the thermal control materials with low resistivity and high secondary electron emission coefficient.

b. Design a good grounding system to provide a discharge channel for the accumulated charges generated by the interaction between the spacecraft and the space environment.

c. Design appropriate shields, including the Faraday cage design of main spacecraft structure, the external cable shield, the ESDS component shield, etc.

d. Protect high-voltage solar arrays against surface-discharging effect through reasonable configuration, filling room-temperature-vulcanizing silicone rubber (Room Temperature Vulcanizing (RTV) silicone rubber) between the solar cells and other measures.

e. Ground testing and operation process control

Requirements for the protection design of internal charging effect

1. Requirements for internal charging effect analysis

a. At the planning phase of spacecraft development, the System Engineering should analyze the high-energy electron radiation environment possibly encountered during the spacecraft's lifecycle and determine the overall situation of internal charging effect.

b. At the prototyping phase of a stand-alone device, a detailed analysis of internal charging should be carried out based on the overall analysis of internal charging effect.

2. Requirements for internal charging effect test

During the process of spacecraft development, the internal charging risk of the new materials/components whose internal charging cannot be quantitatively assessed can be evaluated through tests.

3. Design measures for internal charging effect

During the process of spacecraft development, the following measures can be taken to protect against internal charging effect:

a. Shield design. It is an effective protective measure to reduce the density of the injected current through shielding.

b. Grounding design. The grounding design can provide a discharge channel for the injected current and can also protect against internal charging effect.

Material selection. The selection of a material with sufficient discharging capability can effectively control the accumulation of the injected current in the material, thereby reducing the occurrence rate of internal charging.

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CHAPTER 4

Basis of Spacecraft Orbit Design

Huang Meili, Zhou Jing, Zhang Lei, and Tian Baiyi
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Orbit design is an important part of spacecraft engineering and is based on orbital mechanics, which mainly deals with the particle dynamics of a satellite under gravitational field and other external forces. It is an essential theoretical basis for the design, measurement, control and practical application of a spacecraft.

The upper atmosphere extends up to 160 km above the ground and then thins out into space. The orbits of Earth spacecrafts are classified by typical parameters, as shown in Table 4.1.

TABLE 4.1 Classification of Earth Orbits

Orbit Type	Apogee/km[a]	Perigee/km[b]	Eccentricity[c]	Inclination/(°)[d]	Period[e]
Geostationary	35,786	35,786	0	0	1 sidereal day
Geosynchronous	35,786	35,786	Close to 0	0–90	1 sidereal day
Elliptical	39,400	1000	High	62.9	1/2 sidereal days
Near Earth	Multiple	Multiple	0~high	0–90	>90 minutes

[a] The furthest distance from the Earth's surface; [b]the nearest distance from the Earth's surface; [c]the ratio of the difference between perigee-geocenter distance and apogee-geocenter distance to their sum; [d]the angle between the orbital plane and the equatorial plane; [e]a sidereal day is 23 h56 min4.09 s. Different orbit types are represented by the following abbreviations:

GSO 35,786 km from the Earth
MEO 2000–30,000 km from the Earth
LEO 200–2000 km from the Earth
HEO An orbit with its eccentricity greater than 0.25 and less than 1, such as the Molniya orbit

The main characteristics of each orbit are described below.

Low Earth Orbit (LEO): LEO is a kind of nearly circular orbit with low altitude. A typical LEO satellite has an altitude of 500–1500 km and an orbital period of 1.5–2 hours. During each orbital period, the satellite can be observed by a specific ground station for only a few minutes. All the space shuttles of International Space Station and NASA, as well as most of remote sensing satellites, are working in LEOs. A number of recently planned and/or deployed communication satellite constellations are also located in the LEOs with an altitude of 500–2000 km and an inclination of 30°–90° (on both poles). Because the LEOs are closer to the Earth than other orbits, some smaller and simpler satellites can be deployed in LEOs.

Medium Earth Orbit (MEO): A typical MEO satellite has an altitude of 2000–30,000 km and an orbital period of several hours. The US GPS operates in a MEO orbit with an orbital period of 1/2 sidereal days. Some of China's navigation satellites and some remote sensing satellites also work in the MEO orbits.

Geosynchronous orbit (GSO): GSO is a prograde orbit with an orbital period equal to the Earth's rotation period. The geostationary Earth orbit (GEO), located in the equatorial plane of the Earth, is a GSO with zero orbital inclination and eccentricity and a semi-major axis of 42,164 km. Most of communications satellites operate in GEO orbit.

Highly elliptical orbit (HEO): Among various HEOs, the Molniya orbit is a special orbit named after a Soviet communications satellite, with a perigee of 1000 km and an apogee of 39,400 km. The nice thing about Molniya is that it can travel over the northern hemisphere for a long time with an argument of perigee of 270°. Some U.S. military satellites operate in the Molniya orbit with an inclination of 63.4° so that they can spy on Russia for 10 hours out of a 12-hour orbital period.

The orbits for lunar and deep space exploration spacecrafts are classified as follows:

Transfer orbit: The orbit that a spacecraft passes through when moving from one orbit to another, also called transfer orbit. The orbit through which the spacecraft flies from the Earth to the moon is called Earth-moon transfer orbit. The orbit through which the spacecraft flies from the Earth to Mars is called Earth-Mars transfer orbit. The orbit through which the spacecraft flies from the moon to the Earth is called moon-Earth transfer orbit.

Lunar free return orbit: An Earth-Moon transfer orbit through which a probe arriving on the moon and not braking can return to the Earth without orbital maneuver or with just a minimal thrust for orbital correction.

Regressive orbit: An orbit in which a spacecraft periodically and continuously travels to and from two or more celestial bodies and passes by them without stopping. A spacecraft operating on such an orbit requires no or only a few orbital maneuvers.

Halo orbit: An orbit in which a spacecraft moves in a closed curve around the libration point (Lagrange point) of a restricted three-body problem. The halo orbit around the libration point of the sun-Earth system is called solar halo orbit. The halo orbit around the libration point of the Earth-moon system is called lunar halo orbit.

This chapter summarizes the basic theories and engineering applications of spacecraft orbit design, including various basic knowledge and important theories on spacecraft orbit mechanics, as well as the orbiting theory, calculation method, orbit control, propellant budget and other engineering application methods of various spacecrafts.

4.1 GEOMETRIC ANALYSIS OF TASK SPACE

4.1.1 Basics of Spherical Trigonometry

When calculating the position of a spacecraft in the celestial coordinate system, the following basic relation in spherical trigonometry is often used.

Spherical triangle is a triangle composed of large arcs on a celestial sphere, with sides and angles measured by angle values (as shown in Figure 4.1). Similar to planar triangles, spherical triangles also need to satisfy specific side-angle relations, for example, the sum of the three sides of a triangle is less than 360°, and the sum of the three angles is greater than 180° but less than 540°.

If the three sides of the spherical triangle ABC are denoted as a, b and c, and the corresponding three internal angles are A, B and C, then the formulas commonly used for calculating such a spherical triangle will be

1. Sine formula

$$\frac{\sin a}{\sin A} = \frac{\sin b}{\sin B} = \frac{\sin c}{\sin C} \tag{4.1}$$

2. Cosine formula for sides

$$\begin{cases} \cos a = \cos b \cos c + \sin b \sin c \cos A \\ \cos b = \cos c \cos a + \sin c \sin a \cos B \\ \cos c = \cos a \cos b + \sin a \sin b \cos C \end{cases} \tag{4.2}$$

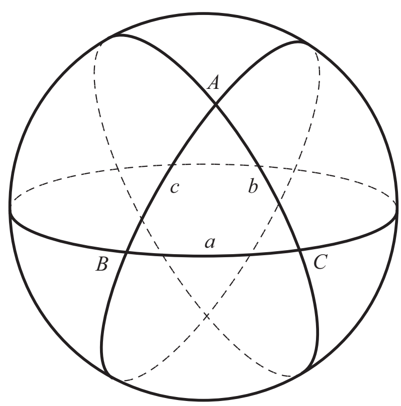

FIGURE 4.1 Spherical triangle.

3. Cosine formula for angles

$$\begin{cases} \cos A = -\cos B \cos C + \sin B \sin C \cos a \\ \cos B = -\cos C \cos A + \sin C \sin A \cos b \\ \cos C = -\cos A \cos B + \sin A \sin B \cos c \end{cases} \tag{4.3}$$

4. The first five-element formula

$$\begin{cases} \sin a \cos B = \cos b \sin c - \sin b \cos c \cos A \\ \sin a \cos C = \cos c \sin b - \sin c \cos b \cos A \\ \sin b \cos A = \cos a \sin c - \sin a \cos c \cos B \\ \sin b \cos C = \cos c \sin a - \sin c \cos a \cos B \\ \sin c \cos A = \cos a \sin b - \sin a \cos b \cos C \\ \sin c \cos B = \cos b \sin a - \sin b \cos a \cos C \end{cases} \tag{4.4}$$

5. The second five-element formula

$$\begin{cases} \sin A \cos b = \cos B \sin C + \sin B \cos C \cos a \\ \sin A \cos c = \cos C \sin B + \sin C \cos B \cos a \\ \sin B \cos a = \cos A \sin C + \sin A \cos C \cos b \\ \sin B \cos c = \cos C \sin A + \sin C \cos A \cos b \\ \sin C \cos a = \cos A \sin B + \sin A \cos B \cos c \\ \sin C \cos b = \cos B \sin A + \sin B \cos A \cos c \end{cases} \tag{4.5}$$

6. Four-element formula

$$\begin{cases} \cos a \cos C = \sin a \cot b - \sin C \cot B \\ \cos a \cos B = \sin a \cot c - \sin B \cot C \\ \cos b \cos A = \sin b \cot c - \sin A \cot C \\ \cos b \cos C = \sin b \cot a - \sin C \cot A \\ \cos c \cos B = \sin c \cot a - \sin B \cot A \\ \cos c \cos A = \sin c \cot b - \sin A \cot B \end{cases} \tag{4.6}$$

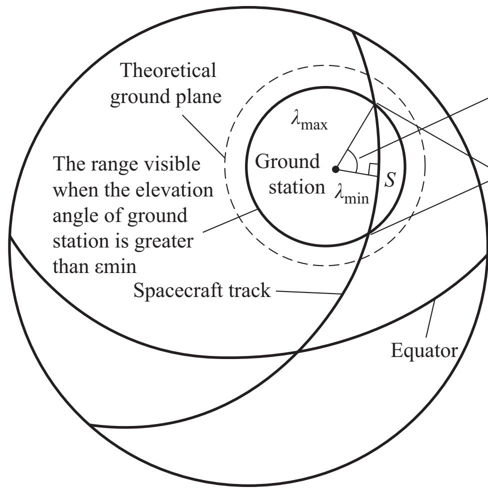

FIGURE 4.2 Diagram of tracking arc of ground station.

4.1.2 Calculation of Ground-Station Tracking Arc

In order to simplify the formulas, we assume that (1) the orbit is circular, that is, the spacecraft moves at uniform speed; (2) the orbital altitude is low, so that the time for a spacecraft to pass the orbit top is short and the arc of the Earth's rotation movement during this period can be ignored.[1]

The geometric relationship between the ground trajectory of a spacecraft and the tracking range of the ground station is shown in Figure 4.2. The minimum elevation angle at which the tracking antenna of a ground station can communicate with the ground plane is required to be 5°. In Figure 4.2, the dotted circle with ground station as its center represents the range of the minimum elevation angle $\varepsilon_{\min} = 0°$, while the solid circle shows the range of the minimum elevation angle $\varepsilon_{\min} = 5°$.

The solid and dotted circles shown in Figure 4.2 are the projections onto the large sphere of the spacecraft's orbit. The two circles and the ground station form two concentric cones. After the value of $\varepsilon_{\min}$ is given, the maximum geocentric angle $\lambda_{\max}$, the maximum FOV (field of view) angle $\eta_{\max}$ of the spacecraft and the maximum distance $D_{\max}$ from the spacecraft to the ground station can be calculated, namely:

$$\sin\eta_{\max} = \sin\rho\cos\varepsilon_{\min} \tag{4.7}$$

$$\lambda_{\max} = 90^\circ - \varepsilon_{\min} - \eta_{\max} \tag{4.8}$$

$$D_{\max} = R_E \sin\lambda_{\max} / \sin\eta_{\max} \tag{4.9}$$

where $\rho = \arcsin\left(\frac{R_E}{R_E + h}\right)$, representing the maximum FOV angle (R_E is the radius of the Earth) covering the Earth.

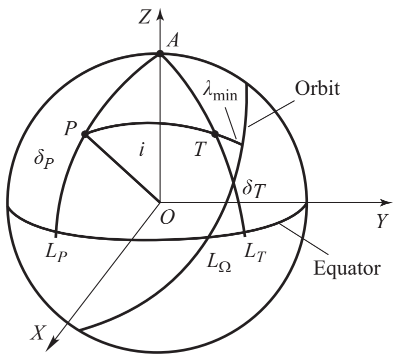

FIGURE 4.3 Diagram of minimum geocentric angle.

Now let's calculate λ_{min} (see Figure 4.3). Suppose the orbit is a circular orbit with the ascending-node longitude of L_Ω, the inclination of i, and the pole of P (the pole refers to the intersection point between the normal line of the orbit plane and the celestial sphere). Then

$$\begin{aligned} \delta_P &= 90^\circ - i \\ L_P &= L_\Omega - 90^\circ \end{aligned} \tag{4.10}$$

By using the four-element formula, we obtain

$$\sin(L_T - L_\Omega) = \tan\delta_T / \tan i \tag{4.11}$$

We use the calculation equation of the geocentric angle from the ascending node to the ground station, as given below, to determine when the spacecraft on the circular orbit is right above the ground station after crossing the equator:

$$\sin u = \sin\delta_T / \sin i \tag{4.12}$$

In the general case (that is, when the spacecraft is not right above the ground station), the minimum geocentric angle λ_{min} between the ground station and the spacecraft's ground trajectory is used to determine the parameters when the spacecraft is closest to the ground station (not necessarily above it). In this case, λ_{min} is equal to 90° minus the geocentric angle between the ground station and the instantaneous orbital pole. If the longitudes and latitudes of the orbital pole and ground station are known, then the value of λ_{min} in the spherical triangle APT (see Figure 4.3) can be given by the following equation according to the cosine law of sides (while paying attention to the supplementary angle relationship):

$$\sin\lambda_{min} = \sin\delta_P \sin\delta_T + \cos\delta_P \cos\delta_T \cos(L_T - L_P) \tag{4.13}$$

When the spacecraft trajectory in any round is closest to the ground station, the maximum tracking angular rate at which the ground station can observe the spacecraft will be:

$$\theta_{\max} = \frac{V_s}{D_{\min}} = \frac{2\pi(R_E + H)}{PD_{\min}} \tag{4.14}$$

where V_s is the orbital velocity of the spacecraft, and P is the orbital period.

Obviously, the angular rate varies with the distance from the spacecraft to the ground station. The closer the spacecraft is to the ground station, the greater the angular rate will be.

By using the relational expression of a spherical right triangle, it is not difficult to calculate the total azimuth angle $\Delta\phi$ and total tracking arc length (radian) S at which the spacecraft is tracked in and out of the ground station:

$$\cos\frac{\Delta\phi}{2} = \frac{\tan\lambda_{\min}}{\tan\lambda_{\max}} \tag{4.15}$$

$$\cos\frac{S}{2} = \frac{\cos\lambda_{\max}}{\cos\lambda_{\min}} \tag{4.16}$$

Similarly, by using the relational expression of a spherical right triangle, the total tracking time T can be calculated:

$$T = \left(\frac{P}{180^\circ}\right)\arccos\left(\frac{\cos\lambda_{\max}}{\cos\lambda_{\min}}\right) \tag{4.17}$$

where arccos is calculated in angular degrees, and the orbital period is calculated in minutes.

4.1.3 Calculation of Lighting Conditions

4.1.3.1 Angle between Sunlight and Orbital Plane

The angle ß between sunlight and orbital plane is shown in Figure 4.4. This angle is a parameter for calculating the time of spacecraft exposure to in-orbit light (sunshine area) and darkness (shadow area) and the sunlight exposure angle of solar arrays. The angle γ in Figure 4.4 is the angle between sunlight and the normal of orbital plane. Because β is supplementary to γ, namely $\gamma = 90^\circ - \beta$, γ can be determined. γ can be calculated according to the above cosine relation of a spherical triangle, as shown in Figure 4.5.

The spherical coordinates of orbital normal are: declination $= 90^\circ - i$, right ascension $= \Omega - 90^\circ$.

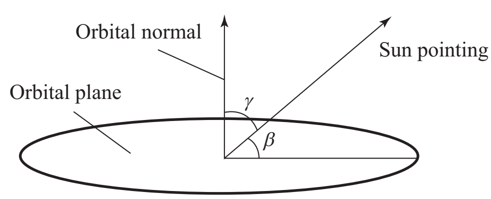

FIGURE 4.4 Angle between sunlight and orbital plane (1).

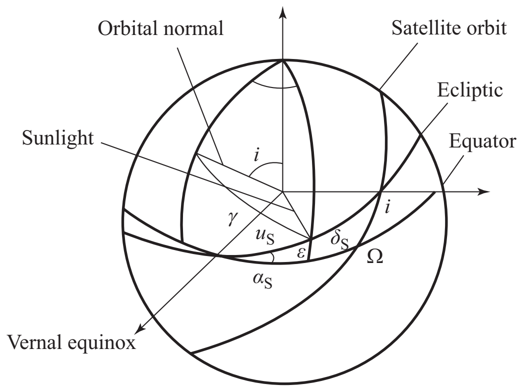

FIGURE 4.5 Angle between sunlight and orbital plane (2).

According to the cosine relation of a spherical triangle, the relationship among the angle β, the sun's right ascension α_s and declination δ_s as well as the position of spacecraft orbit plane in space can be obtained, as shown below:

$$\beta = \arcsin\left[\cos\delta_S \sin i \sin(\Omega - \alpha_S) + \sin\delta_S \cos i\right] \tag{4.18}$$

If the apparent motion orbit of the sun relative to the Earth is approximately circular, the sun's declination can be calculated by using spherical triangle relation:

$$\sin\delta_S = \sin\varepsilon \sin u_S \tag{4.19}$$

where $\varepsilon \approx 23.5°$ is the angle between the ecliptic and declination, u_s is the angular distance between the ecliptic and ascending node, and u_s can be calculated according to the number of days N from the vernal equinox:

$$u_s = \left(360^\circ/365.2422 = 0.9856^\circ/\text{day}\right) \times N\left(^\circ\right) \tag{4.20}$$

Then the following equation can be used to calculate the sun's right ascension α_s (Figure 4.6):

$$\begin{aligned} \tan\alpha_S &= \cos\varepsilon \tan u_S \\ \sin\alpha_S &= \tan\delta_S \cot\varepsilon \end{aligned} \tag{4.21}$$

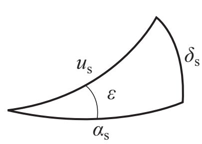

FIGURE 4.6 Solar argument U_s.

4.1.3.2 Sun Elevation Angle

Solar altitude is the elevation angle of the sun relative to the ground plane. It is an important parameter in visible remote sensing. The solar altitude varies at different latitudes at the same time (the same geographical longitude) (because the Earth is spherical). If the solar altitude is denoted as θ_h, then

$$\theta_h = \arcsin\left[\sin\varphi \sin\delta_S + \cos\varphi \cos\delta_S \cos(\Omega - \alpha_S + \Delta\alpha)\right] \tag{4.22}$$

where φ is the local geographical latitude; and $\Delta\alpha$ is the difference between the right ascension and ascending node corresponding to the latitude crossed by the spacecraft and is expressed as

$$\Delta\alpha = \arcsin\left(\frac{\tan\varphi}{\tan i}\right) \tag{4.23}$$

4.1.3.3 Time of Earth Eclipse Times

The time of Earth shadow in general case (when the sunlight is not parallel to the orbital plane) is shown in Figure 4.7, where the sun rays are perpendicular to the paper outward. In Figure 4.7, the ellipse represents an orbit, and AB and AC are both large arcs. The geocentric angle $u_{\max}$ corresponding to the arc AB is the geocentric angle occupied by the in-orbit Earth shadow when the sun is parallel to the orbital plane, that is, half of the maximum Earth shadow (see Figure 4.8). The geocentric angle corresponding to the arc AC is β, namely the angle between sunlight and the orbital plane. In this case, the arc BC (denoted as u) is half the length of the Earth shadow arc. When the angle β between sunlight and the orbital plane achieves $u_{\max}$ (where the spacecraft orbit is tangent to the Earth's shadow, as seen from the dotted line in Figure 4.7), the spacecraft will have no shadow, that is, it will be exposed to full sunshine.

Let's calculate the arc length of Earth shadow, denoted as u. In Figure 4.7, ABC is a spherical right triangle, then

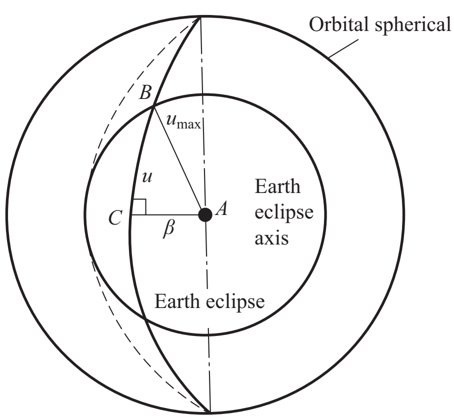

FIGURE 4.7 Relation diagram of Earth-shadow arc length u in general case.

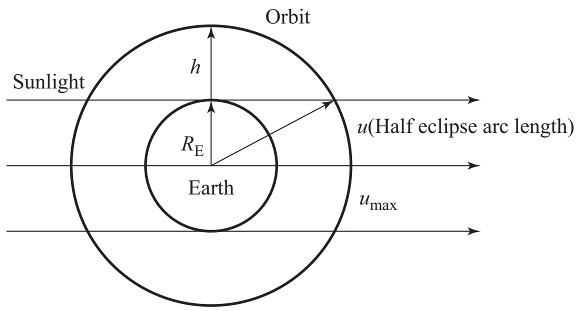

FIGURE 4.8 Diagram of largest Earth shadow.

$$u = \arccos\left(\frac{\cos u_{max}}{\cos\beta}\right) \tag{4.24}$$

$$\cos u_{max} = \frac{\sqrt{h^2 + 2R_E h}}{R_E + h} \tag{4.25}$$

Finally, when the angle between sunlight and the orbital plane is β, the ratio of the Earth shadow arc length to the orbital perimeter, namely f_E, will be

$$f_E = \frac{u}{\pi} = \frac{1}{\pi}\arccos\frac{\sqrt{h^2 + 2R_E h}}{(R_E + h)\cos\beta} \tag{4.26}$$

The product of this equation and the orbital period is just Earth shadow time.

4.1.4 Analysis of Launch Window

The launch window of a spacecraft refers to the date, time and time range available for spacecraft launch. The time for spacecraft launch (generally following the Beijing time in China) is often determined according to the mission requirements, the working conditions of the spacecraft and ground system, and the operation laws of the spacecraft and sun (or moon), while considering the requirements for launch site location and orbit insertion parameters.

4.1.4.1 Constraints on Launch Time

1. The sunlight conditions for a ground target
2. The requirement for sunlight direction to ensure the normal power supply of onboard solar batteries
3. The geometric relationship among the Earth, the spacecraft and the sun required by spacecraft attitude measurement and maneuver
4. The requirement for sunlight direction to support onboard thermal control

5. The required directions of direct sunlight, the sunlight reflected by the moon and the sunlight reflected by the Earth on some special spacecraft components
6. The requirement for the time length of spacecraft stay in the Earth shadow
7. The requirements for the orbit position of the spacecraft when entering and leaving the shadow
8. The requirements for onboard Tracking, Telemetry and Command (TT&C) conditions, as proposed by the ground TT&C station
9. The requirement for recovery time
10. The coplanar requirement to be met during the rendezvous and docking of a manned spacecraft in orbit

The above constraints shall be analyzed and calculated separately, often requiring the use of the knowledge in this chapter to calculate the angle between sunlight and a coordinate axis of the spacecraft as well as the Earth shadow time.

4.1.4.2 Several Elements of Spacecraft Launch from a Launch Site

Generally, the launch vehicle does not fly with lateral maneuvers in the launching process, that is, it does not change the launch trajectory plane. However, the launch of few GEO spacecrafts requires the launch vehicles to change the launch trajectory plane in order to reduce the inclination of the transfer orbit. In this way, the propellant carried by such a spacecraft can be saved or the spacecraft life can be improved. The inclination of the transfer orbit, if not reduced by the launch vehicle, will be not less than the geographic latitude of the launch site. This is not favorable for launching a GEO spacecraft from a launch site with high geographic latitude.

In this section, the change of orbital inclination by launch vehicle will not be considered.

The factors that determine the orientation of the orbital plane of the launched spacecraft in space include the geographical latitude φ and longitude λ of the geographical location S of the launch site, the azimuth angle A of the launch vehicle and the launch time t.

4.1.4.2.1 Determination of Launch Azimuth The orientation of the orbital plane of a spacecraft in space is determined by two parameters: the orbital inclination (i.e., the angle between the orbital plane and the Earth's equatorial plane), and the right ascension Ω of the orbital ascending node (i.e., the right ascension of the orbital ascending node on the inertial equatorial coordinate system, measured from the vernal equinox), as shown in Figure 4.9.

The launch azimuth A is measured eastward starting from due north of the launch site (not greater than 180°). It determines not only the trajectory plane of the rocket, but also the azimuth of the orbital plane of the spacecraft.

The spacecrafts are generally launched in two ways: the ascending launch from the azimuth A, as shown in the position S_1 in Figure 4.9; and the descending launch from the

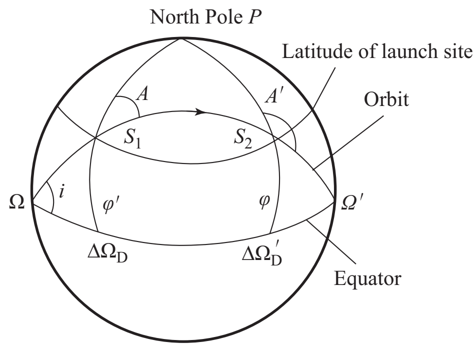

FIGURE 4.9 Diagram of launch azimuth A.

azimuth A', as shown in the position S_2. The two launch methods can obtain the same orbital plane orientation (i.e., the same orbital inclination i).

The relationship between the launch azimuth A and the launch azimuth A' is as follows:

$$A' = 180^\circ - A \tag{4.27}$$

The relationship between the orbital inclination i and the launch azimuth A in the spherical right triangle $S_1\Omega\Delta\Omega_D$ is as follows:

$$\cos i = \sin A \cos\varphi \tag{4.28}$$

It can be seen from the above equation that the accuracy of the launch azimuth is directly related to the orbital inclination i. Therefore, the launch vehicle shall perform accurate aiming before launching. If $A = 90°$ (launch due east), the orbital inclination i is equal to the geographical latitude φ of the launch site. If $A \neq 90°$, the orbital inclination i is greater than the geographical latitude φ of the launch site. To obtain an orbital inclination equal to 90° (approximately 90° for a solar synchronous orbit), the launch azimuth must be $A = 0°$ or $A = 180°$, that is, the spacecraft shall be launched in the direction of due north (ascending launch) or due south (descending launch). Descending launch is generally adopted.

4.1.4.2.2 Determination of Launch Time The above calculation only determined the launch azimuth according to the requirement for the orbital inclination i but did not determine the right ascension Ω of the orbital ascending node or analyze the launch time. The right ascension Ω of the orbital ascending node and the launch time t will be analyzed below.

As can be seen from Figure 4.9, the increment $\Delta\Omega_D$ of the right ascension of the meridian of the launch site relative to the right ascension Ω of the orbital ascending node at the launch time can be determined by the following equation:

$$\Delta\Omega_D = \arcsin\left(\frac{\tan\varphi}{\tan i}\right) \tag{4.29}$$

To achieve the given right ascension Ω of the orbital ascending node, it is assumed that the Earth rotation is not considered. The sidereal hour angle α_L of the launch site (relative to the right ascension of the vernal equinox) at the launch time shall be determined by the following equation:

$$\alpha_L = \Omega + \Delta\Omega_D \tag{4.30}$$

In the analysis of the launch time, the conversion of α_L into time (15°/h) shall also be considered during calculation. In addition, the duration t_A from the launch time to the time of orbit insertion (i.e., the flight time of the launch vehicle) shall be considered, that is, the actual launch time shall be earlier by t_A. In the example given by Figure 4.10, t_A is contained in t_G.

Take ascending launch as an example. To achieve the right ascension Ω of the ascending node of the launch orbit, the universal time t_G at the launch moment on the launch site shall be calculated according to the following equation:

$$t_G = \frac{1}{15}\left[\Omega + \arcsin\left(\frac{\tan\varphi}{\tan i}\right) - (\alpha_G + \lambda)\right] - \frac{1}{60}t_A \tag{4.31}$$

The above equation has been converted into hours. What's in the square bracket is calculated in angles, and the time for orbit insertion t_A is expressed in minutes.

Attention shall be paid to the signs of α_G and t_G in the calculation. In the example given by Figure 4.10, α_G is positive and t_G is negative. Since the geographical longitude of the launch site is λ, the local time t_{s1} at the launch moment on the launch site is

$$t_{s1} = \frac{\lambda}{15} + t_G \tag{4.32}$$

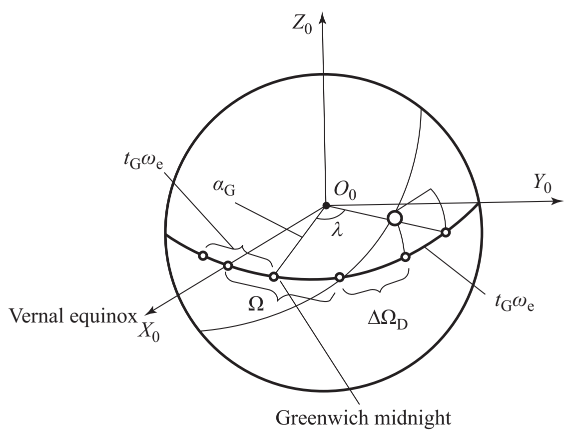

FIGURE 4.10 Relation diagram for launch time determination.

4.2 BASIS OF ORBITAL DYNAMICS

4.2.1 Two-Body Problem

The motion of two celestial bodies under Newtonian gravity is simplified into the motion of two particles under Newtonian gravity, called two-body problem. It is the simplest motion model to study the motion of celestial bodies. Because the model is integrable, its orbit can be used as an intermediate orbit to study the motion of celestial bodies under other perturbative forces. Therefore, the two-body problem model is the basis for studying the motion of celestial bodies and for the applications such as spacecraft orbit design. Next, we proceed from the motion equation of the two-body problem to discuss the basic relations in this problem.

Suppose the masses of two particles are m_1 and m_2, respectively, their coordinates in an inertial coordinate system are $\mathbf{r}_1$ and $\mathbf{r}_2$, respectively, and their relative position is $\mathbf{r} = \mathbf{r}_2 - \mathbf{r}_1$, $r = |\mathbf{r}|$. According to Newton's second law, the equation of absolute motion of the two-body problem is

$$\begin{aligned} m_1 \ddot{\mathbf{r}}_1 &= \frac{G m_1 m_2}{r^3} \mathbf{r} \\ m_2 \ddot{\mathbf{r}}_2 &= -\frac{G m_1 m_2}{r^3} \mathbf{r} \end{aligned} \tag{4.33}$$

The relative motion equation is

$$\ddot{\mathbf{r}} = -\frac{\mu}{r^3} \mathbf{r} \tag{4.34}$$

where $\mu = G(m_1 + m_2)$. From the above equation of motion, the six (center of mass) motion integrals of the two-body problem can be obtained

$$\begin{aligned} m_1 \dot{\mathbf{r}}_1 + m_2 \dot{\mathbf{r}}_2 &= \mathbf{C}_1 \\ m_1 \mathbf{r}_1 + m_2 \mathbf{r}_2 &= \mathbf{C}_1 t + \mathbf{C}_2 \end{aligned} \tag{4.35}$$

The three angular momentum integrals are

$$\mathbf{r} \times \dot{\mathbf{r}} = \mathbf{H} \tag{4.36}$$

and the one energy integral is

$$\frac{1}{2} \dot{\mathbf{r}} \cdot \dot{\mathbf{r}} - \frac{\mu}{r} = K \tag{4.37}$$

where $\mathbf{C}_1$, $\mathbf{C}_2$, H and K are the corresponding integral constant vectors or integral constants, respectively. According to the above integrals, the centers of mass of the two bodies are at rest or in uniform linear motion, the motions of the two bodies are always in the same plane, and the relative motion energy of the two-body problem is constant.

The orbital integral can be obtained from the relative motion equation of the two-body problem by proper transformation in polar coordinate system:

$$r = \frac{p}{1 + e\cos(\theta - \omega)} \tag{4.38}$$

where p is the semi-latus rectum, e is the eccentricity, θ-ω is the true anomaly, and ω is the periastron argument.

The integral shows that the two-body motion orbit is a conic curve.

$$p = h^2 / \mu \tag{4.39}$$

Depending on the eccentricity e, the orbits can be divided into circular orbits $(e = 0)$, elliptical orbits $(0 < e < 1)$, parabolic orbits $(e = 1)$ and hyperbolic orbits $(e > 1)$.

For an elliptical orbit, the semi-latus rectum is $p = a(1 - e^2)$, where a is the semi-major axis of the elliptical orbit. The angular momentum of the orbit motion is:

$$h = \sqrt{\mu p} = \sqrt{\mu a\left(1 - e^2\right)} \tag{4.40}$$

The relation between the mean angular velocity of elliptical motion and the semi-major axis, namely Kepler's third law, is:

$$\frac{4\pi^2}{T} a^3 = n^2 a^3 = \mu = G\left(m_1 + m_2\right) \tag{4.41}$$

The dynamic formula of elliptical motion is:

$$v^2 = \mu\left(\frac{2}{r} - \frac{1}{a}\right) \tag{4.42}$$

Because of $|r - a| \le ae$ in elliptical motion, the auxiliary angle E can be defined as:

$$r = a\left(1 - e\cos E\right) \tag{4.43}$$

Then the Kepler equation of elliptic motion can be obtained:

$$E - e\sin E = nt + M_0 = n\left(t - \tau\right) = M \tag{4.44}$$

where τ is the time for passing the periastron, M is called mean anomaly, and E is eccentric anomaly. Given the mean anomaly M, the eccentric anomaly E can be solved iteratively by the above equation. The relationship between the eccentric anomaly E and the true anomaly $f = \theta - \omega$ is shown in Figure 4.11:

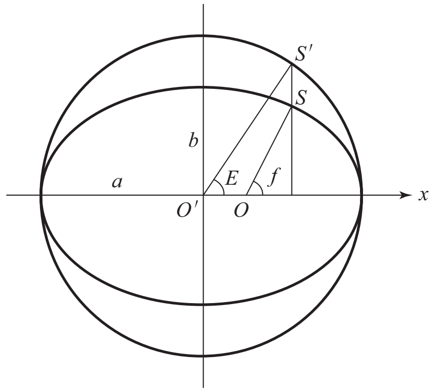

FIGURE 4.11 Relationship between the eccentric anomaly E and the true anomaly f in elliptic orbit.

It is known from Figure 4.11 that

$$\begin{aligned} r\cos f &= a(\cos E - e) \\ r\sin f &= b\sin E = a\sqrt{1-e^2}\sin E \end{aligned} \tag{4.45}$$

For a hyperbolic orbit, the semi-latus rectum is $p = a(e^2 - 1)$, and the corresponding dynamic formula is

$$v^2 = \mu\left(\frac{2}{r} + \frac{1}{a}\right) \tag{4.46}$$

By introducing the auxiliary quantity F:

$$r = a(e\cosh F - 1) \tag{4.47}$$

the Kepler equation of the hyperbolic orbit can be obtained:

$$e\sinh F - F = \nu(t - \tau) \tag{4.48}$$

where $\nu = \sqrt{\mu/a^3}$, and the auxiliary quantity F can also be solved iteratively by the above equation.

Given the semi-major axis a, eccentricity e and true anomaly f (or eccentric anomaly E or mean anomaly M) of an orbit, the orbit shape and the specific position of a spacecraft in the orbital plane can be determined. Considering the situation of three-dimensional space, two additional orbital elements in addition to the periastron argument ω need to be introduced to determine the spatial orientation of the orbital plane. They are respectively the orbital inclination i and the right ascension Ω of the ascending node, as specifically defined in Figure 4.12.

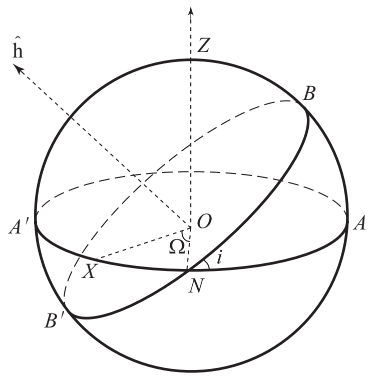

FIGURE 4.12 Orientation of orbital plane in space.

4.2.2 Orbit Perturbation

A spacecraft in orbit will be affected by a variety of perturbation factors, in addition to the center of mass gravity of the Earth. Although the order of magnitude of those perturbation terms is relatively small (about 10^{-3} for the maximum term), they will still have an important influence on the spacecraft orbiting. The following are the main perturbations that need to be considered in the process of spacecraft orbiting.[2]

4.2.2.1 Earth's Non-spherical Gravitational Perturbation

The Earth's non-spherical gravitational perturbation is the most important perturbation factor (up to 10^{-3}) affecting the spacecraft orbit. By using the Legendre expansion, the gravitational potential function of the Earth can be expressed in the following form:

$$\Delta V = \frac{\mu}{R}\sum_{l\geq 2}\sum_{m=0}^{l}\left(\frac{a_e}{R}\right)^l \overline{P}_{lm}\left(\sin\varphi\right)\left[\overline{C}_{lm}\cos m\lambda_G + \overline{S}_{lm}\sin m\lambda_G\right] \tag{4.49}$$

where a_e is the Earth's equatorial radius, R is the geocentric distance of the spacecraft in the Earth-fixed coordinate system, φ is the geocentric latitude of the spacecraft, $\overline{P}_{lm}$ is the normalized combine-Legendre polynomial, and $\overline{C}_{lm}$ and $\overline{S}_{lm}$ are a group of spherical harmonic coefficients related to the Earth's gravitational field (such as WGS84 model). According to the expression of the Earth's non-spherical gravitational potential, the corresponding perturbation acceleration can be obtained as

$$\mathbf{F}_{NSP} = \left(\frac{\partial \Delta V}{\partial \mathbf{r}}\right)^T = \left(\frac{\partial \mathbf{R}}{\partial \mathbf{r}}\right)^T\left(\frac{\partial \Delta V}{\partial \mathbf{R}}\right)^T \tag{4.50}$$

where $\left(\partial\mathbf{R}/\partial\mathbf{r}\right)^T$ represents the coordinate transformation from the Earth-fixed coordinate system to the geocentric celestial coordinate system, including precession, nutation, Earth rotation and polar motion; and $\left(\partial\Delta V/\partial\mathbf{R}\right)^T$ represents the gradient of the non-spherical gravitational potential with respect to the position vector $\boldsymbol{R}$ of the probe in the Earth-fixed coordinate system.

4.2.2.2 Gravitational Perturbation of the Third Body

For a spacecraft moving around the Earth, solar and lunar gravity (the sun and moon serve as particles) is a typical third-body gravitational perturbation with the following perturbation acceleration:

$$\mathbf{F}_{TBP} = \sum_{i=1}^{2} -\mu_i' \left(\frac{\Delta_i}{|\Delta_i|^3} + \frac{\mathbf{r}_i'}{|\mathbf{r}_i'|^3} \right) \tag{4.51}$$

where $\mu_i'(i=1,2)$ are the gravitational constants of the sun and moon; $\boldsymbol{\Delta}_i$ are the position vectors of the spacecraft relative to the sun and moon; and $\mathbf{r}_i'$ are the position vectors of the sun and moon relative to the Earth. The position vectors of various celestial bodies can be calculated by analysis or numerical ephemeris. For a spacecraft in low orbit, the order of magnitude of solar and lunar gravitational perturbation is a small second-order quantity (in which the order of magnitude of lunar perturbation is 10^{-7} and that of solar perturbation is 10^{-8}). For a spacecraft in high orbit, the order of magnitude of solar and lunar gravitational perturbation is slightly larger (in which the order of magnitude of lunar perturbation is 10^{-5} and that of solar perturbation is 10^{-6}).

4.2.2.3 Air Drag Perturbation

A spacecraft (especially low-orbit spacecraft) flying in the upper atmosphere of the Earth will be affected by atmospheric resistance. The resistance acceleration can be written as follows:

$$\boldsymbol{F}_{ADP} = -\frac{1}{2}\left(\frac{C_D S}{m}\right)\rho v^2 \left(\frac{\boldsymbol{v}}{v}\right) \tag{4.52}$$

where $\boldsymbol{v}$ is the flight speed of the spacecraft relative to the atmosphere, and v is its value; ρ is the atmospheric density; s/m is the effective area-mass ratio of the spacecraft to the resistance (referred to as area-mass ratio); and C_D is the resistance coefficient. For a typical spacecraft with the effective area-mass ratio of 10^9 (normalized unit) and the flight altitude above 300 km, the order of magnitude of atmospheric drag perturbation will not be higher than 10^{-6}. In other words, for the spacecrafts flying in low and medium orbits, the magnitude of atmospheric drag perturbation can be treated as a small second-order quantity.

4.2.2.4 Solar Radiation Pressure Perturbation

The solar radiation pressure (or light pressure for short) directly acting on the surface of a spacecraft is not high, but can affect the movement of the spacecraft. By using a simplified cylindrical Earth-shadow model, the perturbation acceleration of solar radiation pressure on the spacecraft can be written as:

$$\mathbf{F}_{SRP} = \nu C_R \left(\frac{S}{m}\right)\rho_\Theta \left(\frac{1AU}{\Delta}\right)^2 \frac{\boldsymbol{\Delta}}{\Delta} \tag{4.53}$$

where C_R is the surface reflection coefficient of the spacecraft, with the value of 1–2; (S/m) is the effective area-mass ratio of the spacecraft; $\rho_\Theta = 4.56\times10^{-6}\,N/m^2$ represents the solar radiation pressure constant near the Earth; $\boldsymbol{\Delta}$ is the position vector of the spacecraft relative to the sun; and ν is the Earth shadow factor, which is determined by the following relationship according to the position of the spacecraft:

$$\cos\psi_1 = \left(\frac{\mathbf{r}}{r}\right)\cdot\left(\frac{\mathbf{r}_s}{r_s}\right) < 0 \ \ \& \ \ \sin\psi_1 = \sqrt{1-\cos^2\psi_1} < \frac{a_e}{r} \Rightarrow \nu = 0 \tag{4.54}$$

where a_e represents the equatorial radius of the Earth, and $\mathbf{r}_s$ is the position vector of the sun relative to the Earth. When the above relationship is satisfied, the spacecraft falls into the Earth's shadow. In this case, the Earth shadow factor is $\nu = 0$, which means that the spacecraft is not affected by solar radiation pressure; otherwise, $\nu = 1$, suggesting that the spacecraft is completely exposed to solar radiation pressure. For a typical spacecraft with effective area-mass ratio, the order of magnitude of solar radiation pressure perturbation is 10^{-7} (in high orbit) or 10^{-8} (in medium and low orbits).

4.2.3 Orbital Maneuver

The transition of a spacecraft from its initial orbit (or parking orbit) to its target orbit is an orbital transfer, usually accomplished by orbital maneuvers. Orbital transfer can be achieved in many ways, including one maneuver, two maneuvers and multiple maneuvers depending on the number of orbital maneuvers. Only when the initial orbit intersects the target orbit, the orbital transfer can be achieved with a single maneuver. The initial and target orbits can be circular, elliptical, or even hyperbolic. They may be coplanar or non-coplanar and may intersect or not intersect with each other. The transfer orbit can be an ellipse, or a hyperbolic curve to save the transfer time. In orbital transition, the most energy-efficient way is often pursued. As far as the orbit is concerned, the most energy-efficient way is to make the orbital maneuver at the perigee and change the semi-major axis a of the orbit. As far as the change of orbital plane is concerned, the most energy-efficient way is to change the orbital inclination i at the argument of latitude of about $u = 0^\circ$ or 180° (i.e., ascending node or descending node) by relying only on the velocity increment in the normal direction of the orbital plane, or to change the right ascension Ω of the ascending node at $u = 90^\circ$ or 270°. In a specific space mission, the factors such as energy consumption, flight time, guidance precision and TT&C convenience shall also be comprehensively considered before choosing the optimum maneuver mode that can be realized. Next, Hohmann transfer will be taken as an example to show how to realize the orbital maneuver.[3]

In the orbital transfer between two coplanar concentric circular orbits, Hohmann transfer is the most energy-efficient double pulse maneuver method. As shown in Figure 4.13, Hohmann transfer orbit is an elliptical orbit that is tangent to two circular orbits along an apse line. Suppose the semi-major axes of the two circular orbits are r_1 and r_2, respectively. If the transition from the lower circular orbit (as the initial orbit) to the higher circular orbit (as the target orbit) is considered, then the semi-major axis of the elliptical Hohmann transfer orbit will be

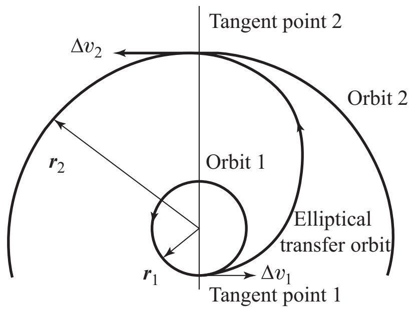

FIGURE 4.13 Hohmann transfer orbit.

$$a = \frac{1}{2}(r_1 + r_2) \tag{4.55}$$

The corresponding semi-major axis change is

$$\Delta a_1 = a - r_1 = \frac{1}{2}(r_2 - r_1) \tag{4.56}$$

Thus, the acceleration pulse Δv_1 at the tangent point 1 is

$$\Delta v_1 = \sqrt{\mu\left(\frac{2}{r_1} - \frac{1}{a}\right)} - \sqrt{\frac{\mu}{r_1}} = \sqrt{\frac{\mu}{r_1}}\left[\left(\frac{2r_2}{r_1 + r_2}\right)^{1/2} - 1\right] \tag{4.57}$$

Similarly, the acceleration pulse of the second orbital transfer at the tangent point 2 can also be calculated from the semi-major axes of the orbit before and after the orbital transfer:

$$\Delta v_2 = \sqrt{\frac{\mu}{r_2}} - \sqrt{\mu\left(\frac{2}{r_2} - \frac{1}{a}\right)} = \sqrt{\frac{\mu}{r_2}}\left[1 - \left(\frac{2r_1}{r_1 + r_2}\right)^{1/2}\right] \tag{4.58}$$

By the same principle, the transition from a higher orbit to a lower orbit is the reverse of the above process and requires two decelerations. Although Hohmann transfer orbit is energy-saving, its flight time and route are long, and it is only applicable to the transfer between coplanar circular orbits. If both the initial orbit and the target orbit are elliptical, the change of the apse line direction may also be involved. If the two orbits are not coplanar, the change of the orbital plane needs to be considered. Therefore, the best choice of spacecraft orbit maneuver should be considered comprehensively according to the concrete situation in an actual mission.

4.2.4 Multi-body Problem

In celestial mechanics, the motion of several celestial bodies under mutual universal gravitation is called multi-body problem. So far, except for two-body problem, these problems cannot be solved analytically. Three-body problem is the simplest multi-body problem and has been studied by many famous mathematicians such as Euler, Lagrange, Laplace, Jacobi and Poincaré. Other than the ten classical integrals (six center-of-mass motion integrals, three angular momentum integrals and one energy integral), the eleventh integral has not been found yet. To simplify the problem, Euler assumed that one of the three bodies had so small mass that it had no effect on the motion of the other two larger bodies, but was subjected to their gravitational pull. Since the two-body problem composed of two large bodies can be solved analytically, the three-body problem can be simplified into the motion of a smaller body under the gravitational pull of two larger bodies whose motions are known. As a result, the restricted three-body problem is obtained.

The restricted three-body problem is only the ultimate approximation of the general three-body problem. Since any small body always has a mass, this model is not an actual mechanical system in celestial mechanics. However, when the mass of a small body is small enough, this difference can be ignored. Although the restricted three-body problem still cannot be analytically solved, some of its special solutions (libration point and periodic orbit) and its Jacobi integral are very important for the study of the motion characteristics of small celestial bodies. In addition, the development of human spaceflight activities has injected new impetus into the qualitative and quantitative research of the restricted three-body problem, especially in the research field related to libration point.

Next, we turn to circular restricted three-body problem (CRTBP), one of the most commonly used basic degradation models for three-body problem. Take Earth-moon system as an example. The model assumes that the Earth and the moon orbit each other in a circular orbit according to the law of two-body motion. Compared with the Earth and the moon, the spacecraft has a negligible mass and moves under the gravitation of both the Earth and the moon.

According to Newton's law of universal gravitation, the motion equation of the spacecraft in the center-of-mass inertial coordinate system of Earth-moon system (denoted as C-XYZ) is

$$\ddot{\boldsymbol{R}} = -\frac{Gm_1\boldsymbol{R}_1}{R_1^3} - \frac{Gm_2\boldsymbol{R}_2}{R_2^3} \tag{4.59}$$

where G is the universal gravitation constant, m_1 and m_2 are the masses of the Earth and the moon, respectively, $\boldsymbol{R}_1$ and $\boldsymbol{R}_2$ are the position vectors of the spacecraft relative to the Earth and the moon, respectively, and $\boldsymbol{R}$ is the position vector of the spacecraft.

For CRTBP, the spacecraft motion is usually studied in a coordinate system rotating with Earth-moon system, which is called center-of-mass rendezvous coordinate system (denoted as C-xyz). The geometric relationship between C-xyz and C-XYZ is shown in Figure 4.14. It can be seen that, the XY (xy) plane is lunar orbital plane, and that the Z(z) axis is perpendicular to the paper outward and forms a right-handed coordinate system with the X and Y (x, y) axes. If the position vectors of the spacecraft relative to the origin, the Earth and the moon in the C-xyz system are, respectively, $\boldsymbol{r}$, $\boldsymbol{r}_1$ and $\boldsymbol{r}_2$, then

$$\begin{cases} \boldsymbol{R} = \boldsymbol{C}\cdot\boldsymbol{r}, \boldsymbol{R}_1 = \boldsymbol{C}\cdot\boldsymbol{r}_1, \boldsymbol{R}_2 = \boldsymbol{C}\cdot\boldsymbol{r}_2 \\ \boldsymbol{r} = \boldsymbol{C}^{-1}\cdot\boldsymbol{R}, \boldsymbol{r}_1 = \boldsymbol{C}^{-1}\cdot\boldsymbol{R}_1, \boldsymbol{r}_2 = \boldsymbol{C}^{-1}\cdot\boldsymbol{R}_2 \end{cases} \tag{4.60}$$

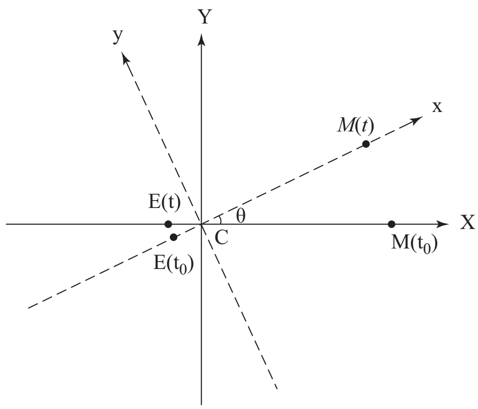

FIGURE 4.14 Center-of-mass inertial coordinate system C-XYZ and center-of-mass rendezvous coordinate system C-xyz.

Here C is a rotation matrix expressed by

$$\boldsymbol{C} = \boldsymbol{R}_Z(-\theta) = \begin{pmatrix} \cos\theta & -\sin\theta & 0 \\ \sin\theta & \cos\theta & 0 \\ 0 & 0 & 1 \end{pmatrix}, \boldsymbol{C}^{-1} = \boldsymbol{R}_Z(\theta) = \begin{pmatrix} \cos\theta & \sin\theta & 0 \\ -\sin\theta & \cos\theta & 0 \\ 0 & 0 & 1 \end{pmatrix} \quad (4.61)$$

Since the orbits of the Earth and moon rotating around each other are assumed to be circular, the following equation can be derived in the dimensionless unit system:

$$\dot{\boldsymbol{C}} = \begin{pmatrix} -\sin\theta & -\cos\theta & 0 \\ \cos\theta & -\sin\theta & 0 \\ 0 & 0 & 0 \end{pmatrix}, \ddot{\boldsymbol{C}} = \begin{pmatrix} -\cos\theta & \sin\theta & 0 \\ -\sin\theta & -\cos\theta & 0 \\ 0 & 0 & 0 \end{pmatrix} \quad (4.62)$$

By substituting the Equations (4.61) and (4.62) and $\ddot{\boldsymbol{R}} = \boldsymbol{C}\cdot\ddot{\boldsymbol{r}} + 2\dot{\boldsymbol{C}}\cdot\dot{\boldsymbol{r}} + \ddot{\boldsymbol{C}}\cdot\boldsymbol{r}$ into Equation (4.60), the motion equation of the spacecraft in the C-xyz can be obtained:

$$\begin{cases} \ddot{x} - 2\dot{y} = \partial\Omega/\partial x \\ \ddot{y} + 2\dot{x} = \partial\Omega/\partial y \\ \ddot{z} = \partial\Omega/\partial z \end{cases} \quad (4.63)$$

$$\text{where } \Omega = \frac{1}{2}\left[\left(x^2+y^2\right) + \mu(1-\mu)\right] + \frac{1-\mu}{r_1} + \frac{\mu}{r_2} \quad (4.64)$$

where $\mu = m_2/(m_1+m_2)$. The above equation is the general form of the CRTBP motion equation. According to the motion equation, a dynamic integral of CRTBP, namely Jacobi integral, can be obtained:

$$2\Omega^2 - v^2 = 2\Omega - \left(\dot{x}^2 + \dot{y}^2 + \dot{z}^2\right) = C_J \tag{4.65}$$

where the integral constant C_J is called Jacobi constant. Its value reflects the magnitude of spacecraft orbit energy and has the following relationship with the mechanical energy E: $C_J = -2E$. Therefore, the greater the Jacobi constant value, the smaller the orbital energy of the spacecraft.

Although the CRTBP corresponds to a time-invariant differential system and has a Jacobi integral, its motion equation still cannot be solved analytically in a strict sense. However, a group of very valuable particular solutions, namely libration point solutions, can be obtained. Libration point is a dynamic equilibrium point with unchanged position in the rendezvous coordinate system. According to the definition, the following basic equations satisfied by the libration points can be obtained if $\dot{x} = \dot{y} = \dot{z} = \ddot{x} = \ddot{y} = \ddot{z} = 0$ in the motion Equation (4.65):

$$\begin{cases} x - \dfrac{(1-\mu)(x+\mu)}{r_1^3} - \dfrac{\mu(x-1+\mu)}{r_2^3} = 0 \\ y\left[1 - \dfrac{1-\mu}{r_1^3} - \dfrac{\mu}{r_2^3}\right] = 0 \\ -z\left[\dfrac{1-\mu}{r_1^3} + \dfrac{\mu}{r_2^3}\right] = 0 \end{cases} \tag{4.66}$$

Since the term in the square bracket of the above third equation is not equal to zero, the equation will hold only if $z = 0$, namely all the libration points are located in the xy plane. It can be seen from the second equation in the above equation set that there are two scenarios of libration point solution. One scenario is $y \neq 0$, where the equations satisfied by the libration points are:

$$\begin{cases} x - \dfrac{(1-\mu)(x+\mu)}{r_1^3} - \dfrac{\mu(x-1+\mu)}{r_2^3} = 0 \\ 1 - \dfrac{1-\mu}{r_1^3} - \dfrac{\mu}{r_2^3} = 0 \end{cases} \qquad 4.67)$$

This equation set has two solutions, namely $L_4 : \left(0.5-\mu, \sqrt{3}/2, 0\right)$ and $L_5 : \left(0.5-\mu, -\sqrt{3}/2, 0\right)$. Because the points L_4 and L_5, together with two main bodies, form two equilateral triangles, they are called triangular libration points.

The other scenario is $y = 0$. In this scenario, the y and z coordinates of the libration points have been determined, and the x coordinate satisfies the following equation:

$$x - \frac{(1-\mu)(x+\mu)}{|x+\mu|^3} - \frac{\mu(x-1+\mu)}{|x-1+\mu|^3} = 0 \tag{4.68}$$

According to the different position relations between the libration points and the two main bodies, the above equation has three solutions, which are all located on the line between the two main bodies and therefore are called collinear libration points. Among them, the libration point between the two main bodies is denoted as L_1, that on the right side of the second main body $(x > 1-\mu)$ is denoted as L_2, and that on the left side of the first main body $(x < -\mu)$ is denoted as L_3. If the distances between the collinear libration points and their nearest main bodies are $\gamma_i\left(i=1,2,3\right)$, then

$$\begin{cases} \gamma_1^5-(3-\mu)\gamma_1^4+(3-2\mu)\gamma_1^3-\mu\gamma_1^2+2\mu\gamma_1-\mu=0 \\ \gamma_2^5+(3-\mu)\gamma_2^4+(3-2\mu)\gamma_2^3-\mu\gamma_2^2-2\mu\gamma_2-\mu=0 \\ \gamma_3^5+(2+\mu)\gamma_3^4+(1+2\mu)\gamma_3^3-(1-\mu)\gamma_3^2-2(1-\mu)\gamma_3-(1-\mu)=0 \end{cases} \tag{4.69}$$

The solutions of the above equation set can be expressed as the following series:

$$\begin{cases} \gamma_1=\left(\dfrac{\mu}{3}\right)^{1/3}\left[1-\dfrac{1}{3}\left(\dfrac{\mu}{3}\right)^{1/3}-\dfrac{1}{9}\left(\dfrac{\mu}{3}\right)^{2/3}-\cdots\right] \\ \gamma_2=\left(\dfrac{\mu}{3}\right)^{1/3}\left[1+\dfrac{1}{3}\left(\dfrac{\mu}{3}\right)^{1/3}-\dfrac{1}{9}\left(\dfrac{\mu}{3}\right)^{2/3}+\cdots\right] \\ \gamma_3=1-\nu\left[1+\dfrac{23}{84}\nu^2+\dfrac{23}{84}\nu^3+\dfrac{761}{2352}\nu^4+\cdots\right],\ \nu=\dfrac{7}{12}\mu \end{cases} \tag{4.70}$$

Usually, the calculation results of the first several terms of Equation (4.70) are taken as initial values, and then substituted into Equation (4.69). Through Newton iteration, the accurate calculation results of the positions of collinear libration points can be obtained. Figure 4.15 shows the positions of five libration points in the center-of-mass rendezvous coordinate system of Earth-moon system.

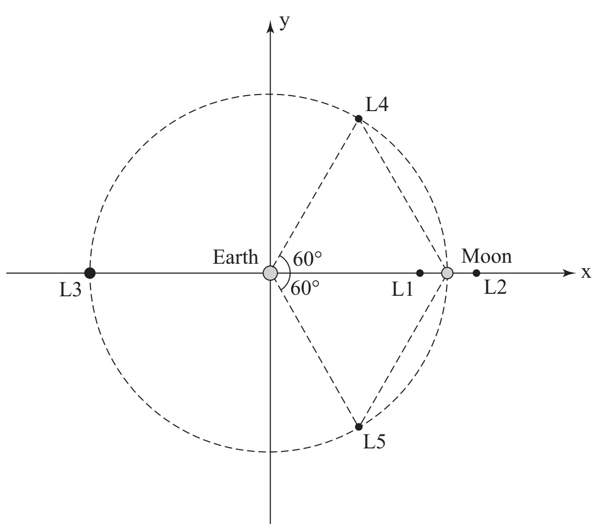

FIGURE 4.15 Five libration points in the center-of-mass rendezvous coordinate system of Earth-moon system.

As for the stability of the libration points, the existing research results have proved that the three collinear libration points are exponentially unstable (under the linearized model), especially when the higher-order terms are considered. The stability of triangular libration points is more complicated and is related to the system parameter μ. In the Earth-moon system, the two triangular libration points are linearly stable. The nonlinear stability of triangular libration points can be proved by KAM theory and will not be detailed here.

4.3 SPACECRAFT ORBIT DESIGN

4.3.1 General Orbit Design for a Single Spacecraft

4.3.1.1 Sun-Synchronous Orbit[4] (SSO)

SSO refers to the orbit in which the angular velocity of the orbital plane precession is equal to the average angular velocity of the sun moving along the ecliptic. In the SSO, the angle between the spacecraft-sun line and the spacecraft orbit plane is constant, so is the angle for viewing the sun. This orbit is used by the vast majority of optical remote sensing spacecrafts.

The orbital inclination i of the SSO satisfies

$$\cos i = -n_S \left(\frac{3J_2}{2p_2} n \right)^{-1} \tag{4.71}$$

where n_S is the average angular velocity of the sun, n is the average angular velocity of the spacecraft orbit, J_2 is the Earth perturbation term, $p_2 = \left(\frac{R_e}{a} \right)^2$, R_e is the Earth's equatorial radius, and a is the semi-major axis of the spacecraft orbit.

SSO is a retrograde orbit with an inclination greater than 90°. For example, the 500–1000 km solar-synchronous circular orbits have an inclination of 97.40°-99.47°.

The SSO has two characteristics. First, the local time for flying over the same latitude in each pass of the ascending (or descending) segment of the orbit is the same. In particular, the local time for flying over the equator in each pass of the descending segment is the local time of the descending node. Second, the angle between the sun ray and the orbital plane does not change much. Owing to the two characteristics, the SSO has a relatively stable light condition, which can help control the spacecraft energy, heat and attitude and reduce the complexity of spacecraft system. The solar altitude at the sub-satellite point of the SSO also does not change much and is beneficial to optical imaging during the Earth observation, so it is widely used by remote sensing spacecrafts. In engineering applications, the SSOs at the descending-node local time of 10:30 and 13:30 are often adopted by optical remote sensing spacecrafts, and the SSOs at the descending-node local time of 6:00 (morning) and 18:00 (dusk) are often adopted by microwave remote sensing spacecrafts.

4.3.1.2 Regressive Orbit

Regressive orbit is the orbit where the spacecraft trajectory at the sub-satellite point will repeat itself whenever a certain number of passes are finished. Suppose the duration of one cycle of orbit precession relative to the Earth surface is T_Ω and the orbital period of the spacecraft (generally referred to as nodal period) is T_e. If there are the reduced positive integers D and N satisfying

$$\mathrm{N} \cdot T_\Omega = \mathrm{D} \cdot T_e \tag{4.72}$$

then the ground trajectory of the spacecraft will repeat itself after D days or N cycles of orbiting. Such an orbit is just regressive orbit.

The operation of some census-type remote sensing spacecrafts (such as cartographic satellite and resources satellite) requires the periodic imaging of the same target. The periodic repeating of ground trajectory at the sub-satellite point of a regressive orbit just meets the requirements of this flight mission. Therefore, in addition to SSO, a regressive orbit or a sun-synchronous regressive orbit combining the characteristics of the two orbit types is often favored by those spacecrafts.

4.3.1.3 Frozen Orbit[5]

Frozen orbit is an orbit whose perigee argument and eccentricity have zero change rates. In general, the frozen orbits include the apse-line geostationary orbits at any inclinations not limited to a particular orbital inclination (such as the critical inclination orbit described above). As far as the low-orbit spacecrafts are concerned, the corresponding frozen orbits have two possible inclinations, i.e., $\omega = 90°$ or $270°$.

When the semi-major axis and inclination of a frozen orbit are given, the corresponding orbital eccentricity can be uniquely determined. For a low-Earth frozen orbit with a general inclination, its eccentricity has an order of magnitude of less than 1‰.

For a spacecraft operating in a frozen orbit, the altitude of the spacecraft passing the same latitude at different time will be unchanged. This feature can lead to a consistent image scale obtained by remote sensing spacecrafts and facilitate the splicing and comparison of the images taken at different time, thus playing an important role in Earth observation. For example, China's CBERS spacecrafts are operating in this type of orbits.

SSO, regressive orbit and frozen orbit are three different orbital characteristics. The orbit of a remote sensing spacecraft can have any one or two or even three of the orbital characteristics.

4.3.1.4 GSO

GSO is a prograde orbit whose orbital period is equal to the rotation period of the Earth. The GEO is located in the equatorial plane of the Earth. It is a GSO with zero inclination and eccentricity and a semi-major axis of 42,164 km.

GEO is a very special and unique GSO and the most precious orbital resource. There are a large number of spacecrafts in this orbit. The angular velocity of a GEO spacecraft is synchronous with the rotation velocity and direction of the Earth (from west to east), so the spacecraft does not move with respect to the Earth, and the static objects on the Earth's surface are always the same in the astronaut's view. The orbital period of the spacecraft is the same as the rotation period of the Earth, that is, 23 h56 min4.09 s. The trajectory of GSO sub-satellite point is an "8" that spans the northern and southern hemispheres and intersects at the equator.

In a remote sensing mission, the GEO spacecraft can provide large ground coverage (about 40%), in which the spacecraft is visible at any point around the clock. The GSO can expand the latitude coverage and provide a good coverage of the Polar Regions.

4.3.1.5 Critical Inclination Orbit

Critical inclination orbit is the orbit with an inclination equal to 63°26′ or 116°34′. With zero change rates of perigee argument and eccentricity, the stability of this orbit depends on its eccentricity. The greater the eccentricity, the better the stability. Because the apse line of critical inclination orbit does not drift, the perigee or apogee is always above certain latitude.

The former Soviet Union's communication satellite Molniya just operated in a critical inclination orbit with large eccentricity to ensure that the apogee was always in the high-latitude region of the northern hemisphere. The trajectory of its sub-satellite point is shown in Figure 4.16. Because the satellite moves slowly at apogee, it has more communication time in the territory of the former Soviet Union and less power consumption for signal transmission.

4.3.2 Constellation Design

Traditional flight missions were accomplished by single spacecrafts. A group of spacecrafts working together to complete a specific mission can be called constellation. All the design rules on single-spacecraft orbit design in Section 4.3.2 can be applied to constellation design. Therefore, we need to consider whether each spacecraft in the constellation can be launched into its orbit and whether it is within the FOV of the ground station. In addition, we should consider the number of spacecrafts in the constellation, their relative positions and how these positions change over time during one round of orbiting or during the spacecraft lifetime. In essence, the constellation design is an optimization problem.

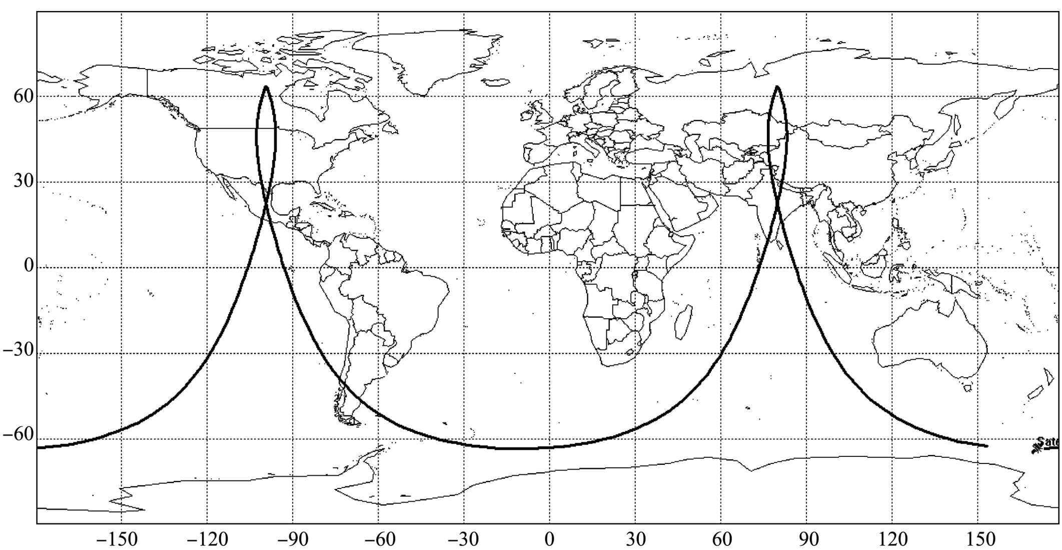

FIGURE 4.16 Sub-satellite point trajectory of Molniya satellite.

While meeting the requirement for ground coverage, the optimization design of a constellation can effectively reduce the total number of spacecrafts and their orbital heights, so as to bring down the total cost of the entire flight mission.

4.3.2.1 Constellation Characterization Parameters

It is very complicated to design a constellation by determining all the orbital elements of the spacecrafts. A reasonable method of constellation design is to assume that the orbits of all the spacecrafts in the constellation have the same semi-major axis, eccentricity and inclination, and that their right ascension of ascending node, argument of perigee and mean anomaly can be optimized according to the requirements of the flight mission.

Suppose the number of orbital planes in the constellation is N and the number of spacecrafts in each orbital plane is $Q_j\,(j=1,2,\cdots,N)$. The total number of spacecrafts in the constellation will be

$$T=\sum_{j=1}^{N}Q_j \tag{4.73}$$

The orbital elements of each spacecraft in the constellation are

$$\left.\begin{aligned} a[j,k]&=a\\ e[j,k]&=e\\ i[j,k]&=i\\ \Omega[j,k]&=\Omega_j\\ \omega[j,k]&=\omega_j\\ M[j,k]&=M_j+\Delta M_{jk} \end{aligned}\right\}(j=1,2,\cdots,N;k=1,2,\cdots,Q_j) \tag{4.74}$$

where a is the orbital semi-major axis, e is the eccentricity, i is the inclination, Ω_j is the right ascension of ascending node in the orbital plane j, ω_j is the argument of perigee in the orbital plane j, M_j is the mean anomaly of the first spacecraft in the orbital plane j and ΔM_{jk} is the mean anomaly difference between the spacecraft k and the first spacecraft in the orbital plane j.

4.3.2.2 Constellation Performance Evaluation Index

4.3.2.2.1 Performance Evaluation Index of an Ordinary Constellation Any constellation must first meet specific coverage requirements, so the coverage performance can be considered as a common criterion for optimizing the design of a constellation. The most commonly used performance evaluation indexes of a constellation are coverage performance indexes, which mainly include coverage percentage, maximum coverage gap, average coverage gap, time-averaged gap and average response time. These statistics can be used for single coverage or multi-coverage:

1. The coverage percentage of a point on the ground is the percentage of the accumulated time for its coverage in the total statistical time.
2. The maximum coverage gap at a point on the ground is equal to the longest coverage gap encountered at that point.
3. The average coverage gap at a point on the ground is the mean of all coverage gaps at that point.
4. The time-averaged gap at a point on the ground is the time-averaged gap duration.
5. The average response time at a point on the ground is the average waiting time from the time when the random coverage request is sent out from that point to the time when that point is covered.

4.3.2.2.2 Performance Evaluation Index of a Specific Constellation For the constellations with specific purposes, various factors should be considered to determine their specific performance evaluation indexes due to their different functions and different user needs. Next, we take navigation constellation as an example for discussion.

The objective function of the navigation constellation should consider the implementation of different performance, as measured by different criteria, in different areas. The navigation performance of the constellation is usually measured by the following parameters: vertical positioning accuracy, horizontal positioning accuracy, three-dimensional positioning accuracy, timing accuracy, total accuracy of position and time, number of visible spacecrafts, vertical precision factor, horizontal precision factor, position precision factor, time precision factor and total precision factor.

4.3.2.2.3 Main Design Methods of Constellations After the constellation characterization parameters and performance evaluation indexes are determined, the constellation design needs to be iterated repeatedly in order to strike a trade-off between performance and cost. During the decades of continuous development of constellation design technology, many constellation design methods, such as Walker method, coverage band method, spacecraft triangulation method, tetrahedron method, timeline grid method and genetic algorithm, have emerged one after another.

4.3.2.2.3.1 Walker Constellation Design Method The Walker constellation is defined as a constellation with the following characteristics: all the spacecrafts travel in circular orbits at the same altitude; each orbital plane contains the same number of spacecrafts; if there are more than one satellite in the same orbital plane, the spacecrafts will be evenly distributed in the orbital plane; the relative phase between the spacecrafts in adjacent orbits is a constant; the ascending nodes of all the orbits are evenly distributed along the equator; and all the orbital planes have the same angle to a reference plane. The Walker constellation can be described by three parameters, namely T, P and F, where T is the total number of spacecrafts, P is the number of orbital planes, and F is the metrics of the relative phase between the spacecrafts in adjacent orbits. The relative phase is $F \times 360^{\circ}/T$, where F = 0, 1..., P -1.

The constellation design using Walker method takes the following steps. Determine the position of the sub-satellite point in a constellation configuration at a given time. Determine the radius of the maximum circumferential circle for every three spacecrafts. Save the radius of the maximum circumferential circle. Repeat this process over time until the repetition of the geometrical relationship between spacecrafts and find the constellation configuration with the minimum radius of the maximum circumcircle.

Although the Walker constellations are important, they may not be the only suitable choice and may not provide the best performance for some missions, so it is necessary to investigate other constellation design methods.

4.3.2.2.3.2 Constellation Design Method Based on Numerical Calculation For global non-uniform constellations, regional constellations and intermittent coverage constellations, the lack of their symmetry increases the design difficulty. Due to their flexibility and diversity, there is generally no universal solution to their design. Most of the design methods in this field are based on numerical algorithms. In recent years, with the significant increase in the speed of numerical calculation, genetic algorithm has played an outstanding role in the constellation design field due to its strong global optimization ability.

Originating from evolution theory and natural selection, the genetic algorithm is to sample the research space for a given problem with a set of character strings (chromosomes), each representing one possible solution, and then process the chromosomes with the greatest potential to improve the result, and constantly optimize the solutions they represent. The genetic algorithm follows a cycle of four steps: producing the genomes; evaluating each set of chromosomes; selecting the most suitable chromosomes and genetically processing the chromosomes (through chromosomal exchange or mutation) to produce new genomes. This process is repeated until the solution is found.

4.3.2.3 Summary of Constellation Design

At present, it is impossible to use analytical method for constellation design. We need to constantly change the constellation parameters, obtain through numerical calculation the constellation performance evaluation indexes required by a flight mission, and use optimization methods (such as genetic algorithm) for repeated iteration in order to find the constellation with the lowest cost to meet the requirements of the flight mission. This section lists the main steps of constellation design adopted in recent years:

1. Determine the requirements of a flight mission, especially coverage requirements, performance indexes and altitude limits.
2. For a single spacecraft, conduct the comprehensive trade-off of all the orbit design by using the method in Section 4.3.2, and preliminarily determine the constellation orbit parameters.

3. Carry out numerical calculation based on the constellation parameters determined in Step 2 and obtain the constellation performance evaluation indexes required by the flight mission.

4. Use genetic algorithm and other optimization methods and repeat Step 3 until all the constellation parameters meeting the flight mission requirements are found.

5. Make a comparative analysis of the constellation design options obtained in Step 4, and finally choose a constellation design with lower cost that meets the flight mission requirements.

4.4 DESIGN OF DEEP-SPACE EXPLORATION ORBIT

4.4.1 Design Process of Deep-Space Exploration Orbit

The main work of deep-space exploration orbit design in the demonstration stage includes launch window analysis, velocity increment budget and transfer orbit design. In the model stage, a comprehensive analysis of orbit characteristics (such as shadow, TT&C and data transmission, orbit control) is required, in addition to refining the demonstration work.

The orbit design work mainly includes the selection of orbit type and orbital elements, orbit analysis, velocity increment budget, orbit control strategy, TT&C and data transmission arc analysis, light condition analysis and other work. While satisfying the design constraints, the orbit design should analyze the orbit design parameters to give an orbit concept that can meet the mission requirements. In addition, the orbit concept should be optimized to meet the engineering requirements.

The first step of orbit design is to define the mission objective. The orbit design varies greatly with the mission objective. At the initial stage of orbit design, the flight mode of the probe, including detection target, detection mode and propulsion mode, is defined. Then, an appropriate dynamic model is selected for calculation and simulation. The characteristics of different orbits are calculated and analyzed. Based on the demands of different scientific exploration missions, the orbits that meet the specific mission requirements are identified. The analysis contents include the influence of launch window; orbit characteristic analysis; velocity increment budget; geometric analysis such as TT&C and light analysis; orbit control strategy analysis, etc.

The orbit design should make continuous analysis and iteration with full consideration of the overall engineering requirements, the capability of ground TT&C system, the performance of launch vehicle, the configuration of ground application system and the requirements of relevant on-board sub-systems. This section mainly addresses several important engineering and technical problems in the overall design of lunar and planetary exploration missions, such as window design and transfer orbit design.

4.4.2 Design of Lunar Exploration Orbit

4.4.2.1 Lunar Exploration Launch Window

A launch window includes "launch opportunity window" and "launch time window". Launch opportunity window refers to the consecutive dates available for launch. Launch time window refers to the time interval available for launch on each launch date. The Moon's position on different dates is very different, and the launch trajectory on different dates is also different.

Theoretically, a probe can be launched into the Earth-moon transfer orbit at any time. However, due to the limitations of China's launch sites and launch vehicles, there are two launch opportunities, each lasting for 3–4 consecutive days, in a month. In addition, because the spacecraft arriving at the moon needs to be adapted for a period of time before a scientific exploration, a certain light condition should be met when the spacecraft enters the initial lunar orbit. As a result, the launch opportunities will be greatly reduced.

Due to the influence of the Earth rotation, only one moment each day is available for launch in every opportunity (3–4 consecutive days) in order to match the initial condition of the Earth-moon transfer orbit with the trajectory of the launch vehicle. Continuous launch moments can be obtained by using multiple trajectories and/or midcourse correction.

4.4.2.2 Earth-Moon Transfer

Direct Earth-moon transfer or phasing orbit strategy can be used to fly a spacecraft from the Earth to the moon.[6] A typical flight using a launch vehicle to directly launch a spacecraft into an Earth-moon transfer orbit only needs 3–5 days. The concept of phasing orbit has been widely used in space flight. Unlike the elliptical orbit, hyperbolic orbit, GSO or SSO with specific meaning, this concept is hard to be strictly defined. Internationally, an explanatory definition of this term is "the temporary orbit used before entering the final orbit". In some European Space Agency (ESA) literature, the phasing orbit used in lunar exploration flight is called intermediate orbit. However, in some American literature, the intermediate orbit used in the launch of a geostationary satellite is also called phasing orbit.

Whether a phasing orbit is used or not, a lunar probe will pass through the Earth-moon transfer orbit before arriving on the moon. It can be approximated as half a large elliptical orbit, much like the transfer orbit used for launching a geosynchronous satellite.

The near-node velocity required by Earth-Moon transfer is high, usually up to 10.3 km/s. To achieve this velocity, a multi-stage rocket must be used. The typical approach is to first launch the spacecraft into a circular parking orbit near the Earth, and then increase its velocity to the required magnitude through orbital maneuver. There are usually two ways to provide such a large velocity increment. One is the use of a solid rocket. In this case, the spacecraft can fly in the parking orbit for several rounds according to the need and then choose a right time for orbital maneuver. The other is to fire the final stage of a liquid rocket twice. After the rocket carries the spacecraft into an approximately circular parking orbit for a period of time, the final stage is fired again to provide the velocity increment needed for entering the Earth-moon transfer orbit. The CZ-3A rocket used by Chang'e 1 just falls into this type.

The main advantage of the first method is that the time for flying from the Earth to the moon is the shortest, which is obviously advantageous for manned flights. In the early Apollo missions, three days or so were required in each flight. However, this type of flight requires a very strict launch window.

4.4.3 Design of Planetary Exploration Orbit

4.4.3.1 Planetary Rendezvous Period and Launch Window

Because the orbits of different bodies are different from the Earth's orbit in terms of geometry and initial phase, the velocity increment required to depart from the Earth at different times is different. When two celestial bodies are near a fixed phase, the velocity increment required for the launch of a probe is small. The moment meeting this phase requirement is just launch opportunity. A set of launch opportunities is called launch window. The time interval between two launch windows is called rendezvous period.

The periods of the rendezvous between the planets in the solar system are shown in Table 4.2.

The time interval of the launch window can be estimated according to the planetary rendezvous period. The specific launch window should be searched for according to the planetary ephemeris.

The design of a planetary exploration orbit should start from the design of the transfer orbit, generally using the Lambert method, namely: fixing the departure time and arrival time, analyzing the departure and arrival speeds within the time frame of the mission, drawing the C3 energy contour (C3 energy is the square of the probe velocity relative to the planet at the boundary of the planet's gravitational field) and analyzing the departure and arrival time with the minimum energy consumption. The transfer during which a probe travels less than one lap of heliocentric orbit is called single-lap Lambert transfer. With the increase in the complexity of deep-space exploration missions, the multi-lap Lambert has also had good applications, especially in the sampling-returning exploration missions. Multi-lap Lambert is one of the good solutions to increasing the window interval between two launches or to optimizing the departure parameters and ensuring the carrying capacity.

The C3 energy contour has been widely used in various deep-space exploration missions. It can intuitively depict the change of the launch set in a given period and provide a visual basis for the design of the optimal launch window and a good initial value prediction for the optimization design of the optimal launch opportunity. The C3 energy contour is obtained by solving the Lambert problem with a given predetermined launch time interval and flight time (or arrival time interval). This method is an ergodic search algorithm.

TABLE 4.2 Periods of the Rendezvous between Solar System Planets

	Mercury	Venus	Earth	Mars	Jupiter	Saturn	Uranus
Mercury	–	–	–	–	–	–	–
Venus	0.3958	–	–	–	–	–	–
Earth	0.3173	1.5987	–	–	–	–	–
Mars	0.2762	0.9142	2.1354	–	–	–	–
Jupiter	0.2458	0.6488	1.0920	2.2350	–	–	–
Saturn	0.2428	0.6283	1.0351	2.0089	19.8618	–	–
Uranus	0.2415	0.6198	1.0121	1.9241	13.8324	45.5665	–
Neptune	0.2412	0.6175	1.0061	1.9026	12.7945	35.9576	170.5175

Take the Mars exploration in 2020 as an example. The contour plots of launch C3 energy, arrival C3 energy and total C3 energy of the Mars exploration mission are shown in Figure 4.17, Figure 4.18 and Figure 4.19, respectively. Each point in the plots represents a viable launch opportunity. Different points correspond to different launch times and flight times and require different amounts of C3 energy. Among them, the optimal short-transfer Mars exploration windows are concentrated in July 2020, and the corresponding interplanetary flight time is about 200 days.

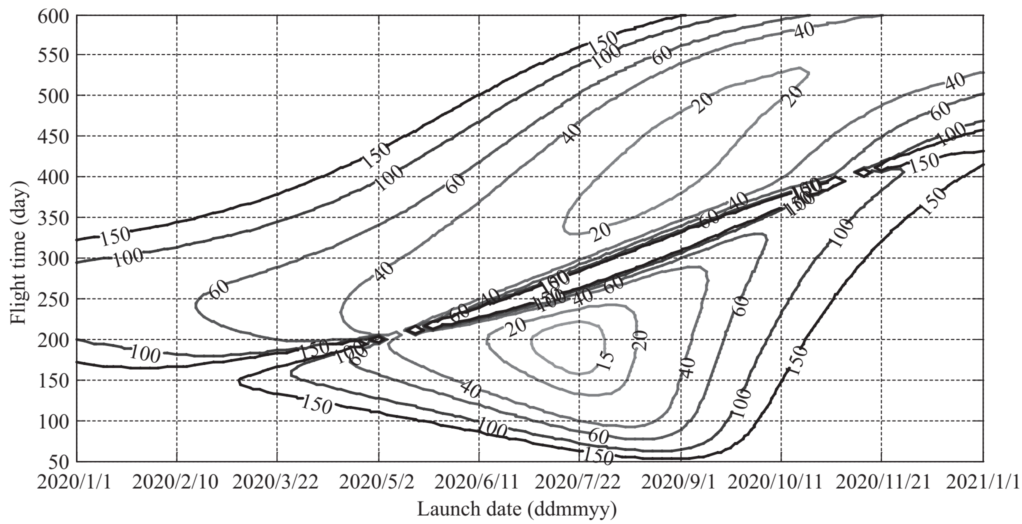

FIGURE 4.17 C3 energy contour of Earth launch in Mars mission.

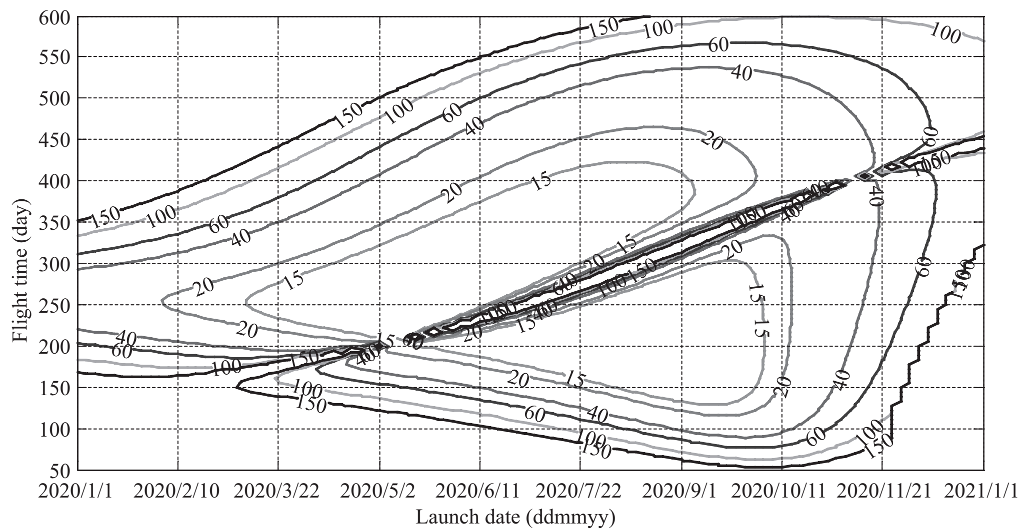

FIGURE 4.18 C3 energy contour of Mars arrival in Mars mission.

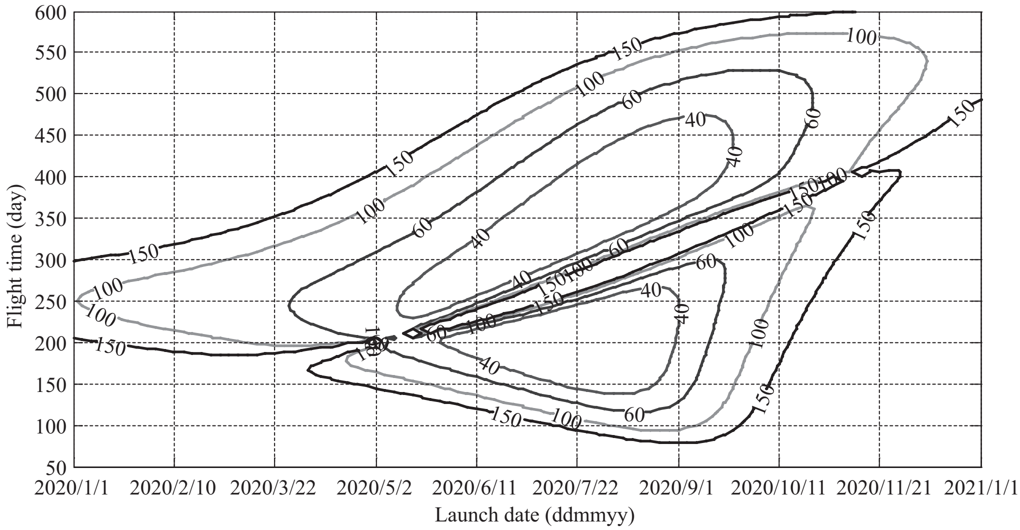

FIGURE 4.19 Total C3 energy contour of Mars mission.

4.4.3.2 Optimization Design of Interplanetary Transfer Orbit

After the launch window for planetary exploration is obtained, the interplanetary transfer orbit shall be designed and optimized according to the overall mission requirements. The orbit with the optimal launch mass, launch C3 energy, total velocity increment or remaining arrival mass is selected, while satisfying the constraints of the delivery system, launch site and probe sub-systems. Next, we take the optimization of Earth-Mars transfer orbit with deep-space maneuver as an example (Figure 4.20) to address the optimization design of an interplanetary transfer orbit.

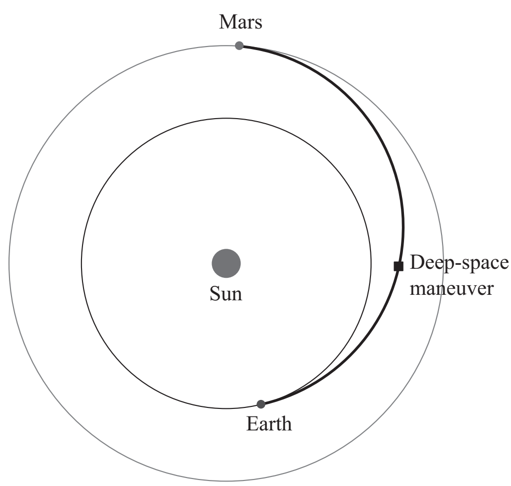

FIGURE 4.20 Diagram of Earth-Mars transfer orbit with deep-space maneuver.

The purpose of Earth-Mars transfer orbit design is to select the appropriate launch window and flight time, so that the probe can fly from spacecraft-rocket separation point to periareon, the parameters such as launch inclination, launch C3 energy and injection perigee argument can meet the mission constraints, and the remaining mass or total velocity increment of the probe can become optimal.

The design of Earth-Mars transfer orbit is transformed into the solving of multi-dimensional nonlinear programming problem, so the optimization parameters are selected as follows:

1. Probe launch time, t_L
2. Earth-Mars transfer time, *TOF*
3. Deep-space maneuver time, T_{DSM}
4. Deep-space maneuvering speed $\mathbf{V}_{DSM}$, 3×1 matrix
5. Hyperbolic overspeed of the probe arriving at Mar $\mathbf{V}_{\infty A}$, 3×1 matrix
6. Launch inclination of the launch vehicle, i_L

After selecting the above parameters, the orbit design problem is transformed into a multi-dimensional nonlinear programming problem by solving the "Lambert" problem several times. The nonlinear programming problem with constraints can be solved through sequential quadratic programming. Thus, the optimization design of Earth-Mars transfer orbit with deep-space maneuver strategy is completed.

4.5 PROPELLANT BUDGET

For either a GEO or LEO spacecraft, the actions such as orbital correction, orbital transfer (or maneuver) and orbit maintenance are needed. This requires the spacecraft to carry enough propellant to ensure the completion of orbital control work. Next, the overall design of a GEO spacecraft is taken as an example to analyze the technical issues related to propellant budget. The following analysis assumes a due-east launch from the Xichang launch site.

4.5.1 Analysis of Orbital Maneuvering Velocity

4.5.1.1 Determination of GEO Period

The time we spend on a daily basis is defined with the sun as a reference point, and thus is called mean solar time. As mentioned earlier, in the system of mean solar time, the time for an observation point on Earth to make a complete rotation relative to the sun (reference point) is 24 hours. Because the Earth also revolves around the sun by an angle of θ (about 360/365.2422 degrees/day), it takes the Earth less than 24 hours to rotate on its axis once relative to a star in the inertial space. If a star is used as the reference point, one round of GEO orbiting will have a smaller angle than one rotation around the sun (reference point) by θ. The GEO period (i.e., the time of one sidereal day) has been calculated to be $T = 23$ hours 56 minutes 04 seconds (in mean solar time).

4.5.1.2 Determination of GEO Semi-major Axis

The orbital period formula derived from Kepler's third law is as follows:

$$T = 2\pi\sqrt{\frac{a^3}{\mu}} \tag{4.75}$$

where T is the orbital period of the spacecraft; a is the semi-major axis of the orbit; $\mu =$ 3.986005 × 105 km^3/s^2 is the Earth's gravitational constant ($\mu = G \times M$, $G = 6.668462 \times$ 10–20 km^3/kg·s^2 is the universal gravitational constant, $M = 5.977414 \times 10^{24}$ kg is the Earth's mass).

Given the orbital period (i.e., $T = 23$ hours 56 minutes 04 seconds), the semi-major axis of the GSO can be calculated to be $a = 42164.6$(km) according to the above equation. Since the average radius of the Earth's equator is $RE = 6378$ km, the altitude of a geosynchronous satellite above the ground is $H = a - RE = 35{,}786.6$ km.

4.5.1.3 Orbital Maneuvering Method

The geographical latitude of the launch site has a great influence on the launch of a GEO spacecraft. That is, if a rocket is launched in the due east direction (the firing direction is 90°), the inclination of the transfer orbit (if unchanged) will be the geographic latitude of the launch site. For example, if the rocket is launched from the Xichang Launch Site in China, the inclination of the transfer orbit (*i*) will be 28.50° because the geographical latitude of Xichang is 28.50°.

Therefore, if the upper stage of the rocket is not used, the spacecraft needs to have sufficient maneuvering capability in order to enter the geostationary orbit. The orbital maneuvering method of a common spacecraft is shown in Figure 4.21. Before the apogee maneuver, the thrust axis of the engine shall be adjusted to the desired direction, which is consistent with the desired velocity increment direction. If a solid engine is used, this adjustment can be accomplished by applying only one velocity increment (ΔVi) at the apogee, as shown in Figure 4.21. If a liquid engine is used, the adjustment needs to be done by applying 2–3 speed increments at the apogee, as shown in Figure 4.21.

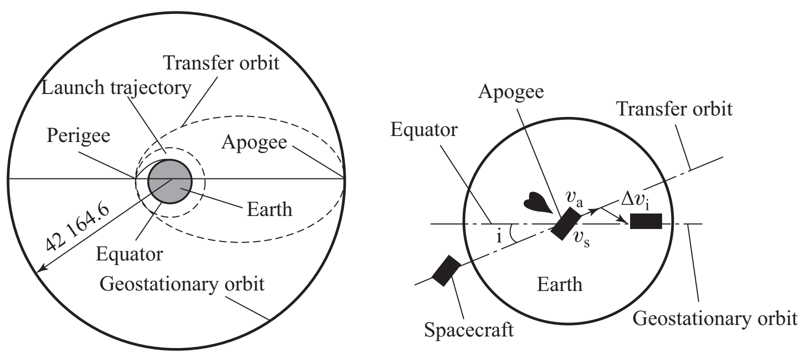

FIGURE 4.21 Diagram of orbital maneuvering method.

4.5.1.4 Delta V Magnitude Calculation

The velocity at any position in the transfer orbit is

$$v(r)=\sqrt{\mu\left(\frac{2}{r}-\frac{1}{a}\right)} \tag{4.76}$$

where r is the geocentric distance of the spacecraft at an instantaneous orbital position. First, calculate the semi-major axis of the transfer orbit according to the above equation. Second, calculate the apogee velocity v_a. The semi-major axis of the transfer orbit is

$$a_G=\frac{r_{\min}+r_{\max}}{2}=\frac{(200+6378)+(35786+6378)}{2}=24371(km) \tag{4.77}$$

Then the apogee velocity is

$$v_a=\sqrt{\mu\left(\frac{2}{r_{\max}}-\frac{1}{a_G}\right)}=1.595(km/s) \tag{4.78}$$

It can be seen that there is a variable angle i between the apogee velocity v_a and the required GEO velocity v_s (i.e., the depicted resultant velocity). See Figure 4.22 for details. The resultant velocity in Figure 4.22 is the velocity v_s required by the GSO. Since the orbit is circular, v_s can be determined by the following equation:

$$v_s=\sqrt{\frac{\mu}{a}}=\sqrt{\frac{\mu}{42164}}=3.074(km/s) \tag{4.79}$$

In Figure 4.22, v_a is the apogee velocity of the transfer orbit. The angle between v_a and v_s is the inclination of the transfer orbit (i). The velocity increment Δv_i to be generated by the engine at apogee can be derived from the illustrated geometric relationship:

$$\Delta v_i=\sqrt{v_a^2+v_s^2-2v_av_s\cos i}=1.835(km/s) \tag{4.80}$$

Based on the assumptions in this chapter, the velocity increment calculations required by orbital maneuver are summarized in Table 4.3. It is not difficult to see that the most fuel-efficient way is to change the orbital inclination at apogee, because the apogee has the lowest velocity.

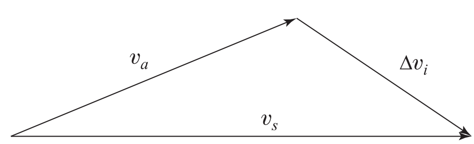

FIGURE 4.22 Apogee velocity relation diagram.

TABLE 4.3 Calculation Results of Orbital Maneuver Velocity Increment

Latitude of Launch Site	Inclination of Transfer Orbit	v_a	v_s	Δv_i
28.5(°)	28.5(°)	1.595(km/s)	3.074(km/s)	1.835(km/s)

4.5.2 Propellant Budget Analysis

For a GEO spacecraft, the Earth's non-spherical gravity and solar radiation pressure perturbation will cause its west-east (longitude) drift, and the solar and lunar gravitational perturbation will cause its north-south (latitude) drift. The spacecraft needs to carry out periodic position maintenance to keep the longitude and latitude drifts within the allowable range of the flight mission. See Section 4.5.1 for the calculation of the velocity increment required to maintain the orbital position.

By adding the velocity increments required by orbital maneuver and orbital position maintenance, the velocity increment required over the lifetime of the spacecraft can be obtained:

$$\Delta V_A = \Delta V_i + \Delta V_{ANS} + \Delta V_{AWE} + \Delta V_{AT} + \Delta V_{AN} \tag{4.81}$$

where ΔV_i is the velocity increment required by the maneuver from the transfer orbit to the geostationary orbit; ΔV_{ANS} is the velocity increment required by the orbital north-south correction during the spacecraft lifetime; ΔV_{AWE} is the velocity increment required by the orbital west-east correction during the spacecraft lifetime; ΔV_{AT} is the velocity increment required by spacecraft attitude adjustment; ΔV_{AN} is other velocity increments.

According to international regulations, a certain velocity increment is also required to push the spacecraft out of the geostationary orbit at the end of its life so as to prevent it from becoming space debris in the geostationary orbit.

The main items of the propellant budget are calculated by the following equation with the above velocity increments:

$$\Delta m = m_0[1 - \exp(-\Delta V / w)] \tag{4.82}$$

where m_0 is the satellite mass (kg); w is the engine exhaust velocity, $w = I_{sp} \cdot g_0$; I_{sp} is the specific impulse (m/s); and g_0 is the gravitational constant of the Earth.

In the calculation of propellant mass consumption, the specific impulse of the engine needs to be determined, because the same propellant will produce different specific impulses for different engines. For example, in a specific spacecraft, the specific impulse of a 490N engine is 305s, while that of a 10N engine is only 260s. The 490N engines made by different development units also have different specific impulses due to their different design and technological levels. Even the engines produced by the same unit have different specific impulses. When calculating the mass consumption of propellants, we should consider the mixing-ratio errors of the delivered oxidizer and combustion agent, that is, the deviations between the actual delivery values of the two propellants and the required values. The propellant mass calculated at the above velocity increments is a consumption of one year, which should be multiplied by the spacecraft's in-orbit operating life to obtain the consumption caused by their perturbations.

In addition, the following factors should be considered: (1) position capture (that is, to drift toward a fixed point and then stop drifting); (2) the user's requirement for moving the orbital position during the spacecraft lifetime; (3) attitude control and adjustment to be reckoned in; (4) a certain decrease of engine thrust efficiency η caused by engine installation deviation and spacecraft center-of-mass error; (5) a certain amount to be reserved (generally 10% of the propellant mass should be reserved after orbit insertion as an allowance for deorbit and other unforeseeable circumstances); (6) propellant residue (unusable) in the tank and pipeline.

Table 4.4 gives an example of the propellant budget for a GEO spacecraft.

TABLE 4.4 Propellant Budget for a GEO Spacecraft

Flight Event		**Velocity Increment ΔV(m/s)**	**Specific Impulse Isp(s)**	**Efficiency η**	**Propellant Consumption Δm(kg)**	**3σ Error (kg)**	**Spacecraft Mass m(kg)**
Take-off							5040.00
Transfer-orbit attitude control					6.50		5033.50
Orbit maneuver	ΔV1	395.51	312	0.999	617.05		4416.45
	ΔV2	535.88			717.32		3699.14
	ΔV3	604.34			670.25		3028.89
	ΔV4	186.89			181.44		2847.44
Delivery 3σ dispersion correction		18.29			16.99	0.34	2830.46
Quasi-synchronous orbit attitude control					4.00		2826.46
Position capture		5.69	285	0.825	6.97	0.14	2819.49
North/south (N/S) position maintenance		356.41		0.851	392.37	7.85	2427.12
East/west (E/W) position maintenance		25.31		0.607	35.94	0.72	2391.18
Orbital maneuver		5.69		0.746	6.52		2384.66
In-orbit attitude control					15.00	0.30	2369.66
End-of-life deorbit		11.00		0.794	11.72		2357.94
Residual Sum of Squares (RSS)					7.89		2350.05
Mixing-ratio deviation					33.49		2316.56
Residual amount in storage tank and pipeline					29.58		2286.98
Propellant consumption					2753.02		
Propellant deadweight					83.45		
Injected propellant amount					2958.00		
Reserve					121.53		2074.00
Helium					8.00		

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CHAPTER 5

Spacecraft System Mission Analysis

Liu Jie

Beijing Institute of Spacecraft System Engineering (ISSE)

CONTENTS

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Mission analysis is the top-level design of a spacecraft system, which analyzes the user or country's requirements for a spacecraft mission, clarifies the input conditions of spacecraft design, and finds out the technological approaches to mission requirements. After receiving initial mission requirements from the user, the General Design Department should first analyze, clarify, and coordinate the user's mission requirements. Then it should select an orbit or constellation for the mission, analyze the payload and spacecraft makeup/concept suitable for the mission, analyze the existing key technologies, select and coordinate large-scale engineering systems, analyze the space environment of the spacecraft, and define the input conditions of general spacecraft concept design. Finally, the department should propose a general spacecraft conception.[1]

Based on the characteristics and objectives of space missions, this chapter describes the process of identifying and analyzing the spacecraft system missions.

5.1 CHARACTERISTICS AND BASIC ANALYSIS METHODS OF SPACE MISSIONS

5.1.1 Classification and Objectives of Space Missions

5.1.1.1 Classification of Space Missions

General space mission is a human-specific mission in which the spacecraft flies for a certain time in outer space on the predetermined orbit according to the law of celestial mechanics and then returns to the Earth or arrives at other celestial bodies.

The main space missions for space exploration and utilization are as follows:

1. Earth-orbiting satellite missions, such as very practical satellite communication, broadcasting, navigation, meteorology, reconnaissance, surveying and mapping, marine monitoring, Earth resources exploration, and Earth disaster monitoring.

2. Space environment exploration missions, using the Earth satellites or probes to explore and study the space environment.

3. Celestial observation missions, using astronomical satellites (such as solar telescopes) for the scientific observation of celestial bodies in order to have a deeper understanding of the universe.

4. New technology tests for spacecrafts, testing new technologies on various Earth-orbiting spacecrafts.

5. Manned space missions, using manned LEO spacecrafts for a variety of space science researches, the establishment of space industry, or the exploration and research of other celestial bodies.

6. Deep-space exploration missions, using deep-space probes to explore and exploit the moon and other planets.

The spacecrafts used for various space missions are different. Therefore, before designing a spacecraft, we should understand what its basic mission is. The content and technology of spacecraft design vary greatly with spacecraft mission.

5.1.1.2 Characteristics of Spacecraft Missions

In general, because of the high cost, space mission engineering is used to achieve specific mission objectives only when those objectives are not achievable on the ground and in the air, or when they are achievable on the ground and in the air but at higher cost. For example, communication and broadcasting can also be achieved on the ground, but with a much higher cost than satellite communications in the long-distance and large-coverage cases, especially in sparsely populated remote areas. Table 5.1 shows the space characteristics utilized by different space missions.

Table 5.1 also shows that the utilization of each space characteristic is different. Most of the missions, such as satellite communications, navigation, meteorology, and reconnaissance, take advantage of the global coverage of space. Among them, satellite communications have become an important industry. Space material manufacturing and space breeding, which are still in the development stage, are based on weightless and radiation environment, and may evolve into a new industry in the future. Exploiting the inexhaustible energy and natural materials of the universe to replace the limited Earth resources is an effective way for the future.

5.1.1.3 Objectives of Space Missions

The objectives of general space missions can be divided into two categories: basic objectives, which are the user requirements that shall be met by space missions; secondary objectives, which are the incidental targets that can be achieved by space missions on the premise of meeting the basic objectives.

Basic objectives are the most important and fundamental mission objectives the spacecraft design must achieve, while secondary objectives are minor incidental mission

TABLE 5.1 Space Characteristics Utilized by Different Space Missions

Space Characteristics	• Relevant Missions	Current Utilization	Mission Examples
Global coverage	• Communication and navigation Meteorology and Earth observation	• A basically mature application mission that will continue to improve in the future in terms of new exploration system and on-board intelligence	• International communication satellite BeiDou navigation satellite FY meteorological satellite Land resource satellite
Exoatmospheric observation	• Full-band scientific observation	• A basically mature application mission that will continue to expand in the future in terms of new observation method and other aspects	• Hard X-ray astronomical satellite
Zero-g environment	• Space material processing Space breeding	• Initially developed and expected to be applied in large scale in the future	• Space station
Resource development	• Lunar resources development Mars resources development	• The development of the relevant technology has just begun	• Manned lunar base Manned Mars base
Space exploration	Deep-space exploration Comet exploration	• Complete initial flight missions and landing missions and accelerate the development of manned space exploration in the future	Mars probe

objectives. The basic objectives are derived from the task description given by the user. The design process of spacecraft system is to carry out continuous iterative design work around the basic objectives, and always check whether the work in each process is consistent with the basic objectives.

The basic objectives are usually relatively fixed, while the auxiliary objectives may have to change frequently to meet the user's needs and to continuously improve the application potential of a space mission plan.

5.1.2 Basic Methods for Space Mission Analysis

5.1.2.1 Contents of Mission Analysis

After receiving the spacecraft mission requirements from the user, the system designers can carry out mission analysis. The mission analysis is to make clear the basic requirements of the user's mission, and to find out the basic technical approach to accomplishing the mission. If the user's requirements are unreasonable or the spacecraft development department encounters technical difficulties, it is necessary to coordinate with the user or modify the user's requirements.[2]

After making clear the user's basic requirements for a spacecraft mission, the general designers should fully understand those requirements, and analyze their rationality, correctness, and integrity. Meanwhile, the system analysis is carried out to find out the basic technological approach. Through the mission analysis, the basic technological approach to the user's spacecraft mission requirements is found out. The basic technological approach includes the selection and conception of orbit, payload, spacecraft platform, and large systems. The basic technological approach at this stage is only a brief design and analysis, which can be used as a reference for the overall design at the next step.

5.1.2.2 Basic Technological Approach in Mission Analysis

The basic technological approach is usually found out by the following three steps. First, develop a list of common alternatives. Second, establish an analysis tree. Third, prune the analysis tree. Next, the establishment of a national disaster mitigation satellite mission (hypothesized) will be taken as an example to describe how to identify a basic technological approach.

Develop a list of common alternatives: There may be numerous optional alternatives. However, under the existing background, the general designers have accumulated a lot of mature experience. Large mature engineering systems are available for launching all kinds of orbits. The spacecrafts are equipped with mature equipment, subsystems, and even public platforms. For the newly developed spacecrafts, the relevant foreign literature can also be referred to. Therefore, the general designers can choose the right designs from a limited number of alternatives. For the disaster mitigation satellite mission, a simple list of alternatives is developed, as shown in Table 5.2. In the design, the general designers shall first select the alternatives meeting the user's mission requirements, and then, depending on special circumstances, may consider other alternatives that are not listed in the table.

Establish an analysis tree: After initial consideration, a variety of possible combinations can be suggested. To facilitate the discussion, analysis, and selection, an analysis tree can be established. When establishing an analysis tree, the general designers need to explore ways to reduce the combinations without missing the potentially important alternatives. The major influencing factors at the system level should be identified and placed at the top of the analysis tree. Because these factors often dominate the design process, the selection of other factors based on these factors can reduce the alternatives and simplify the analysis tree. The factors that have little to do with defining the overall concept, such as thermal control subsystem and even common platform (once the orbit and payload are determined, the lowest cost onboard common platform that meets the mission requirements can be selected) should be identified and not included in the analysis tree. Figure 5.1 gives an analysis tree to find out the technical approach to the design of a disaster mitigation satellite.

Prune the analysis tree: After the analysis tree is established, it should be examined and analyzed. By comparison, the better combinations are kept and the worse ones are deleted. For example, any launch vehicles with a carrying capacity larger than the designed spacecraft could meet the requirements. Then the least expensive one should be kept and the more expensive ones should be canceled. Of course, the designed spacecraft should be compatible with the interfaces of a variety of launch vehicles (including domestic and foreign ones) or should be adapted to other spacecrafts. In this way, there is greater flexibility in the selection of a launch vehicle, which makes it easy to achieve a lower launch cost.

TABLE 5.2 Simple Alternatives for Disaster-Mitigation Satellite Design

Category	• Optional Item	Optional Scheme	Constraints
Payload	• Observation system	• Optics, microwave	• According to the user's requirements, microwave detection can be selected to meet the all-day and all-weather needs
	• Frequency	• Optics: infrared light, visible light Microwave: L, S, C, X, and other frequency bands	• The higher the frequency band, the higher the resolution
	• Sensitivity	• High sensitivity	• The resolution depends on the user requirements
	• Communications	• Standard satellite-ground communication	In accordance with ITU regulations and user requirements
Platform	• Attitude and orbit control system	• Control: spinning stabilization, triaxial stabilization, gravity gradient stabilization Propulsion: cold gas propulsion, single – component propulsion, dual – component propulsion, electric propulsion	Depending on the payload, orbit selection, and external constraints
	• Energy	• Source: solar energy, chemical battery, nuclear power supply, etc. Form of solar array: single solar array, double solar array, fixed solar array, rotating solar array	
	• Thermal control	• Passive thermal control, active thermal control	
Orbit	• Orbit type	• Geosynchronous orbit, sun-synchronous orbit, highly elliptical orbit	• Depending on payload characteristics, user requirements, and constraints
	Orbital altitude	• Low Earth orbit, medium Earth orbit, high Earth orbit	
	• Orbital inclination	• Different orbital inclinations	
Launch system	• Launch vehicle	• Long-March II, III, IV, V sockets	• Depending on the spacecraft and orbit
	• Launch site	• Launch sites in Taiyuan, Jiuquan, Xichang, Wenchang, and other places	
TT&C system	• TT&C station	• China Xi'an Satellite Control Center S, C, Ku, and other TT&C frequency bands	• Depending on the spacecraft type

If the general designers choose by comparison several design combinations from the concepts numbered 1–9 as basic technological approaches, those approaches can be used to develop the general concept. Of course, with the deepening of the overall spacecraft concept, this analysis tree will be updated and re-evaluated by the general designers.

5.1.3 Constraints on Space Mission Design

5.1.3.1 Analysis of Environmental Constraints

The constraints on overall spacecraft concept design include the user's requirements for the developer, the constraints on spacecraft engineering systems (launch vehicle, launch

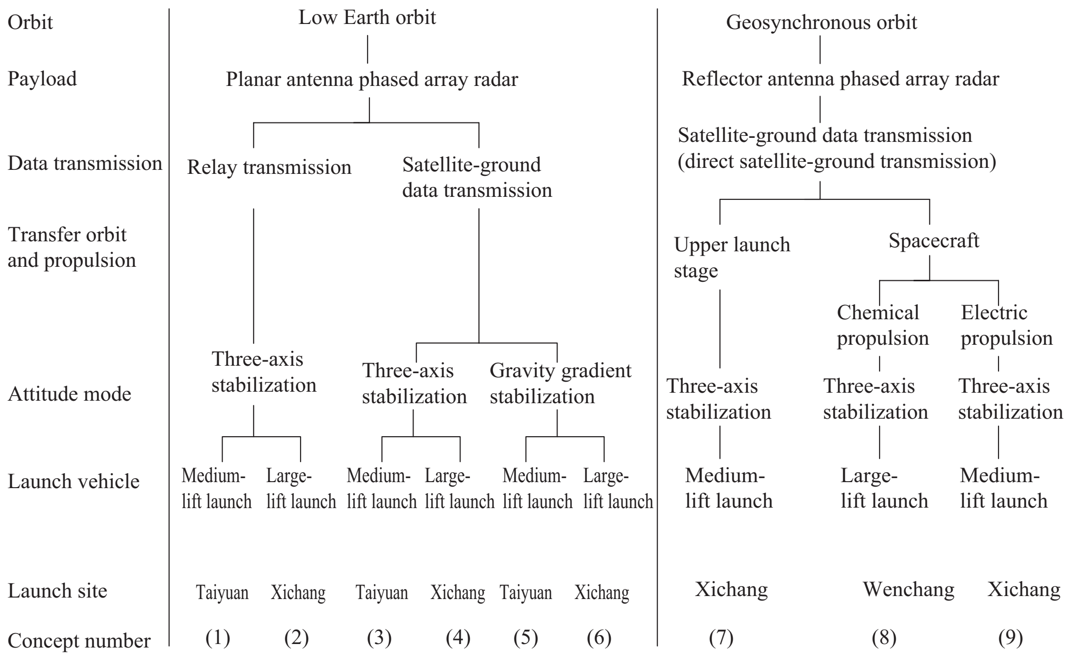


FIGURE 5.1 Hypothesized technological-approach analysis tree for disaster mitigation satellite.

site, TT&C center, ground application center, etc.), the impact of in-orbit space environment, the limitations of the existing technological base, and other constraints.

According to the system engineering concept, the constraints constitute a spacecraft environment system. Therefore, these environmental constraints must be first determined, analyzed, and coordinated in the overall spacecraft concept design. Then, under these constraints, the overall spacecraft concept design shall be carried out to ensure that the developed spacecraft can meet the user's requirements for the developer and can adapt to these environmental constraints.

5.1.3.2 User's Constraints on Mission Requirements

Technical requirements are the user's top-level constraints on spacecraft design system. In the overall spacecraft concept design, the general designers shall take these constraints as both the original design basis (input condition) and the final goal.

In addition, the development cost and lead time are also constraints. The development cost required by the user is also a constraint to be considered in the overall concept design. The amount of development funds directly affects the overall concept, performance, and lead time of a spacecraft. The lead time in the user requirements is also a constraint on the overall concept design. If the overall concept design is too complex, the lead time will be too long to accomplish the mission on schedule. This will not only affect the user's spacecraft application, but also affect the cost and benefit of the spacecraft developer.

5.1.3.3 Limitations of the Existing Technological Base

The existing technological base includes national industrial base and the developer's technological level and management ability, which can be divided into development means (including software and hardware) and personnel quality.

The high performance of a spacecraft needs to be guaranteed by high-performance raw materials, components, processing equipment, measuring equipment, and test equipment. The poor performance of these materials and equipment will make it very difficult to develop a high-performance spacecraft. Their performance is guaranteed by national industrial base. For example, a load-bearing structure with high stiffness and strength can't be built without high-performance carbon fiber; a subsystem-level instrument with a long life, high reliability, light weight, and high-performance can't be built without the radiation-resistant components with a long life, high reliability, and high integration; a high-performance equipment or even spacecraft can't be built without high-precision test devices.

The technical level includes design level and technological level. The design level is mainly reflected in the new problems and technologies found in the development of a spacecraft. With the technological development of large complex spacecrafts, the dynamics problems such as multi-body, flexibility, and sloshing have been found in the overall spacecraft design. Therefore, the designers need to study and analyze these new complex dynamics problems (through modeling, algorithm, and software) and suggest how to avoid and overcome the impact of these problems on a spacecraft. Now, another development trend of spacecrafts is miniaturization, so spaceborne mechanical/electrical/thermal multi-functional structures, micro-electromechanical components, cable-free design technology, and integration technology are emerging. These new technologies are constantly being developed and applied, so the spacecrafts become smaller and smaller. Similarly, if the technological level is low, high-performance components cannot be developed even with high-performance raw materials and elements. For example, when the technological level is poor, a load-bearing structure with high stiffness and strength can't be built even with high-performance carbon fiber.

The technical quality of satellite designers is also important. If advanced development means (including hardware and software) are available, but the designers are not familiar with, not proficient in, or even unable to use these development means, high-performance spacecrafts still can't be designed.

5.1.3.4 Other Constraints

Other constraints should also be taken seriously. For example, the radio frequency (RF) bands used by various spacecrafts, as specified by International Telecommunication Union (ITU), must be complied with; otherwise, the spacecrafts will interfere with each other and can't work normally. As RF resources are limited, coordination is required if there is interference with the spacecrafts from other countries, even if the frequencies used are within the ITU limits. Especially in geostationary orbit, the communications satellites are already crowded and the available frequencies are limited. A designed frequency cannot be used if it is not registered in advance, or has been registered but without coordination with an adjacent satellite.

In addition, some regulations and requirements related to international treaties and national policies should be considered as constraints in the system design process. For example, the international treaties stipulate that the launch vehicles and spacecrafts should take measures (such as the discharge of residual propellant and the passivation of batteries) to avoid or reduce the production of space debris.[3]

5.2 ANALYSIS PROCESS OF SPACECRAFT SYSTEM MISSION

5.2.1 Analysis Process and Contents of Spacecraft System Mission

The general analysis process of a spacecraft system mission is shown in Figure 5.2. It starts from the user's initial technical requirements for the satellite to the completion of preliminary system conception and overall mission analysis process.

The specific analysis of a spacecraft system mission is as follows.

5.2.1.1 Analysis of Large System Constraints

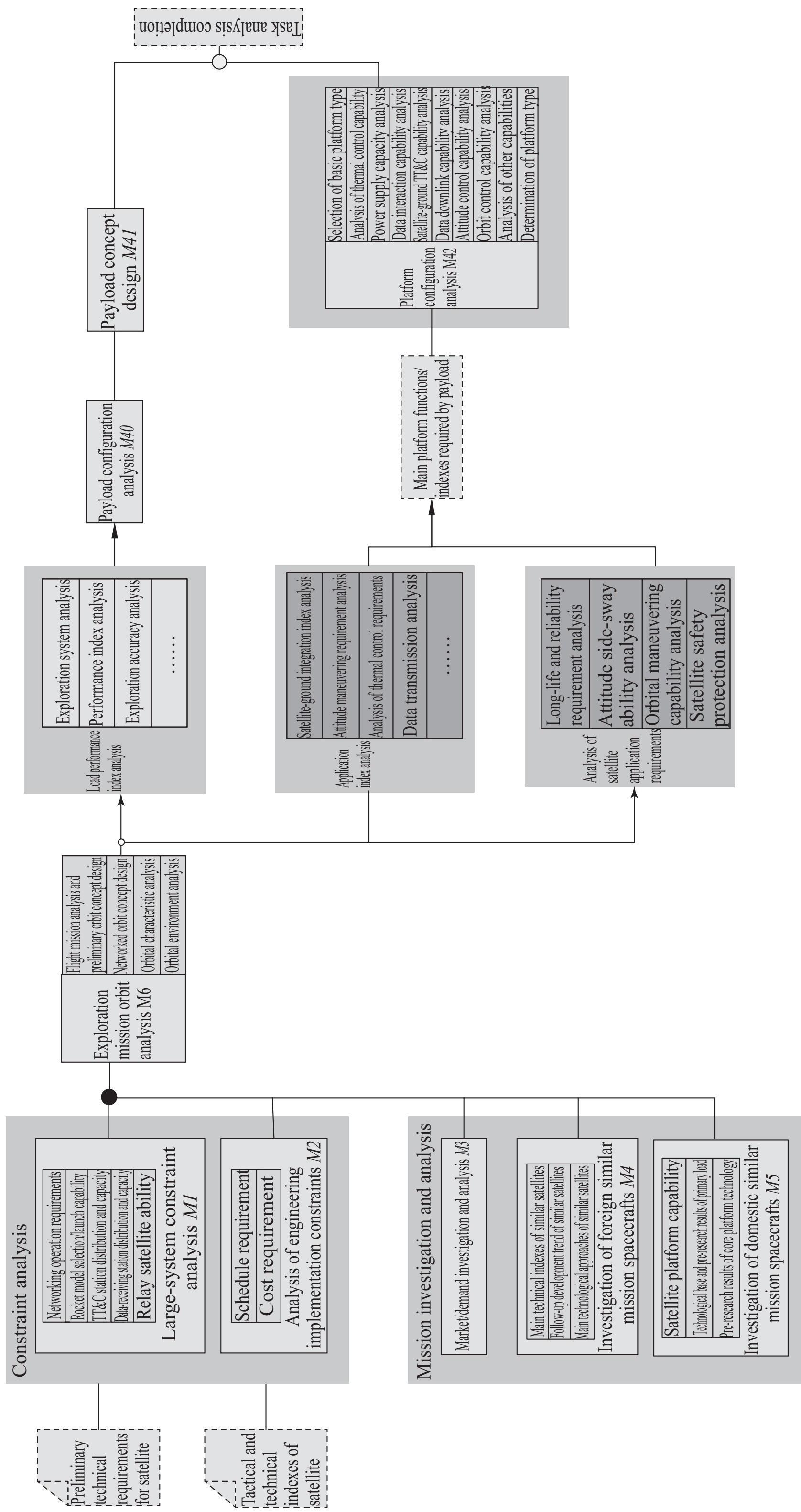
5.2.1.1.1 Purpose and Tasks Analyze the constraint conditions and boundary conditions of various large systems in spacecraft engineering. Identify the main interface relations between large systems in the engineering implementation as a precondition of system concept design, so as to ensure the optimization and coordination between large systems.

5.2.1.1.2 Analysis Elements and Contents

1. Networking operation requirements
2. Rocket model selection/launch capability
3. TT&C station distribution and capacity
4. Data-receiving station distribution and capacity
5. Relay satellite ability

5.2.1.1.3 Main Process

1. Sort out the common and special needs of spacecrafts and clarify the division of tasks among large systems.
2. Comb the requirements for other large systems, including the requirements for the geographical location and service time of the launch site, for the maintenance of the TT&C orbit, for the initial orbit of the carrier rocket, and for the data interface of the application system.
3. Identify the constraints of large systems to other systems, such as the transport, storage, and test environment conditions of the launch site, the mechanical and EMC environment conditions of the launch vehicle, the resource conditions of the TT&C



General analysis process of spacecraft system mission

FIGURE 5.2 General analysis process of spacecraft system mission.

station, the G/T value of the data transmission station, and the resource conditions of available stations.

4. Sort out the satisfiability of large systems with the interfaces, and identify the items that do not meet the requirements.

5.2.1.2 Analysis of Engineering Implementation Constraints

5.2.1.2.1 Purpose and Tasks Analyze the constraints on the whole spacecraft project to determine the external support and limitations to project implementation, which shall be considered in the system concept design to ensure the optimization and implementability of the system concept.

5.2.1.2.2 Analysis Elements and Contents

1. **Analysis of basic engineering conditions:** analyze the technical basis related to the project implementation to determine whether the technologies are mature or technical breakthroughs can be achieved within the project implementation period.
2. **Analysis of peripheral technical support:** analyze the technical support available through procurement, cooperation, and other means, determine whether the procurement channel and period of the imported equipment and devices meet the project requirements, and identify the availability of international and domestic resources such as external calibration field and precision orbit determination.
3. **Analysis of project schedule requirements:** to determine the time resources available for project implementation.
4. **Analysis of project consumption requirements:** to determine the cost input required for the project implementation.

5.2.1.2.3 Main Process

1. User demand coordination and confirmation.
2. Comprehending of National approval policy and relevant international treaties.
3. Investigation and preliminary coordination of the technological base.
4. Investigation and preliminary coordination of the procurement and cooperation.

5.2.1.3 Demand Investigation and Analysis

5.2.1.3.1 Purpose and Tasks Analyze the user's mission requirements, define the input conditions of spacecraft design, analyze the contribution of the mission to the user and other related industries, understand the demand for this mission in various fields, and evaluate its market value.

5.2.1.3.2 Analysis Elements and Contents

1. The information available for the mission and the effect achieved.
2. The demands for the mission in the application domain.
3. The demands for the mission within the business scope of various national departments.

5.2.1.3.3 Main Process

1. Identify the users' demands for this mission, and understand what information and effects its applications need to obtain.
2. Investigate similar applications at home and abroad, and learn about the encountered problems and the obtained achievements to guide the design work in this mission.
3. Investigate the demands for this mission within the business scope of various national departments to understand its extended application value.

5.2.1.4 Investigation of the Spacecrafts in Similar Missions at Home and Abroad

5.2.1.4.1 Purpose and Tasks Investigate the R&D status of similar spacecrafts at home and abroad, identify the relevant technical indicators and key technological solutions, and analyze the trend of technological development to provide a basis for general development thinking.

5.2.1.4.2 Analysis Elements and Contents

1. Analyze the relevant development information of foreign spacecrafts of the same type, including spacecraft parameters, orbital parameters, payload technology indicators, and application indicators.
2. Analyze the implementation form and capability elements of similar foreign spacecrafts, including high-efficiency payload configuration mode and load-platform integration design, common platform technology and series configuration concept, methods of in-orbit test and calibration, and methods of realizing the long life and high reliability of the whole satellite.

5.2.1.4.3 Main Process

1. Conduct this analysis in parallel with user requirements analysis, determine the spacecraft mission type and its application fields, and identify the key technical difficulties in user requirements. Then use this information as index to study the

spacecrafts of the same mission type at home and abroad, and collect comprehensive spacecraft development parameters.

2. Further analyze the implementation form and capability of key technical indicators at the satellite system level based on the collected development parameters of several similar foreign spacecrafts, and find out the technological development trend.

3. Analyze the demands and compare the spacecrafts in terms of their performance and capability in key technical indicators, and identify and determine the development ability gap between China's spacecrafts and their foreign counterparts, which will provide a basis for further shaping the overall spacecraft development idea in line with the actual situation.

5.2.1.5 Analysis of Exploration Mission Orbit

5.2.1.5.1 Purpose and Tasks Orbit design is an important part of the overall spacecraft design. It should meet the relevant requirements of a flight mission. A reasonable orbit type and nominal orbital parameters should be determined according to the requirements of the flight mission. Reasonable orbital parameters should be selected according to the characteristics of the spacecraft mission and payload. For the spacecraft systems with specific requirements (such as deep-space exploration satellites), the orbital parameters that meet the requirements should be selected.

5.2.1.5.2 Analysis Elements and Contents Analysis elements and content include spacecraft flight mission analysis, orbit constraints analysis, orbit type determination and determination of orbit altitude, inclination, eccentricity, regressive characteristics, etc.

5.2.1.5.3 Main Process

1. Summarize and sort out the spacecraft requirements, analyze its mission characteristics, and select an orbit type suitable for the flight mission, such as sun-synchronous regressive orbit, frozen orbit, general inclination orbit, and geosynchronous orbit.

2. Analyze the characteristics and capabilities of the main spacecraft payload.

3. In the whole process of orbit design, full consideration should be given to the supporting capacity of the ground TT&C system, the performance of the carrier rocket, the geographical locations of the launch site and landing site, the configuration of the ground application system, and the requirements and limitations on the relevant on-board subsystems.

4. Design a reasonable orbit according to the requirements for revisit, coverage, propellant, life, energy balance, and thermal control. The specific factors to consider include payload application analysis, sub-satellite trajectory analysis, orbit adjustment analysis, fuel consumption analysis, orbital environment analysis, carrying capacity analysis, and thermal control analysis, as well as the analysis of the interference with other similar satellites in orbit.

5.2.1.6 Networked Orbit Concept Design

5.2.1.6.1 Purpose and Tasks Some missions require the spacecraft to run in a network. The problem to be solved in the design of a networked orbit is to select appropriate single-satellite orbital parameters and a proper networking operation concept according to the mission requirements and the characteristics of a single-satellite orbit. It includes two aspects. First, design a reasonable orbit for the successor satellite that is based on and networked with the orbit of the existing satellite. Second, determine the orbital parameters of a single satellite according to the number of networked satellites, coverage and revisit requirements, and design the networking operation concept.

5.2.1.6.2 Analysis Elements and Contents

1. System requirements, main tasks, and mission analysis
2. Network type determination
3. Design of single-satellite orbital parameters
4. Networking concept design

5.2.1.6.3 Main Process

1. Summarize and sort out the requirements for spacecraft networking, analyze the mission characteristics, and select a suitable networking mode, such as coplanar networking, non-coplanar networking, equal-phase networking, or inequal-phase networking.
2. Analyze the characteristics and capabilities of the main remote-sensing payloads on the spacecraft. Analyze the resolution and breadth of different optical and microwave payloads.
3. Design the orbit concept for a single satellite. Design the network concept according to the number of networked satellites and the coverage and revisit requirements. If there is no reasonable network concept, the optimization of the single-satellite orbit concept shall be continued and then followed by the networking concept design. This is an iterative design process.

5.2.1.7 Analysis of Payload Indexes and Configuration

The analysis of payload indexes and configuration is a key part of mission analysis. For the spacecrafts in different fields, the characteristics of payload task analysis are different. According to the characteristics of different mission applications, an appropriate payload is selected and the payload configuration concept is analyzed.

5.2.1.7.1 Purpose and Tasks According to the user needs and actual application needs as well as the ability of scientific implementation, the payload indexes are preliminarily analyzed, and the payload configuration is determined.

5.2.1.7.2 Analysis Elements and Contents

1. Application demand analysis: analysis of specific application demands and user-oriented requirements in different fields, such as remote sensing, communications, navigation, and deep-space exploration.
2. Payload type analysis: analysis of optical payload, microwave payload, etc.
3. Analysis of observed objects: analysis of main observed objects and their radiation and geometric characteristics
4. Analysis of main payload indexes
5. Development of payload engineering prototype

5.2.1.7.3 Main Process

1. Fully communicate with the user, define the type and application of the spacecraft mission, and analyze and refine specific observation mission requirements (including observation area and observation object).
2. Fully analyze the characteristics of the observation area and object, and determine the basic type of the required payload and the indexes related to main observation performance according to the orbit information.
3. Fully investigate the progress in relevant fields at home and abroad, analyze the engineering difficulties and summarize the technological breakthroughs in accordance with the development condition of payload engineering prototypes, and analyze the ability to satisfy the requirement for each payload index.
4. Review the satellite ability to complete the payload mission in consideration of the spacecraft platform capability and data transmission capability. For the spacecraft system carrying a variety of payloads, analyze the payload synergy concept.
5. Develop the payload configuration concept and the main performance indexes of each payload.

5.2.1.8 Analysis of Spacecraft Service Requirements

5.2.1.8.1 Purpose and Tasks Identify the key links, components, and indicators affecting the use of the spacecraft according to the requirements for in-orbit operation environment and service life, and thus obtain the service-related technical requirements (such as spacecraft operation mode).

5.2.1.8.2 Design Elements and Contents

1. Analysis of data transmission requirements and strategies
2. Attitude maneuvering analysis
3. Analysis of orbit determination accuracy
4. Analysis of time synchronization accuracy
5. In-orbit deformation analysis
6. Analysis of moving parts
7. Analysis of solar array battery degradation
8. Analysis of thermal control material degradation
9. Redundancy analysis of key components
10. Analysis of operating mode

5.2.1.8.3 Main Processes

1. Analyze the life of the current model in accordance with the in-orbit spacecraft experience and long-life index, and identify the weak links affecting the life of spaceborne moving parts.
2. According to the analysis of space environment, calculate the change of solar incident angle and the accumulated flux of various radiations, and propose the requirements for the degradation performance of solar array batteries.
3. Calculate the space radiation and other environmental parameters experienced by the spacecraft during the required life, and propose the requirements for the degradation effect of thermal control materials.
4. Identify the key components by analogy, task analysis, and other methods, sort out the weak links in components that need the redundancy design, and determine the redundancy concept.

5.2.1.9 Preliminary Design of Payload Concept

5.2.1.9.1 Purpose and Tasks Analyze the mission in accordance with the payload index and satellite-ground integration index proposed by the user in the general development requirements. Decompose the payload index layer by layer to form the main technical parameter system for payload development. Determine the payload's operating mode; define the payload function; sort out the interfaces between payload subsystem and other on-board subsystems to obtain the overall payload concept.

5.2.1.9.2 Analysis Elements and Contents

1. Top-level index decomposition
   a. Clarify the overall requirements for various subsystem-level technical indicators in the process of payload development.
   b. Analyze the influence of relevant on-board subsystem-level technical parameters on payload task indicators.
2. Functional design
   a. Sort out the payload functions necessary for the completion of the tasks assigned by the user, including the communication and data transmission between payload and other subsystems.
   b. In general, the functional design shall be comprehensive, rational, and ease to use.
3. Payload performance requirements: specific performance requirements are put forward for different types of payloads.
4. Makeup of payload subsystems
   a. Single-unit makeup concept of the payload subsystems
   b. Interface relationships among individual units in the subsystems
   c. Basic working principle of subsystems
5. Operating mode of subsystems

   The payload subsystems are the equipment not in long-time operation. Generally, they are in waiting, operating, or calibration mode.
6. Mechanical, electronic, and thermal design of payload subsystems
   a. Mechanical design concept of payload subsystems (including weight, size, installation accuracy, and mechanical properties)
   b. Electronic design concept of payload subsystems (including power supply, communications, information flow, and data flow)
   c. Thermal control design concept supported by thermal analysis

5.2.1.9.3 Main Process

1. Analyze the user's top-level indexes one by one, and then identify the main technical indexes of each influencing factor, which will become the constraints on payload subsystem design. Propose the required technical indexes for the related on-board subsystems.

2. Design the functions and operating modes of payload subsystems to ensure that each function meets the actual in-orbit service requirements of the payload, and that the design is reasonable and convenient to use.

3. Determine the division plan of individual units and functional modules in the payload subsystems and the working principle of the subsystems on the basis of full communication with subsystem designers; identify the electrical and communication interfaces between units to establish a block diagram.

4. Complete the following work on the basis of full communication with subsystem designers: formulating a main technical plan for every payload; on-orbit mechanical/thermal environment analysis; selection of key component types; identification of key technologies and risks; reliability analysis; communicating and coordinating with the designers of other onboard subsystems on specific interfaces. The coordination contents shall be implemented as a coordination summary.

5.2.1.10 Platform Configuration Analysis

5.2.1.10.1 Purpose and Tasks The platform configuration shall be analyzed according to the initial technical requirements of the spacecraft, the constraints on large systems and engineering implementation, and the main platform functions and indicators required by payloads. By optimizing the selected models and configurations and referring to the contents of single-unit type spectrum, the configuration and inheritance of the platform and each product can be determined.

5.2.1.10.2 Design Elements and Contents

1. Analyze the mission requirements and spacecraft technology indicators. Analyze the payload quality assurance requirements according to the requirements for spacecraft orbit, space environment, life, function, and performance.

2. Clarify the constraints on large systems and engineering implementation, including networking operation, carrying capacity, launch site, TT&C station distribution, ground-data reception distribution, and relay-satellite transmission capacity.

3. The general principle for selection of a basic platform is to analyze the load-carrying capacity and power supply capacity of the platform according to the constraints on large systems and engineering implementation as well as the main platform functions and indicators required by payloads by fully inheriting the mature technology and common equipment of the previous spacecrafts.

4. After defining the configuration principle, clarify the main platform functions and indicators required by payloads based on the mission demand analysis, so as to complete the selection of the platform configuration. The main contents include:

   a. Analyze the thermal control capability according to thermal control requirements. An appropriate thermal environment is provided for onboard instruments and equipment by controlling the heat exchange inside and outside the satellite in a reasonable way. For the instruments or components with strict temperature requirements (such as camera and other critical equipment), consideration should be given to providing an appropriate interface temperature. In case of a change of the selected product model, a feasibility analysis should be carried out.

   b. Analyze the onboard data interaction ability according to payload data, platform data, and other information flows, and define the configuration and functions of onboard data management, so as to realize the whole-spacecraft telemetry data collection and organization, telecommand reception and output, injection data management and distribution, spaceborne program control and autonomous control, satellite time management, bus management, whole-spacecraft data management, and other functions. Analyze the feasibility of any change in the selected product model.

   c. Determine the satellite-ground TT&C system and define the functions like telemetry, telecontrol, ranging and timing as well as the tasks such as ground TT&C and relay TT&C according to the constraints on large systems and engineering implementation and the mission requirements. Analyze the feasibility of any change in the selected product model.

   d. Analyze the data downlink capacity according to the requirements for payload data transmission, define the data transmission channel configuration and antenna model, and realize the functions such as data compression and formatting and data transmission to ground stations or relay satellites. Analyze the feasibility of any change in the selected product model.

   e. Determine the attitude maneuver strategy according to the analysis of attitude maneuver imaging and its impact on payload imaging, and define the capability and model of attitude maneuver actuator. Analyze the attitude precision requirements of attitude and orbit control system according to the target-positioning accuracy requirements of the satellite, so as to determine the model and configuration of the sensors in the system. Analyze the attitude and orbit control ability of the platform according to the spacecraft service requirements and the requirements for attitude control ability during payload imaging, and define the configuration of this system. Analyze the feasibility of any change in the selected model of attitude and orbit control subsystem and products.

   f. Analyze the satellite structure and mechanism configuration according to the carrying capacity of the spacecraft.

5. Design the redundancy of key components and long-term moving parts according to the analysis of satellite service requirements and the design analysis of long life, reliability, and redundancy.

6. Determine the configuration and inheritance of each subsystem and product on the platform.

5.2.1.10.3 Main Process Select the model of onboard products by fully inheriting the mature technology and common equipment of the previous spacecrafts and complying with the preliminary technical requirements and tactical and technical index requirements for the satellite, the constraints on large systems and engineering implementation, and the payload's requirements for main platform functions and indexes. Analyze the platform's carrying capacity, power supply capacity, thermal control ability, data interaction ability, satellite-ground TT&C ability, data downlink ability, attitude control ability, orbit control ability, and structural load-bearing capacity. Determine the configuration and inheritance of onboard products by fully inheriting the mature technology and common equipment of the previous spacecrafts and considering the requirements for long life, reliability, and redundancy.

5.2.2 Analysis Example of a Mission with Typical Earth-Sensing Spacecraft System

Earth-sensing spacecraft is a spacecraft system widely used at present. The analysis of its mission is universal and can be used as a reference for all kinds of spacecraft systems in mission analysis.

The analysis process of a typical Earth-sensing spacecraft system mission is as follows:

1. **Mission requirement analysis:** Clarify the user's definition of spacecraft development task, the requirements for service technology indicators, and the development funds and costs through the analysis of mission requirements. Coordinate with the user multiple times during the analysis of this mission and its requirements.

2. **Select the orbit and its parameters:** Select the orbit type according to the mission requirements. For example, if the mission of a spacecraft is remote sensing of the Earth, the orbit chosen is generally a sun-synchronous (regressive) orbit of a few hundred kilometers. The orbital altitude determines the space environment, and thus determines the design requirements and design life of onboard instruments and equipment against space environment. The orbital altitude also has an influence on the selection of launch vehicle, the resolution and coverage of the remote sensor in the payload, and the power of data transmission transmitter.

3. **Design the payload performance:** Select the payload concept and design the payload parameters according to the mission requirements and the selected orbit. In this case, determine the resolution and field of view of a remote sensor according to the requirements for orbital altitude and service technology indicators. Initially design the size, weight, electrical power, and thermal control of the sensor, and propose the

requirements for pointing, telemetry, and telecontrol. Tackle the key problems in new technologies in advance.

Orbital parameters can be further designed according to the payload design parameters and the requirement for spacecraft re-entry period in the mission. A trade-off shall be stricken between payload design and orbit design. For example, the higher the orbital altitude, the lower the resolution of the remote sensor and the longer the re-entry period. In other words, to obtain high resolution in high orbit, a remote sensor with a large aperture, long focal length, and large size shall be designed.

4. **Design the spacecraft platform:** Design the spacecraft platform according to the payload design parameters. Once the payload design parameters are determined, an appropriate common spacecraft platform can be selected or a new platform can be designed, ultimately determining the size of the entire spacecraft. Spacecraft platform is to provide support for the payload and meet the payload pointing requirement. The related subsystems on the platform provide guarantees to the payload in the forms of power consumption (including long-term and short-term power consumption), telemetry parameter measurement, type and quantity of commands, data processing and transmission (storage capacity, transmission rate and bit error rate, etc.), and thermal control (such as high temperature and low temperature). The data transmission subsystem is selected and designed according to the data volume of the remote sensor. In addition, the designer should decide whether a relay satellite is needed and whether a relay data transmission subsystem should be designed. Therefore, the payload and the spacecraft platform are very closely related to each other with mutual influence and requirements.

   The design of a spacecraft platform includes the design of subsystem composition and concept, the configuration design (including the design of overall layout, shape and main force-bearing components, and the calculation of mass characteristics), and the analysis of subsystem performance parameters and overall performance parameters (overall dimensions, dry weight, propellant weight, electrical power, life, etc.).

   In the analysis of overall spacecraft concept, consideration should be given to the influence of spacecraft environment system, such as the load-bearing capacity, internal fairing space and mechanical environment of the launch vehicle, the communication link requirements of ground TT&C system (such as the equivalent isotropically radiated power EIRP and the G/T quality of the receiving system), the communication link requirements of ground application system, the requirements of the launch site, and the influence of space environment. In the design process, the relationship between spacecraft and orbit should also be analyzed. For example, due to the influence of various perturbations, the orbital elements will change, which requires the spacecraft to have the ability of orbit adjustment. In addition, the design life and reliability of the spacecraft should consider the relationship with the fuel carried by the spacecraft, the grade of the selected components, the design of backup parts, and the design of power supply system.

5. **Select large engineering systems:** Each large engineering system can be selected according to the initial platform choice and its design scale and the orbit choice. After decades of development, the large systems such as carrier rocket, launch site, and ground TT&C center have become very mature and have evolved into common and serialized service systems that can be directly provided for optional use. Among them, the ground application system (such as Earth communication station and remote-sensing data processing center) is generally customized and is specially purchased by the user or developed through bidding. When choosing large engineering systems, the general designers shall fully analyze their performance and their interfaces to the spacecraft, so that the designed spacecraft can be adapted to and matched with each system.

5.3 PRELIMINARY SYSTEM CONCEPTION

The preliminary conception of the overall spacecraft design is necessary in the mission analysis. It is mainly to carry out the preliminary demonstration and design iteration of each subsystem of the payload and platform.

After decades of research and development, the function, principle, technology, and design of each subsystem of the spacecraft platform have become mature. However, in spite of continuous technological development, the compositions of the subsystems are still quite different, and their performance is also very different. They should be selected and analyzed according to the payload concept and requirements. In the demonstration of the overall spacecraft concept, the concepts of the payload subsystems should be selected and analyzed first, and then each subsystem of the spacecraft platform should be selected and analyzed.

In the demonstration and design of the overall concept, the subsystem concepts selected by the general designers can be different. The difference in the selected subsystem concepts will have an impact on the generation of the overall spacecraft concept. In particular, the payload, control, propulsion, and power subsystems have the greatest impact on the overall concept.

This section only gives a general introduction to the main types and requirements of each subsystem of the Earth-applied spacecraft. Only when the main types and requirements of the subsystems are obtained can the subsystem concepts be further selected and demonstrated. The process of selecting and demonstrating the subsystem concepts of each spacecraft is different, and will not be specifically introduced here.

5.3.1 Preliminary Selection of Spacecraft Mission Orbit

With the development of applied spacecrafts over the decades, the process of choosing the orbit types for various spacecraft missions has basically matured. The general designers can easily determine the orbit type, which may even have been proposed in the user requirements. Table 5.3 lists several orbit types in common use and their applications. According to Table 5.3, the general designers can easily select the orbit types for various applied spacecrafts. In the demonstration of the overall concept, the general designer should, on the one hand, select specific orbital parameters and design an orbit. On the other hand, they

TABLE 5.3 Several Types of Orbits and Their Applications

Orbit Type	• Application
Geostationary orbit and its constellation	• International communications, regional and domestic communications and broadcasting, maritime communications, regional navigation, meteorological observation, etc.
Sun-synchronous (regressive) orbit and its constellation	• Earth resources exploration, global meteorological observation, global reconnaissance, space environment exploration, marine environment monitoring, etc.
Very low Earth orbit	• Recoverable remote-sensing satellites, manned spacecrafts, space shuttles, space stations, etc.
Critical inclination HEO and its constellation	• Long-term continuous observation and communication in middle and high latitudes
Combination of high, medium, and low Earth orbits to achieve global coverage	• Global mobile communication, global navigation, global observation satellite network, etc.

should devote more energy to analyzing the selected orbital parameters and suggest a kind of spacecraft concept to meet the user's requirements for the space mission.

The choice of spacecraft orbit or constellation will directly affect the overall concept and configuration design of a spacecraft. For example, when sun-synchronous orbit is selected for a meteorological satellite, the overall concept of the satellite can obtain global coverage, and the satellite body can achieve triaxial stability. In this case, the satellite configuration is generally designed into a cube with double solar wings, in which the payload is Earth-oriented and the solar wings are sun-oriented. If geostationary orbit is selected for the meteorological satellite, the overall concept of the satellite can achieve regional coverage. The satellite body is generally under simple double-spin stabilization, and the payload is Earth-scanning. The satellite configuration is generally cylindrical, and the solar arrays are body-mounted. On the other hand, the spacecraft concept also has some requirements for the orbit (or constellation). For example, the orbital altitude is required to be as low as possible in order to improve the ground resolution. In addition, the regressive (revisit) period required by the spacecraft concept needs to be realized by the orbital design. Although trade-offs are needed between them, orbit selection is the top-level spacecraft design and is one of the prerequisites or bases for spacecraft system concept design.

5.3.2 Preliminary Payload Conception

The selected payload type is primarily dependent on the space mission (the user's requirements). There are many types of payloads, each with many optional concepts.

Payload is one of the most important subsystems in the spacecraft that is finally provided to the user for service. The final characteristics and size of a spacecraft system concept depend on the type, function, and performance of the payload and its requirements for the spacecraft (especially its weight, size, and power consumption). The primary task of general designers in the system concept design is that after the analysis of the user requirements, the designers shall select and analyze the overall payload concept and its requirements for the spacecraft.

Before the selection and design of a payload, the designers shall have a deep understanding of various technical requirements for the payload. The technical requirements for different types of payloads are different. Here are the general technical requirements for several payloads:

1. **Various general requirements:** for orbit, mass, size, electric power, attitude pointing accuracy, attitude stability, attitude maneuver capability, telemetry parameters, telecommands, thermal control temperature range and temperature gradient, mechanical and space environments, life, reliability, etc.
2. **Communication requirements:** for coverage area, frequency band selection, saturated power flux density (SFD), EIRP, G/T, amplitude-frequency characteristics, in-band clutter, out-of-band rejection, anti-interference, destroy resistance, spot-beam antenna, nulling antenna, onboard data processing, encryption, antenna gain, polarization loss, duplexer isolation, multi-beam antenna isolation, etc.
3. **Earth observation requirements:** for optical camera spectrum, aperture, focal length, field angle, pixel resolution, ground resolution, modulation transfer function, signal-to-noise ratio; microwave remote-sensing frequency band, antenna size, transmitter output power, antenna gain, receiver sensitivity; data transmission rate, memory capacity, data compression ratio, bit error rate, transmitter output power, antenna gain, antenna pattern, etc.

5.3.3 Preliminary Subsystem Conception

5.3.3.1 Requirements and Types of Control Subsystem

5.3.3.1.1 Requirements for Control Subsystem The task of control subsystem is to control the orbit and attitude of the spacecraft. The attitude control requirements are determined by the concepts of payload and common spacecraft platform.

1. Payload requirements

   The payload needs to be oriented in inertial space. The entire payload needs to be oriented. For example, the payload (such as optical camera system) of an Earth observation satellite, astronomical satellite, or solar telescope needs to be oriented to the ground, sky, or sun. The spacecraft orientation is generally controlled by the control system. Some payloads require in-orbit fast attitude maneuver orientation, which is realized with two control methods: controlling the whole spacecraft into attitude maneuver, and controlling the payload into attitude swing. Some of the payload components are required to be oriented, for example, the Earth antenna is required to be oriented to the Earth. The specific requirements are as follows:

   a. **Direction of orientation:** as mentioned above, the relative reference datum needs to be determined.

b. **Range of orientation:** for example, the optical camera needs to swing to the left and right within the range of about 30°.

c. **Pointing accuracy:** namely the absolute angle control requirement for target pointing, such as 0.1°.

d. **Pointing stability:** the maximum change rate of the pointing angle, for example, 0.0001°/s.

e. **Maneuvering rate:** the angle of rotation per unit time when redirecting from one orientation to another.

2. The platform requirements for control subsystem orientation

   The orientation of data transmission antennas and TT&C communication antennas is to be verified. Most of those antennas are required to point to the Earth, and some even required to pointing to relay satellites. The antennas on a spacecraft with inter-satellite communication are required to point to other spacecrafts while capturing and tracking the target spacecraft. They are driven by a two-dimensional drive mechanism.

   The orientation during orbital transfer is to be verified. The engine used in the orbital transfer has the pointing requirement, which is determined according to the orbital transfer strategy.

   The orientation of solar wings is to be verified. The solar wings are required to point to the sun generally through one-dimensional rotation (or two-dimensional rotation in few cases).

3. General requirements for control subsystem

   The general requirements for control subsystem are: orbit type; subsystem concept (e.g., gravity gradient stability, spinning stability, or triaxial stability); the main function and performance indexes of control subsystem; orbit/attitude maneuver capability; the mass, size, and electric power of various instruments and equipment; telemetry parameters, telecommand, thermal control, and other interfaces; mechanical and space environments; life, reliability, etc.

5.3.3.1.2 Attitude Stabilization Concept Attitude stabilization concept is the main factor that decides the control subsystem concept. There are three main attitude stabilization concepts: gravity gradient stabilization, double-spin stabilization, and triaxial stabilization.

1. Gravity gradient stabilization

   Gravity gradient stabilization refers to the use of the moment generated by gravity gradient to keep the minimum rotational inertia of the spacecraft in the vertical direction of the Earth by means of an extensible pole on the spacecraft (withdrawn during launch and deployed after orbit insertion). This stabilization concept is relatively simple, but its attitude control accuracy is low, about 1°–5°. It is generally used for a low-accuracy spacecraft pointing to the Earth.

2. Double-spin stabilization

   Dual-spin stabilization is to maintain the spacecraft stability by using the gyroscopic inertia of a spinning body in inertial space. The concept is to use an Earth infrared sensor and a sun sensor to determine the attitude. Therefore, its attitude control accuracy is medium, about 0.1°–1°. It is generally used for a spacecraft that is perpendicular to the orbital plane and points to the Earth, such as a spinning spacecraft with Earth-oriented despinning antenna.

3. Triaxial stabilization

   Triaxial stabilization is to keep the three axes of the spacecraft in a certain orientation in orbit by using various actuators. The three-axis stabilization concept is to use an Earth infrared sensor, a sun sensor, a star sensor, and various gyroscopes (or magnetometers) to measure the spacecraft attitude. Therefore, its attitude control accuracy is higher and can be better than 0.1°. Most of the modern spacecrafts pursue triaxial stabilization.

5.3.3.2 Requirements and Types of Propulsion Subsystem

5.3.3.2.1 Requirements for Propulsion Subsystem The propulsion subsystem of the spacecraft can be used as either an independent subsystem, or a subsystem or an executive part of the control subsystem. It is generally regarded as an independent subsystem. The propulsion subsystem has two tasks: one is attitude adjustment and orbital maneuver during orbital transfer; the other is attitude and orbit maintenance when the spacecraft is working normally in orbit.

During the orbital transfer of a launched GEO spacecraft and the atmospheric reentry of a return spacecraft, a large-thrust solid engine can be used. A unified system with two-component liquid engines is often used for launching the GEO spacecraft. That is, during orbital transfer, a large-thrust engine (often 490 N) is used; after orbital transfer, the propellant in the storage tank continues to be used for in-orbit attitude and orbit maintenance, during which a low-thrust (e.g., 10 N) engine or a smaller thruster is generally used.

The requirements for propulsion subsystem include not only the subsystem concept but also total impulse, specific impulse, mixing ratio, thrust, duty cycle, residual amount, and working times (life). In addition, high-pressure vessels and flammable dangerous products are designed, so this subsystem must secure not only performance but also safety.

5.3.3.2.2 Types of Propulsion Subsystem The types, propellants, specific impulses, advantages and disadvantages as well as applications of the propulsion subsystems in common use are shown in Table 5.4.

TABLE 5.4 Types of Several Typical Propulsion Subsystems

Type	• Propellant	Specific Impulse (s)	Advantage and Disadvantage	Application
Solid engine	• Double-base propellant	• 280	• Simple, reliable, and low-cost	• Apogee maneuver and return braking
Cold gas propulsion	• Nitrogen, helium, etc.	• 50~75	• Simple, pollution-free, low-performance	• Small satellites
Single-component propulsion	• Anhydrous hydrazine, H_2O_2	• 200	• Simple and reliable, low-performance	• Maneuver-less orbits and small satellites
Dual-component propulsion	• MMH and N_2O_4	• 310	• High performance but complex system	• Orbital maneuver
Arc heating propulsion	• Nitrogen, ammonia, hydrogen, etc.	• 450~1500	• High performance but large power consumption	• Attitude and orbit adjustment, orbital transfer
Ion electric propulsion	• Xenon	• 2000~6000	Very high performance but very large power consumption and small thrust	• Attitude and orbit adjustment, position maintenance

5.3.3.3 Types and Requirements of Power Subsystem Concept

5.3.3.3.1 Requirements for Power Supply Subsystem The function of power subsystem is to provide electric energy to the spacecraft in the sunshine period and Earth shadow period. The power subsystem should have the functions such as power generation, energy storage, power distribution, busbar voltage regulation, and battery charge/discharge control. Sometimes, a secondary power supply is needed to transform and stabilize a variety of voltages. The spacecraft life requirement is very important to the power supply subsystem, because the performance of various spaceborne power supplies will gradually decline with the increase of in-orbit life.

At present, the power supply subsystem consisting of both solar arrays and batteries (often called primary power supply) is widely used. In addition, it includes power control devices (battery charge/discharge controller, solar array shunt regulator, busbar regulator, etc.). According to the above functions, secondary power supply (for regulating the voltage of primary power supply into various voltages required by the instruments and equipment in each subsystem) is sometimes incorporated into the power supply subsystem.

The general spacecraft configuration has special requirements for the type of solar arrays. In the spin-stabilized spacecrafts, the body-mounted solar arrays are generally installed. The three-axis stabilized spacecrafts (such as the spacecrafts in solar synchronous orbit and geosynchronous orbit) generally need one-dimensional sun-pointing solar arrays, and a few of them (those in non-SSO low orbit) need two-dimensional sun-pointing solar arrays.

5.3.3.3.2 Types of Spacecraft Power Supplies The power supply types, energy conversion devices, and their applications are shown in Table 5.5.

TABLE 5.5 Types of Several Typical Spacecraft Power Supplies

Type of Power Supply	• Energy Conversion Device	Space Applications
Chemical primary battery	• Silver-zinc battery, lithium battery, zinc-mercury battery	• Used for short-term LEO spacecrafts
Chemical battery	• Cadmium-nickel, nickel-hydrogen and Li-ion batteries, etc.	Used with solar batteries to power the spacecraft during the Earth shadow period, and charged by solar batteries during the sunshine period
Fuel cell	• Hydrogen-oxygen fuel cell	• Used for short-term low orbit spacecrafts. In a manned spacecraft, the water and heat from fuel cell decomposition can be used to support the astronaut's life
Solar battery	• Silicon, gallium-arsenide, and indium-phosphide solar batteries, etc.	• Used with chemical batteries to power the spacecraft and charge the batteries in the sunshine period; used in majority of the existing spacecrafts
Nuclear power	Thermocouple and thermionic converters	• Used for the spacecrafts with poor sunlight condition or great power demand, such as deep-space probes or super-power space-based radar satellites

5.3.3.4 Requirements and Types of TT&C and Data Management Subsystem Concept

5.3.3.4.1 Types of Subsystem Concept The TT&C and data management subsystem is used for spacecraft tracking, orbit measurement, telemetry, remote control, and data management.

With the development of space technology, the subsystem of TT&C and data management has been improved continuously. In the early spacecrafts, the telemetry system, telecontrol system, and tracking system were decentralized and independent from each other. Later, they developed into a unified microwave system. In the late 1970s, the European Space Agency developed the Onboard Data Handling (OBDH) subsystem. In the 1990s, the Onboard Data System emerged.

1. Decentralized systems

   The telemetry system, telecontrol system, and tracking system are independent from each other. In addition to video independence, the telemetry system has its own transmitter and antenna, the telecontrol system has its own receiver and antenna, and the tracking system has its own transponder and antenna.

2. Unified microwave system

   The telemetry system and the tracking system share a transmitter, and the telecontrol system and the tracking system share a receiver. The telemetry and telecontrol signals and the tracking and ranging tones are modulated onto the same carrier wave. This allows telemetry and telecontrol to keep only the video portion, thereby simplifying the device, reducing the size, power consumption and mass, saving the frequency resources, and avoiding electromagnetic interference.

3. Spaceborne data management system

   Apart from RF, the functions such as remote-sensing information, telecommand, program control, data storage, and autonomous control are integrated. In other words, the spaceborne computer is used for the integrated management of the on-board data, including time synchronization, program control management, autonomous subsystem control management, system-level safety management, data acquisition, data storage, data processing, data exchange, and connection of data stream to the RF band. In order to achieve the above functions, the related hardware, software, interfaces, and communications at the system, component, and circuit levels are comprehensively designed to expand the functions, achieve the resource sharing (a unified business design for the same hardware and software) and interactive support, and avoid repeated backups. In this way, the reliability is improved, the connecting cables between devices are decreased, and the spacecraft weight and cost are reduced.

   The OBDH subsystem can connect the central unit to all kinds of remote terminals through the data bus and can distribute the data acquisition, data processing, and instruction-issuing capabilities to several decentralized modules. Compared with centralized management, this management concept can simplify the interface design, data transmission, software design, and testing. It facilitates not only the modular design but also the function expansion and reconfiguration.

4. Spatial data system

   It is developed on the basis of spaceborne data management system. By extending its scope to the operational data of the payload and expanding its function to the service layers of spatial data network, the spatial data system has become a unified information system for the spacecraft. In this way, a larger integration can be achieved, that is, the space-based network and ground Internet can be integrated to achieve the integration of spacecraft and ground-based TT&C and application system.

5.3.3.4.2 Requirements for Subsystems In addition to the general requirements mentioned above, the main requirements for TT&C and data management subsystem are as follows:

1. What kind of orbit (LEO, MEO, HEO, lunar orbit, or deep-space exploration orbit) is adopted – the orbital altitude will affect the communication link design.
2. Whether to transmit the payload data of Earth observation spacecraft – the transmission of this data will affect the transmission data rate.
3. Whether to transmit data via a relay satellite (at the data rate of about 300 Mbps) – if so, an automatic tracking antenna is needed.
4. On-board storage and processing.
5. Uplink and downlink RF bands (C, S, Ku, and other bands).
6 Modulation and coding scheme.

7. Bit error rate (10^{-5} for uplink, and 10^{-4} for downlink).
8. Antenna requirements (radiation pattern, gain, polarization, side lobe).
9. Number of telemetry parameters and telecommands.
10. Anti-jamming and encryption requirements.

5.3.3.5 Types and Requirements of Thermal Control System Concept

5.3.3.5.1 Requirements for Thermal Control System The thermal control subsystem is related to all other subsystems. Its task is to keep each subsystem within its operating temperature range at each stage of in-orbit operation. The required operating temperature ranges of onboard equipment are generally as follows:

1. The operating temperature range of general electronics is 0°C–40°C.
2. The operating temperature range of cadmium nickel battery is 5°C–20°C.
3. The operating temperature range of solar battery is controlled within −100°C to +100°C (required at the low end, because the efficiency of solar battery will increase with the decrease of the operating temperature).
4. The temperature of liquid propellant is required to be maintained at 7°C~35°C. As the pipelines connected to the small thruster are installed at the far end without heat source, almost all of those pipelines need to be heated.
5. The infrared camera sensor is required to work at a very low temperature (less than 80 K), so passive radiation refrigeration is required. If a large cooling capacity is required, active Stirling cryocooler or pulse tube cooler should be used.
6. High-resolution cameras have very high requirement for their operating temperature ranges (some are required to be controlled within ±0.5°C~2°C) to reduce thermal deformation, so complex thermal control measures should be taken.
7. The high-power amplifier (≥100 W) needs local cooling.

Other requirements related to the design of thermal control subsystem include orbital altitude, solar angle, satellite structure materials, instrument heat output, and antenna and solar-wing shielding.

5.3.3.5.2 Common Methods of Thermal Control Generally, the thermal control methods can be divided into passive, semi-passive, and active methods.

Passive thermal control refers to the thermal control without moving parts and electric energy consumption. It has the advantages of simple technology, reliable operation, long life, and good economic performance. The elements and components used for passive thermal control include: thermal control coating, secondary surface mirror, multilayer thermal insulation materials (aluminized polyester film in most cases), heat pipe (its heat transfer is based on the phase change and circulating flow of the working medium), heat-conducting silicone grease, heat-conducting plate, phase-change materials (such as paraffin, used at temperature-jump peak), radiation refrigeration, heat radiator with special structure, heat insulation pad, and heat screen. Passive thermal control is used in more than 95% of the thermal control design of an unmanned spacecraft. Figure 5.3 shows an advanced passive thermal control device using a capillary suction pump and deployable heat radiator.

Semi-passive thermal control refers to the thermal control with moving parts but without electric energy consumption. In this method, a simple shutter driven by thermal sensitive device is generally used to open or close the heat conduction channel, so that the heat can escape or not escape.

Active thermal control refers to the thermal control that requires the consumption of electrical energy. For example, electric heaters, mechanical circulating pumps, Stirling cryocoolers, and pulse tube coolers are of this thermal control type. Figure 5.4 gives the schematic diagram of an active mechanical circulating pump system used in manned spacecraft.

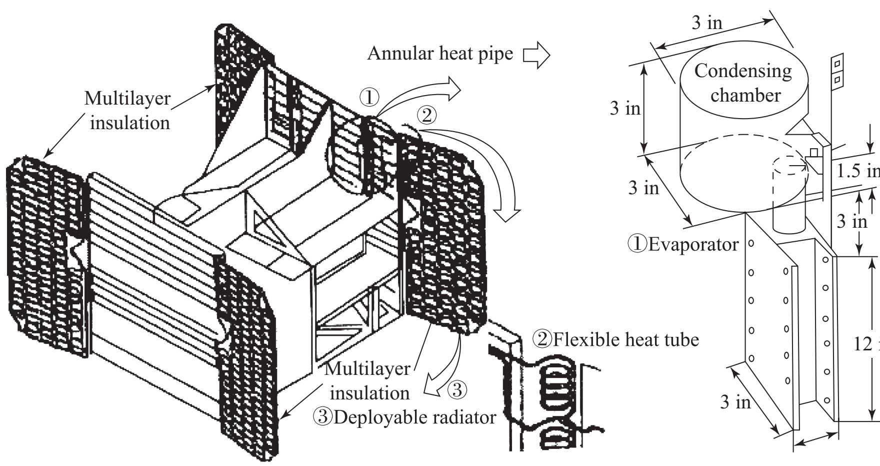


FIGURE 5.3 A deployable radiator.

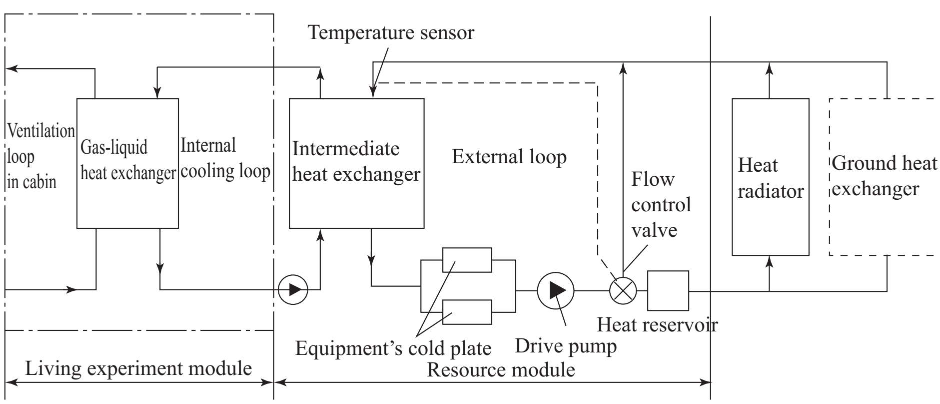


FIGURE 5.4 Schematic diagram of a mechanical circulating pump system.

5.4 ANALYSIS AND SUMMARIZATION OF GENERAL PERFORMANCE INDEXES

5.4.1 Main Contents of General Performance Indexes of a Spacecraft

Take a three-axis-stabilized communication satellite in GEO as an example. The main contents of the satellite's general performance parameters are listed in Table 5.6. It is not difficult to see from Table 5.6 the correlations between the listed contents and the satellite

TABLE 5.6 Main Contents of Performance Parameters of a Communication Satellite

No.	• Item	Main Contents of Performance Parameters
1.	• Satellite orbit	• Maintenance accuracy of geostationary orbit, fixed-point position, north-south position, and west-east position
2.	• Attitude stabilization	• Attitude stabilization mode, long-term offset ability, short-term offset ability, attitude control accuracy (pitch, roll, yaw)
3.	• Antenna pointing accuracy	• Pitch, roll, yaw (normal mode and position-holding period)
4.	• Satellite size	• Satellite body, (single) solar wing length, total satellite altitude with antennas folded, total satellite altitude with antennas deployed
5.	• Satellite mass	• Mass at satellite-rocket separation, and satellite dry mass
6.	• Power supply	• Solar wings (quantity, single-wing single-panel state, single panel size), solar-wing output power (beginning of life, end of life), batteries (type, quantity, capacity, maximum discharge depth), power supply bus voltage
7.	• Satellite life	• Design life, operating life, deorbit requirements
8.	• Reliability	• EOL reliability
9.	• Communication frequency band	• Uplink band, downlink band (single band, multi-band)
10.	• Communication coverage area	• Primary service area, secondary service area (whether a point beam is needed)
11.	• Number of communication transponders	Number of different frequency bands
12.	• Mode of receiver backup	• Cold backup, hot backup
13.	Mode of power amplifier backup	• Cold backup, hot backup
14.	• EIRP	• Each frequency band and its power
15.	• G/T of receiver system	• G/T for each frequency band
16.	• SFD	• Flux density (attenuator levels)
17.	• TT&C frequency	• Uplink frequency, downlink frequency (different frequency bands)
18.	• Receiver sensitivity	• Sensitivity and dynamic range
19.	• Uplink carrier modulation	• Modulation mode, modulation signal, subcarrier dot frequency number
20.	• Downlink carrier modulation	• Modulation mode, modulation signal, subcarrier dot frequency number
21.	• G/T of TT&C receiver system	• G/T (before and after the fixed point)
22.	• TT&C EIRP	• Power (before and after the fixed point)
23.	Telemetry capacity	• Analog quantity, digital quantity
24.	Telecontrol capacity	• Direct instruction, indirect instruction, bus instruction

engineering subsystems at the upper and lower levels. Among them, Items 1 and 5 are closely related to the carrying capacity of the launch vehicle. Item 4 is closely related to the fairing space of the launch vehicle. Items 7–16, 2, and 3 are directly related to the ground application system and are also the user performance indexes. Items 1–3 and 17–24 are directly related to the ground TT&C center. Items 4–8, 23, and 24 are directly related to each satellite subsystem. Items 1–3 are related to the attitude and orbit control system. Items 7, 8, 11–15, and 21–24 are directly related to Items 4–6. Of course, there are much more indirect correlations.

5.4.2 Initial Distribution of General Performance Indexes

Some of the general performance indexes are only related to one subsystem (such as its performance indexes), while some are related to each subsystem (such as size, accuracy, heat dissipation, telemetry parameters and telecommands, mass, power consumption, propellant, reliability, life, and other performance indexes). The former needs to be coordinated, analyzed, and distributed with large systems (such as the performance indexes of the payload and TT&C subsystem), while the latter needs to be coordinated, analyzed, and distributed with each subsystem of the spacecraft.

These performance indexes can be analyzed and calculated quantitatively. Some of them have to be analyzed and determined empirically. It is worth mentioning here that many indexes cannot be treated by simple summation, but should be treated correctly by analyzing their characteristics.

For example, when the dry weight of a spacecraft (especially GEO spacecraft) is increased by several folds, the propellant weight needs to be increased accordingly. In this way, the total weight of the spacecraft must be greater than its dry weight by several folds. For example, the total weight of DFH-3 satellite in GEO has been amplified by about 2.3 times. If the general designers are not mindful of this amplification relationship, they will make a mistake in coordinating the spacecraft weight with the weight of the launch vehicle or apogee engine.

Another example is that at the initial stage of overall spacecraft design, the power consumption of each subsystem instrument and equipment needs to be analyzed before the design of power supply. In this case, all the power cannot be added up mechanically, which will result in the design of very large primary power supply (generally as a combination of solar arrays and batteries). The correct approach is to classify various power-consuming devices into long-term loads, short-term loads, and pulse loads. The pulse loads (such as EED and solenoid valves) are characterized by high power and short power utilization time, so they have very small power consumption and can be directly powered by batteries. The short-term loads (such as Earth observation cameras) can be staggered according to the time program to minimize the peak demand. Then, the in-orbit power consumption program graph of the spacecraft can be obtained by adding the short-term loads and long-term loads that are staggered according to the time program. Finally, the primary power supply is designed according to the power consumption program graph. The primary power supply designed in this way is more reasonable.

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CHAPTER 6

Design of Spacecraft System Concept

Wang Rui, Yuan Zhi, Liu Jiuli, Zhou Liping, Xu Hao, and Li Xiaofei

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CONTENTS

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Based on the analysis of a spacecraft mission, the spacecraft system concept design can be carried out. The design of the spacecraft system concept is complex, involving the design of multiple disciplines such as mechanical, electrical and thermal disciplines and including the design of internal and external interfaces of the spacecraft system.

Based on the introduction of mission profile analysis, this chapter describes in detail the overall design contents of the system, puts forward the overall design requirements for the subsystems, presents the contents of flight program design and discusses the design and verification of internal and external system interfaces.

6.1 MISSION PROFILE ANALYSIS

The purpose of mission profile analysis is to identify all the events, environmental conditions and operating conditions experienced by a spacecraft during its mission, so as to provide input for the overall spacecraft design.

The items and contents of mission profile analysis are shown in Table 6.1.

The mission characteristics are obviously seen in the division of flight phases. The flight phases of Chang 'e-3 are shown in Figure 6.1.

The analysis results of a mission are generally expressed by the mission profile, which reflects the main stages, events, job description, environment and time of the flight mission. An example of mission profile is shown in Figure 6.2.

6.2 OVERALL SYSTEM DESIGN

The overall design of a spacecraft system is to complete the design of spacecraft system concept according to the user's requirements. It mainly includes orbit design, configuration and assembly design, power supply and distribution design, information flow design, overall attitude and orbit control design, overall thermal control design, overall electromagnetic compatibility (EMC) design, accessibility design, external system interface design and other related design contents. Among them, the orbit design, configuration and assembly design, external system interface design and other contents will be discussed in other chapters (Figure 6.3).

TABLE 6.1 Items of Mission Profile Analysis of a Spacecraft System

No.	Item	Contents
1.	Phase	According to the flight mission characteristics, the flight phases of a spacecraft are divided into launch phase, state establishment phase and other phases. Different spacecrafts have different mission characteristics. For example, a deep space exploration mission can include the Earth-Moon transfer phase, circumlunar phase, powered descent phase etc.
2.	Event	Describe the key events at each stage, such as solar wing deployment.
3.	Job description	Describe the key work that needs to be done in each phase.
4.	Environment	Describe the environments of the spacecraft in each phase, including force, heat, space etc.
5.	Time	The approximate duration of each phase

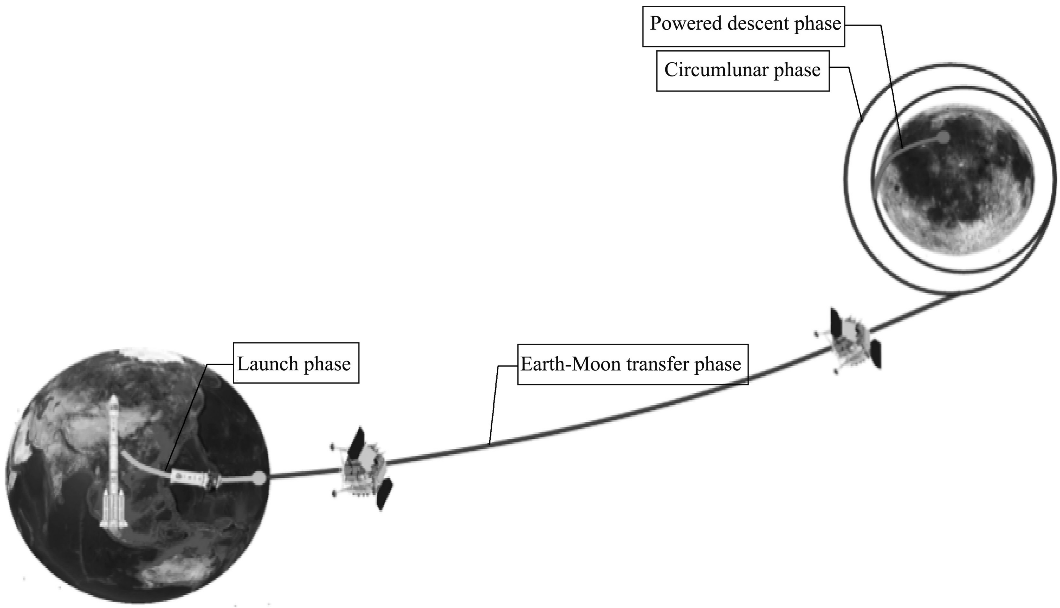

FIGURE 6.1 Schematic diagram of the flight process of Chang ‘e-3.

Phase	Launch phase	State establishment and in-orbit testing phase	In-orbit flight phase
Event	Launch vehicle firing; Fairing jettison; Ground capture of satellite; Satellite-rocket separation	Initial deflection elimination; Solar wing deployment and locking; Attitude disturbance elimination; Infrared introduction and sun sensor analog; Satellite establishment gyro + infrared attitude determination; SAR antenna unlocking and deployment; Elimination of antenna deployment attitude disturbance; Deployment and locking of data transmission antennas a/b; Momentum wheel activation for whole-satellite control; Orbital maneuver; Orbital adjustment; Right way-up position adjustment and star sensor introduction; Data transmission startup; Side-sway to right side view; Platform service system function and performance testing; Payload function and performance testing	Normal operation of all subsystems; Orbital adjustment;
Job description	Orbit insertion (ground capture of satellite); Orbit insertion (satelliterocket separation); Each subsystem works under the preset initial condition; The battery, secondary power supply, data management, TT&C, thermal control and other subsystems are in operation; The computer, Earth sensor and gyroscope combination in the attitude control subsystem are all started, and the thruster is warmed up.	Initial deflection elimination, satellite attitude capture, solar wing deployment and locking, attitude disturbance elimination; Satellite’s +Z axis pointing and tracking the Earth, and establishment of coarse-precision 3-axis stable attitude; Payload unlocking, deployment and locking; Normal satellite setting; Introduction of star sensor into the control loop, and establishment of high-precision Earth-pointing control mode for the satellite; Entry into frozen orbit by maneuver; Payload power-on and initial testing.	Payload in operation; Payload data downlink during transit; Orbit maintenance; Completion of follow-up tasks.
Environment	Noise, vibration, overload, heat, vacuum (after fairing jettison), impact (separation), low pressure discharge	Dynamic environment with EED unlocking and load rotation, and space environments such as solar electromagnetic radiation, Earth's neutral atmosphere, Earth's ionosphere, Earth's magnetic field, plasma, space charged-particle radiation, micrometeoroids and orbital debris, low gravity, vacuum, heat, pollution, etc.	
Time	About 20 min.	5-100 days	3-8 years

FIGURE 6.2 Mission profile of a remote sensing satellite in sun-synchronous orbit.

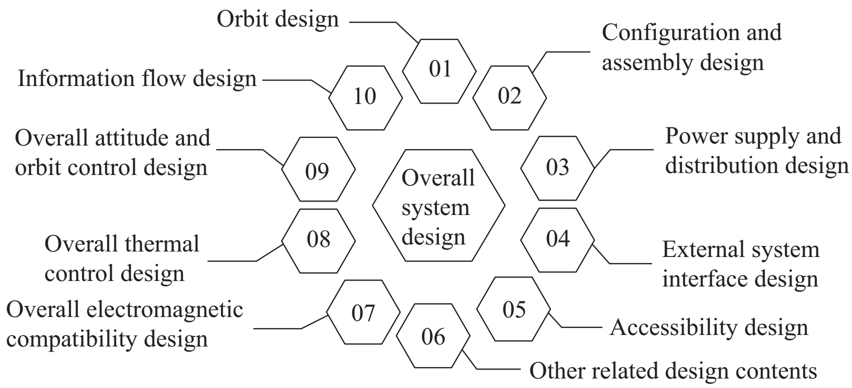

FIGURE 6.3 Contents of overall system design.

6.2.1 Overall Power Supply and Distribution Design

The overall energy flow design is to complete the analysis and distribution of spacecraft power demand and to formulate the basic requirements and specifications for power supply and distribution system and distribution path protection that need to be followed in the design and development of a spacecraft.

6.2.1.1 Design Principles of Spacecraft Energy Flow

The spacecraft energy flow should be designed according to the principles of reliability, rationality, advancement and economy, while meeting the normal operating requirements of spacecraft platform and payload.

Reliability: considering the harsh space environment conditions, irreparability, high-performance requirements, low production, high price and other characteristics of a spacecraft, the design process of energy flow must put reliability in the first place to ensure the reliability of the spacecraft power system and distribution system.

Rationality: the energy flow design must meet the overall design requirements of a spacecraft or its basic load requirements in the worst case and shall have a certain ability to adapt to failure.

Advancement: the energy flow design shall adopt mature and advanced Chinese/foreign technologies to the greatest extent and achieve optimization, so that the main performance indexes of the spacecraft power system and distribution system can reach the advanced level.

Economy: the energy flow design shall consider the standardization, choosing to inherit the mature technologies and achievements with in-orbit flight experience and the mature processes, materials and components so as to reduce the R&D and test expenses and improve the cost performance.

6.2.1.2 Design Ideas on Spacecraft Energy Flow

The spacecraft energy flow design includes power supply design, power distribution design, energy balance budget and distribution path protection design.

6.2.1.2.1 Power Supply Design The primary task of power supply design is to choose the optimal power regulation topology. Secondly, the busbar voltage and the technology of energy generation and storage should be selected in conjunction to optimize the overall structure of power supply subsystem. The main factors to be considered include the characteristics of payload power demand and its power level, the characteristics of spacecraft orbit, the in-orbit life of the spacecraft and so on.

6.2.1.2.2 Power Distribution Design The power distribution design is to distribute the energy in accordance with the power supply/distribution requirements and system requirements. It shall be consistent with the overall design of power system. The power distribution design shall meet the mission demand with the least mass, volume and cost and optimize the energy distribution design with a verified distribution strategy.

6.2.1.2.3 Energy Balance Budget The energy balance budget includes two aspects. First, the power output of solar arrays shall meet the requirements of load power consumption and battery charging. Second, the discharge depth of the batteries shall always be controlled within the range required to ensure the cycle life.

The energy balance budget of a spacecraft includes power demand analysis, power supply capability analysis, energy balance mode determination and other contents. The spacecraft power demand can be analyzed in accordance with the power demand budget. The power supply capacity analysis is mainly to analyze the solar array output capacity and battery power supply capacity.

6.2.1.2.4 Distribution Path Protection Design The distribution path protection is designed to prevent damage to spacecraft power system due to internal failures or load circuit failures. The following methods are often adopted to protect the spacecraft:

1. Fusing overcurrent protection

   Fusing overcurrent protection is an unrecoverable disconnect of busbar power path to achieve short-circuit protection. This circuit is simple in structure and easy to implement but is narrowly used because the fuse cannot be recovered after providing the protection and cannot be used for the power supply protection of key equipment and subsystems. In addition, the fusing time of millisecond level will lower the input voltage and affect the power supply to other equipment.

2. Recoverable overcurrent protection

   Compared with fusing overcurrent protection, this protection method has the advantages of fast response, recoverability and remote control. It can provide the subsystem-level power supply protection, mainly in the forms of inverse time protection and current-limiting protection.

6.2.2 Overall Information Flow Design

The spacecraft information flow design is to complete the analysis and distribution of spacecraft information flow demands under the constraints of ground TT&C and data transmission network. The telemetry/telecontrol channel code rate and payload downlink data rate are determined. The protocol of network communication and the telemetry/telecontrol interface circuit are developed.

6.2.2.1 Classification of Spacecraft Information Flows

The information flows in a spacecraft system are divided into control flow and data flow according to the contents of information flow. The control flow refers to the telecommands, uplink injected data, autonomous control information, safety control information and other information for maintaining the working condition of the satellite and manipulating the satellite to complete the payload tasks. It has a small data size but high requirements for the reliability and timeliness of data transmission. The data stream refers to the telemetry data, remote sensor data and auxiliary data generated by the onboard equipment and transmitted to the ground for processing. It has a relatively large data size and mainly focuses on the transmission efficiency and the requirements for data processing.

According to the configuration and basic operating mode of satellite platform and payload, the spacecraft information flows can be divided into the following four information flows (as shown in Figure 6.4 and 6.5):

1. **Uplink control flow:** the control instructions and injected data are generated autonomously from the ground or onboard to help equipment complete the switching control, mode setting and program loading. This type of information is directly related

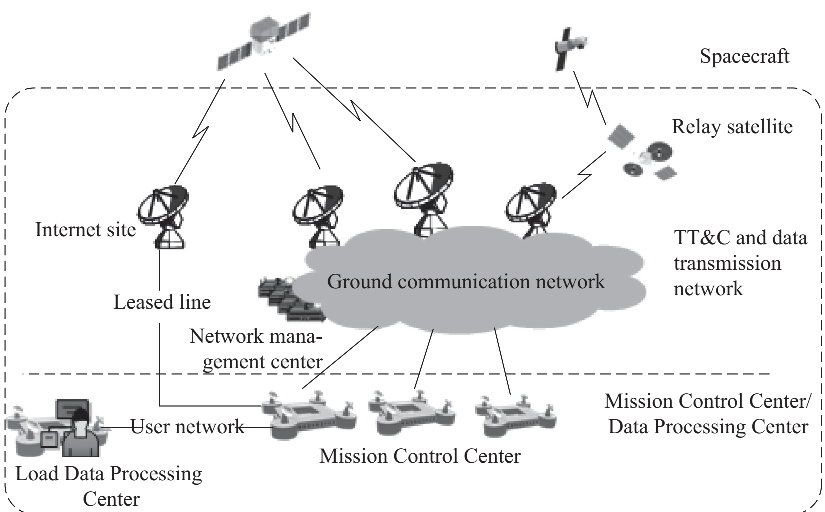

FIGURE 6.4 Schematic diagram of TT&C and data transmission network for spacecraft and ground.

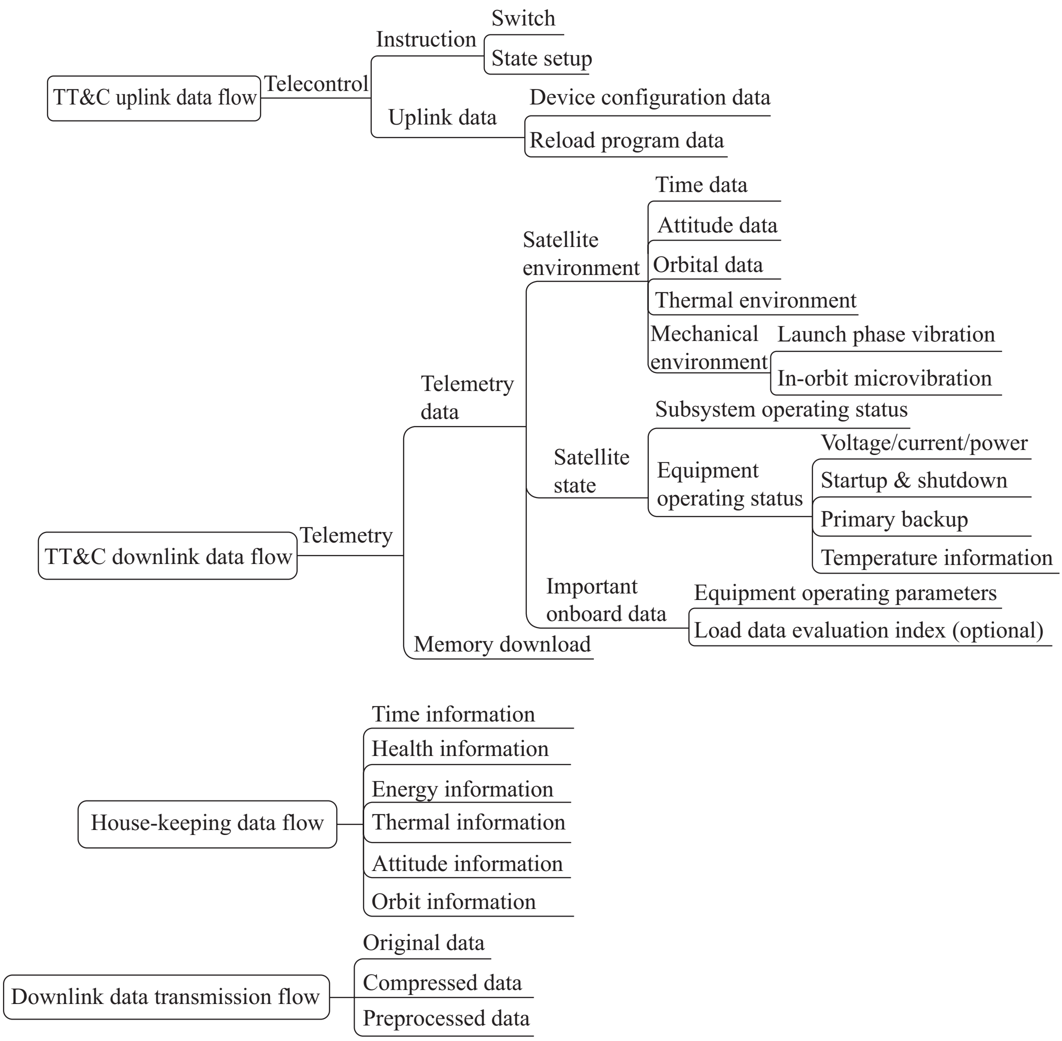

FIGURE 6.5 Diagram of spacecraft information flow.

to the satellite mission or even the safety of the whole satellite, requiring high reliability of the information channel and high accuracy of the data.

2. **House-keeping data flow:** the information source is onboard equipment. The information is routed through onboard transmission network and is used by other equipment as algorithm input and data reference. Such information includes the data on satellite time, orbit and attitude, as well as the important data for recovering the working condition of the equipment. It has high timeliness and requires the equipment to operate under a unified time reference.

3. **TT&C downlink data stream:** including real-time and delayed telemetry data, event data, computer memory readout data etc. This type of information is limited by

TT&C channel rate and TT&C arc. Therefore, its downlink scheme shall be optimized to ensure that the ground can obtain the largest amount of information for health status interpretation and fault troubleshooting.

4. **Downlink data transmission flow:** including image data and auxiliary data, as well as high-volume onboard data (such as mechanical sensor data and historical telemetry data) that cannot be downlinked through TT&C channels. This type of information has higher requirements for onboard storage performance and downlink channel efficiency.

6.2.2.2 Design Principles of Spacecraft Information Flow

The information flow design is a key part of the overall satellite design and should be carried out according to the principles of trade-off design, simplification design, reliability design, redundancy design, transmission margin design and safety design.

Trade-off design: to optimize the design of spacecraft system information flow under a series of constraints, by analyzing and trading off the effects and costs of various design measures (such as reliability and safety) at the system level and subjecting local optimization to global optimization.

Simplification design: to reduce the complexity of information flow relationship while meeting the overall spacecraft requirements. The simplification design can reduce the possibility of the existence of abnormal information flow channels or abnormal information (invalid information, error information, unexpected information etc.) and weaken their negative impact on the information flow.

Reliability design: to take measures to ensure that the information flow network can realize normal or degraded running in case of failure.

Redundancy design: to design appropriate redundancy for the trunks, branches and nodes of the information flow network.

Transmission margin design: a certain margin should be reserved for the information transmission by information flow channels to avoid local deadlock of the information flow network and local blockage of the information flow channels.

Safety design: to take measures to prevent the local damage to information flow network that may affect the safety of the spacecraft system. To ensure the security of information itself, important data shall be saved and processed. For example, important control information can be saved and important data can be modified through ground uplink injection. To ensure the security of information control instructions, the measures such as multi-level control and authentication are taken for critical or important control instructions to avoid accidental execution of the instructions. The security design of information flow shall also consider the information security. Important external information flow data shall be encrypted.

6.2.2.3 Analysis of Spacecraft Information Requirements

Statistics on the information types, information amount, flow channel requirements (the transmission rate and function requirements of each flow channel included), information processing function and performance and other information requirements of spacecraft system information flow are collected. The information of telemetry, telecontrol, bus communication, and the information transmitted through RS422, LVDS and TLK2711 time synchronization are contained in information types. The influence on the design of spacecraft information flow is analyzed and finally the code rate of telemetry/telecontrol information channels and the rate of payload downlink are determined.

6.2.2.4 TT&C Channel Design

6.2.2.4.1 Main Tasks of TT&C Channel The TT&C channel is mainly to complete the radio frequency (RF) transmission of TT&C uplink data stream and TT&C downlink data stream. It mainly has three tasks:

1. **Orbit tracking and measurement:** measure the orbital position of the spacecraft and track the spacecraft in the visible arc of ground TT&C network.
2. **Telemetry:** provide a downlink channel, and transmit the spacecraft's analog quantity, digital quantity, state quantity, temperature telemetry information, attitude, and orbit data as well as time code to the ground station for the ground personnel to analyze and judge the spacecraft state, understand the working condition of the spacecraft, and judge how to implement the spacecraft control.
3. **Telecontrol:** provide an uplink channel, receive and demodulate the uplink telecommands sent by the ground station, and send the demodulated video signal to the data management system of the spacecraft. Control the spacecraft according to the spacecraft's needs, and complete the attitude adjustment, orbit control, energy control, satellite status switching, troubleshooting and other work.

6.2.2.4.2 Classification and Characteristics of TT&C Channel The TT&C channels can be divided into independent carrier TT&C system, unified carrier TT&C system and space-based TT&C communication system.

Independent carrier TT&C system is composed of different independent devices, including the devices for orbit tracking and measurement, telemetry and telecontrol. In this system, the telemetry device has its own transmitter and antenna, the telecontrol device has its own receiver and antenna, and the tracking device has its own transponder and antenna. The system is characterized by decentralized equipment and functions, complex spaceborne system and low reliability. With the rapid development of spaceflight, the TT&C system is required to have multi-functions, long operating distance and good EMC. However, independent carrier TT&C system cannot meet these requirements, so the unified carrier system arises at the right moment.

The birth of a unified carrier TT&C system is a leap in TT&C technology. This system is a combination of range and velocity measurement, telemetry and telecontrol devices. In one carrier, several sub-carriers (for telemetry, telecontrol and ranging) are modulated to realize the multi-channel signal transmission based on frequency division multiplexing, so as to integrate multiple functions into the TT&C system. In other words, multiple TT&C

functions are unified into one carrier. The characteristics of this system are multi-function integration, centralized equipment and functions, long operating distance, simplified equipment, reduced volume, frequency resources saving, good EMC, TT&C standards internationalization, medium accuracy for velocity measurement and orbit determination and narrow TT&C coverage. The unified carrier TT&C system is called unified ultrashort wave (USW) TT&C system when using the USW carrier, or called unified S-band (USB) TT&C system when using the S-band carrier, or called unified C-band (UCB) TT&C system when using the C-band carrier. In this system, both the TT&C station and the spacecraft equipment use one carrier, one common channel and one common baseband to achieve the TT&C.

Because the unified carrier system of "frequency division system" has the problems of multi-carrier combination interference, nonlinearity of single angular-modulated wave time channel, multi-carrier intermodulation interference and data rate not too high, the "time division system" has emerged. The "time division system" is to transmit digital signals. In the system, the signals of telecontrol, telemetry, voice and video can be unified by time division multiplexing and then transmitted by one carrier, so the system has a low linear requirement and high transmission rate. In the spread-spectrum TT&C, the data is processed through pseudo-random code modulation, and is transmitted after spectrum spreading; on the other hand, the receiving end uses the same code for demodulation and related processing and recovers the original data. This system is called "spread-spectrum unified TT&C system". It has the advantages of anti-jamming, anti-multi-path fading, low transmission power spectrum density and good CDMA (code division multiple access) abilities. Owing to these advantages, the "spread-spectrum unified TT&C system" has good anti-jamming ability and low interception probability so that it can't be detected easily and can improve the utilization of the limited frequency resources.

Space-based TT&C communication system is the combination of tracking and data relay satellite system (TDRSS) and telecontrol & telemetry system and is characterized by high TT&C coverage. With the increase of low and medium orbit spacecrafts, the TT&C tasks of ground stations are becoming more and more arduous, so there is a crying need for autonomous TT&C service. Moreover, some spacecrafts also need real-time arc tracking and TT&C. However, due to the limitation of the Earth curvature and linear microwave propagation, a large number of ground stations are required to be arranged in the world. It is not only uneconomical but also impractical to install foreign and seaborne stations. The design idea of "space-based" system is just to solve the above problems. Figuratively speaking, "space-based" means moving a TT&C station into the sky. The 36,000 km-high TDRSS enables global uninterrupted real-time tracking and communication.

6.2.2.4.3 Requirement for Telecontrol and Telemetry Code Rate in TT&C Channel According to the requirements of the spacecraft mission, the telecontrol code rate, telemetry code rate and bit error rate (BER) can be determined. In general, the telecontrol bit error rate is required to be better than 1×10^{-6}, and the telemetry bit error rate is required to be better than 1×10^{-5}.

6.2.2.5 Design of Data Transmission Channel

6.2.2.5.1 Main Tasks of Data Transmission Channel The data transmission channel is mainly to transmit the image data and auxiliary data as well as the high-volume onboard data that cannot be downlinked through the TT&C channel.

6.2.2.5.2 Classification and Characteristics of Data Transmission Channels The data transmission channels are divided into RF channel system and laser communication system.

The RF channel system is characterized by high technological maturity and good reliability. The first thing for this system is to select the working frequency band – generally X-band or Ka-band – for data transmission to the Earth. With the increase of the information rate required for transmission, the bandwidth of X-band can no longer meet the demand of satellite data downlink. More and more satellites use Ka-band for data transmission to the Earth. To select a modulation mode, the modulation rates and losses of different modulation/demodulation modes (at present mainly including QPSK, 8PSK, 16QAM, 32QAM and so on) shall be compared to choose an appropriate modulation mode. The gains of different error-correcting coding methods shall be studied. At the given bit error rate, the energy of the signal to be sent shall be reduced to realize reliable communication under the limited power. At present, the commonly used encoding methods include RS code, cascaded code of RS+ convolutional code, Turbo code and Low-density Parity-check (LDPC) code. In terms of antenna, an appropriate antenna system shall be chosen. At present, there are two main types of antenna: Earth-matched beam antenna and high gain point beam antenna. The Earth-matched beam antenna is mainly used in the transmission with a relatively low rate, while the point beam antenna is used in the transmission with a high data rate.

Due to the limitation of transmission capacity, traditional RF communication is subjected to the “bottleneck” of reaching the communication rate above Gbps and is difficult to meet the requirements of future high-speed broadband communication. However, intersatellite optical communication can effectively solve this “bottleneck”. Compared with RF communication, intersatellite optical communication has strong advantages, including high transmission data rate, small size, light weight, low power consumption, and strong anti-interference and anti-interception ability. The selection of laser wavelength for laser channel system requires the consideration of background radiation effect, beam divergence angle, antenna size, feasibility of laser and modulation technology etc. To select the signal system and modulation mode, the modulation modes, detection sensitivities and information-carrying capacities of different signal systems should be compared. Currently, there are mainly two modulation modes in space laser communication: intensity modulation/direct detection (IM/DD) system and multi-modulation/coherent detection system. The coding gains of different error-correcting coding methods shall be studied. At the given bit error rate and symbol error rate, the energy of the signal to be sent shall be reduced to realize reliable intersatellite laser communication under the limited power.

6.2.2.5.3 Requirement for Transmission Rate in Data Transmission Channel According to the requirements of the spacecraft mission, the storage capacity, transmission rate and bit error rate can be determined.

6.2.2.6 Spacecraft Information Flow Design

The information flow design for a spacecraft system mainly includes information flow network design and information flow processing design. In most of the cases, the two tasks need to be iterated in order to obtain the optimal or reasonable information flow design.

6.2.2.6.1 Information Flow Network Design Information flow network design is to design the relations between information flow channels and nodes in the information flow network to ensure that the information flow network architecture meets the information demand and the overall spacecraft requirements. It mainly includes hierarchical design, backbone structure design and information interface design. Different information flow network solutions should be analyzed and compared, and a feasible information flow network concept suitable for the specific model is obtained.

1. Hierarchical design of information flow network

   The hierarchical design of information flow network is mainly to determine the hierarchy of the information flow network structure. For example, the information flow network is determined to be a three-level structure composed of a backbone network, a lower-level network and a terminal node.

   The information flow networks can be divided into bus type, star type, ring type, tree type and mixed type according to their topology structure. The spacecraft information flow network is generally a mixed topological structure.

   According to the way of information processing, the information flow networks can be divided into decentralized structure, centralized structure and mixed structure. In order to reduce the complexity of information flow relationship, the information flow network of a large spacecraft is generally a decentralized structure with a “system-subsystem-equipment-component” hierarchical processing mode. In the information flow network with a decentralized structure, the host node of information processing is generally a computer dealing with the system-level tasks, and the slave node is generally an intelligent processing unit for a single function. For a medium-scale spacecraft, an appropriate structure can be chosen through comprehensive trade-off according to the complexity of system information processing. For a small spacecraft with an urgent need for centralized information processing, the information flow network is usually a centralized structure.

2. Design of information flow network backbone

   The backbone structure of information flow network is generally a bus, such as CAN bus, 1553B bus or other serial data bus. Data bus is the information exchange center of information flow network, through which the distributed control of a spacecraft and the transmission of instructions, telemetry data, broadcast data and other information are realized. The core of information flow network is the bus of data management (house-keeping) subsystem.

3. Information flow interface design

   The goal of information interface design is to ensure that both sides of an information interface can transmit information reliably and securely in line with the expected designed logic. According to the signal characteristics, the information flows in a spacecraft system are divided into two categories: low-frequency information flows and RF information flows. The low-frequency information flows mainly include telemetry information flow, telecontrol information flow, and attitude and orbit control information flow. The RF information flows mainly include TT&C information flow, data transmission information flow, intersatellite link information flow etc.

4. Analysis of information flow network traffic

   In the design of an information flow network, the constant value, peak data flow and margin of the information flow network are generally calculated from bottom to top and step by step. For the information categories not included in the calculation, their possible effects on the constant value, peak data flow and margin should be given.

Based on the analysis results of information flow network traffic, the designers shall evaluate whether the network is locally or globally congested, and improve the information flow network design accordingly.

6.2.2.6.2 Information Flow Processing Design Information flow processing design is to design the information flow processing scheme at the information flow nodes to ensure that the nodes will not affect normal network operation or to use nodes to provide management and maintenance for the operation of the information flow network.

The information processing mainly includes information flow protocol design, timing design, information flow encryption, autonomous information flow management, in-orbit information flow network maintenance, comprehensive information application etc.

Information flow protocol design: the application layer protocol of information flow should be designed according to the specific situation of the model mission, and the protocols of data link layer, network layer and transmission layer should be designed or implemented according to relevant standards. For example, the communication protocol of CAN bus should conform to the three-layer protocol structure and format specified in special technical documents. The communication protocol of 1553B bus should comply with the technical requirements, bus information flow requirement and electrical and functional formats specified in GJB 289A.

Timing design: the timing relation exists between adjacent nodes with information flow relation in the information flow channel. In general, the timing design of information flow should be carried out in the same protocol layer used by both communication parties.

Information flow encryption: to ensure the safety of a spacecraft and prevent the malicious damage caused by illegal users, the corresponding encryption measures, i.e., information flow encryption design, should be taken for external communication links (such as TT&C, intersatellite links etc.).

Autonomous management of information flow: as the main node of information flow, the data management (house-keeping) computer runs software to complete some autonomous information flow management functions, such as fault diagnosis, fault isolation and reconstruction.

In-orbit maintenance of information flow network: the key nodes of information flow network (such as data management computer) generally can realize in-orbit maintenance through ground instructions to repair the software defects or extend the software functions so as to maintain the normal operation of spacecraft information flow network.

Integrated information application: the sharing of information in the spacecraft information flow network, namely, integrated information application, can improve the function, performance, reliability and security of the spacecraft system.

6.2.3 General Design of Attitude and Orbit Control

6.2.3.1 General

The attitude and orbit control is mainly to control the spacecraft attitude and orbit so as to achieve the control of spacecraft orientation and stability. The subsystems include sensor subsystem, controller subsystem and actuator subsystem.

The sensor subsystem is mainly to complete the satellite attitude measurement; the controller subsystem is mainly to complete the ephemeris calculation, attitude determination, control law selection and subsystem power-on/off control etc.; the actuators subsystem is mainly to provide the satellite attitude control torque, momentum wheel–unloading torque and orbit maintenance/change thrust. In addition, as a relatively independent system, the propulsion part of the actuator mechanism can perform the velocity damping and emergency control after satellite-rocket separation, the main functions of orbit transfer, as well as the momentum wheel–reserved momentum unloading and orbit maintenance when necessary (Figure 6.6).

6.2.3.2 Classification of Attitude and Orbit Control Methods

6.2.3.2.1 Attitude Control Type and Method Attitude control types are mainly divided into passive control and active control, including gravity gradient stabilization, spin stabilization and three-axis stabilization.

6.2.3.2.1.1 Gravity Gradient Stabilization The operating principle of gravity gradient stabilization is that the longitudinal axis of a slender body in a gravitational field always points toward the center of the Earth. Because the value of the gravity gradient moment

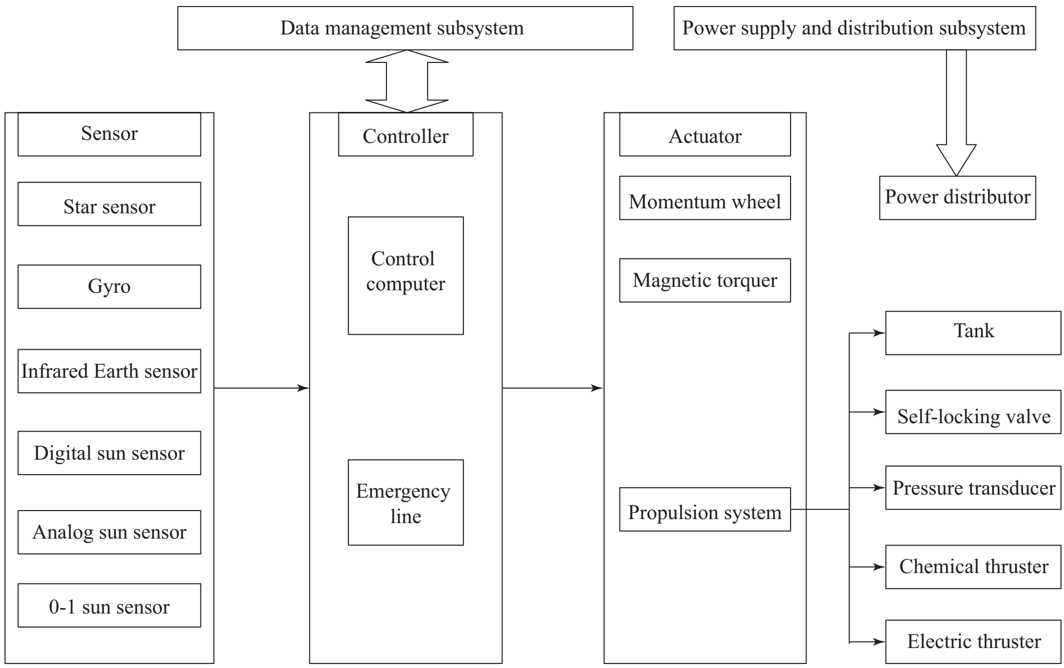

FIGURE 6.6 Block diagram of spacecraft attitude and orbit control structure.

is inversely proportional to the cube of the spacecraft's distance to the center of the Earth, the gravity gradient stabilization method is usually only applicable to low Earth orbit. The gravity gradient stabilization on a spacecraft is generally realized by installing an extensible pole (withdrawn during launch and extended after orbit insertion), which can keep the minimum moment of inertia of the spacecraft in the vertical direction of the Earth by using the moment generated by gravity gradient. This stabilization method is relatively simple, but with a low attitude control accuracy of about 1°. It applies to a low-accuracy spacecraft pointing to the Earth.

6.2.3.2.1.2 Spin Stability The operating principle of spin stabilization is that, as the entire spacecraft rotates, its angular momentum vector is almost constant in the inertial space. A more practical method is dual-spin stabilization, that is, the spacecraft is composed of two rotating parts, whose gyroscopic inertia in the inertial space is used to maintain the spacecraft stability. In this scheme, the Earth infrared sensor and the sun sensor are used to determine the attitude. The attitude is adjusted with an axial thruster, which is installed far away from the spin axis and produces a thrust parallel to the spin axis. The spin velocity is adjusted with a tangential thruster, which is installed far away from the spin axis and produces a thrust perpendicular to the spin axis and the horizontal line connecting the thruster to the spin axis. When the spin axis is perpendicular to the orbital plane, the orbit can be adjusted with a radial thruster, whose installation shall ensure that the thrust will pass through the center of mass. When the spin axis is in orbit and parallel to the flight direction, the axial thruster can be used for continuous operation. The attitude control accuracy is medium, up to 0.1°. This stabilization method is generally used for a spacecraft that is perpendicular to the orbital plane and points to the Earth, such as a spinning spacecraft with ground-oriented despun antenna.

6.2.3.2.1.3 Three-Axis Stabilization Three-axis stabilization is to keep the three axes of the in-orbit spacecraft pointing to a certain direction by using various actuators. In this scheme, the Earth infrared sensor, sun sensor, star sensor and various gyroscopes (and magnetometer sometimes) are used to measure the attitude of the spacecraft. The actuators (small thrust engines, momentum wheels, flywheels, magnetic torquers, torque gyros etc.) are used to adjust the attitude and orbit. When a thruster is used to adjust the attitude, it shall be installed far away from the center of mass, so that a larger torque can be generated. When adjusting the orbit or performing a larger orbital maneuver, a thruster with a greater thrust shall be used and the thrust shall pass through the center of mass. A small orbital maneuver can be realized by a thruster with a smaller thrust, which also needs to pass through the center of mass. The attitude control accuracy of the three-axis stabilization scheme is high, much better than 0.1°. Most of the modern spacecrafts pursue three-axis stabilization.

6.2.3.2.2 Type of Orbit Control The spacecraft orbit control is generally realized by one of the following four systems: cold gas propulsion system, chemical propulsion system, electric propulsion system and hybrid propulsion system.

6.2.3.2.2.1 Cold Gas Propulsion System The cold gas propulsion systems use the high-pressure inert gas stored at room temperature as propellant. It is a simple system with low specific impulse. With the development of chemical propulsion system, the use of cold gas propulsion system has been gradually reduced.

6.2.3.2.2.2 Chemical Propulsion System The existing chemical propulsion systems for spacecrafts mainly include monopropellant propulsion systems, bipropellant propulsion systems and dual-mode propulsion systems. Most of the monopropellant propulsion systems use anhydrous hydrazine as propellant, have low specific impulse and high reliability and find their application in low-orbit remote sensing satellites. Most of the bipropellant propulsion systems use nitrogen tetroxide and methylhydrazine as propellants, have high specific impulse and achieve wide use in large spacecrafts and HEO spacecrafts. Nitrogen tetroxide and hydrazine are used as propellants in the dual-mode propulsion systems, with bipropellant used at apogee and monopropellant hydrazine used for attitude control. Thus the advantages of monopropellant, namely high reliability and low thrust, have been combined with the advantages of bipropellant, namely high specific impulse.

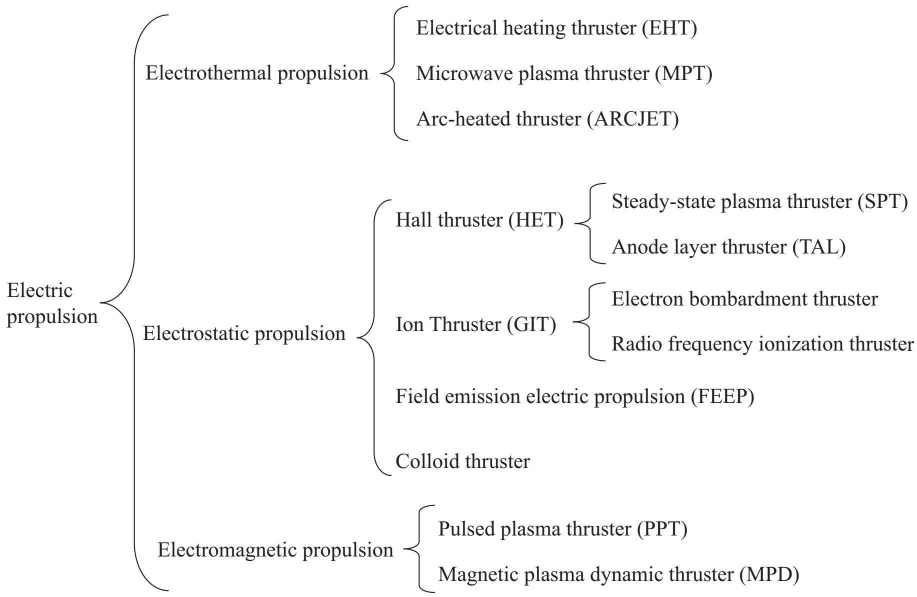

FIGURE 6.7 Categories of electric propulsion systems.

6.2.3.2.2.3 Electric Propulsion System With the advantages such as high specific impulse, long life, repeatable start, small thrust and high control accuracy, the electric propulsion system plays a significant role in high-precision orbit maintenance and spacecraft weight reduction and is being used more and more widely. As an advanced space propulsion technology, electric propulsion platform has been widely used on the spacecrafts of the United States, Russia, Europe, Japan and other countries and has achieved remarkable technical and economic benefits.

According to the working principle of electric propulsion, electric propulsion methods are traditionally divided into three categories: electrothermal propulsion, electrostatic propulsion and electromagnetic propulsion, each of which includes several different types, as shown in Figure 6.7. Among them, electrothermal propulsion means that the propellant energy is increased by electrothermal method and the thrust is generated by accelerating the nozzle. Electrostatic propulsion is to generate a thrust by using electrostatic field to accelerate the propellant. Electromagnetic propulsion is to generate a thrust by using electromagnetic field to accelerate the propellant.

Among the various types of electric propulsion systems, electrothermal propulsion system is a simple system with a low dry weight and is easy to integrate with the monopropellant chemical propulsion system. It has been widely used in the United States since the 1990s. However, its specific impulse is relatively low – about 600s. At present, pulsed-plasma electric propulsion, one type of electromagnetic propulsion, is only used for the attitude control of small spacecrafts. The magnetic-plasma electric propulsion technology with a large thrust ratio is still in the research and development stage and has a considerable gap to its mature application. Electrostatic propulsion is a type of electric propulsion with strong performance advantages, good technical maturity and wide application. Ionic electric propulsion and Hall electric propulsion are the two main electrostatic propulsion products (Figure 6.7).

6.2.3.2.2.4 Hybrid Propulsion System The chemical propulsion system has a large thrust range but a relatively low specific impulse, while the electric propulsion system has a high specific impulse but a small thrust. In order to give full play to the advantages of chemical propulsion system and electric propulsion system, a hybrid satellite-specific propulsion system has been developed gradually. The satellite propulsion system that integrates multiple propulsion modes and uses different propulsion modes at different working stages is called hybrid propulsion system.

6.2.3.3 Design Constraints

User requirements (such as coverage requirement) are the first factor to be considered in determining functional requirements. The System Engineering needs to combine user requirements with load capacity in order to determine the requirements of control subsystem. The main user requirements are discussed in the following subsections.

6.2.3.3.1 Requirement for Payload Orientation According to the requirements of a spacecraft mission, some payloads need to be oriented in the inertial space. For example, for the payloads in the Earth observation satellites, astronomical satellites and solar telescopes, their optical systems shall be oriented to the ground, the sky or the sun. The control of payload orientation is generally achieved in three ways: using the control system to control the spacecraft orientation, swinging of the swing mechanism inside the payload and joint maneuver of the satellite and the payload.

6.2.3.3.2 Requirement for Mission Agility With the continuous improvement of the user's requirements for spacecraft usability, the requirement for spacecraft agility has also been continuously raised. During the in-orbit mission, several attitude maneuvers and even orbital transfers will be carried out in order to complete different tasks. The specific requirements for attitude maneuvering include the followings.

6.2.3.3.2.1 Range of Maneuver For three-axis attitude maneuver, the range requirements can be put forward for each axis separately or for all the three axes. For example, the range of rolling-axis attitude maneuver is −30° to +30°, and the range of pitching-axis attitude maneuver is −50° to +10°.

6.2.3.3.2.2 Agility Requirement Since the relevant tasks (such as SAR payload imaging) will be performed after fast maneuver, a certain pointing accuracy and stability, such as 25°/30 s (including stabilization time), need to be achieved after maneuver in order to meet the mission requirements.

6.2.3.3.2.3 Mode Requirement The payloads in different working modes (such as imaging) sometimes need to meet the detailed attitude maneuvering requirements at each key step, such as active push-broom imaging mode and the establishment of maneuvering state within 20 seconds.

6.2.3.3.2.4 Maneuver Restrictions Considering the TT&C, safety and other factors, some maneuver indicators will be limited. For example, in the maneuver process, the maximum angular velocity must not be less than 4.5°/s, and the maximum angular acceleration must not be less than $2°/s^2$.

6.2.3.3.3 Orientation Constraints

6.2.3.3.3.1 Orientation Constraints During Orbital Maneuver The orbital maneuver of some spacecrafts requires the engine orientation, which can be determined according to the strategy of orbital transfer.

6.2.3.3.3.2 Constraint on Solar Wing Orientation The normal solar wing is required to point to the sun to ensure the spacecraft energy supply. When the solar wing rotation cannot meet the demand of energy supply, both the spacecraft attitude adjustment and the solar wing rotation are needed to ensure that the normal of a solar wing points to the sun.

6.2.3.3.4 Mission Constraints on Propulsion System In recent years, with the gradual improvement of the user's requirements for sub-satellite point control accuracy, the deviation of sub-satellite point from the trajectory should not exceed ±2 km. In order to guarantee the accuracy, the jet time for sub-satellite point trajectory control is generally not less than 10 seconds. If the selected thruster is too large, the time for each sub-satellite point trajectory control will be too short, the accuracy cannot be guaranteed and even the accuracy of sub-satellite point trajectory control will be out of tolerance. The orbit control thrusters (such as 1 N, 5 N, 20 N etc.) can be selected according to the need. The ion, Hall and arc thrusts in electric propulsion are in the order of tens to hundreds of millinewtons. Compared with the former, the sub-satellite point control accuracy of electric propulsion is greatly improved. However, to achieve the sub-satellite point control accuracy in the order of 100 m and even higher accuracy, a chemical thruster with smaller thrust (such as 0.5 N thruster), a combination of chemical and electric propulsion or an all-electric propulsion can be adopted.

6.2.4 Overall Thermal Control Design

6.2.4.1 Analysis of Overall Thermal Control Task

The task of spacecraft thermal control is to control the heat exchange between the inside and outside of the spacecraft and maintain the balance of the energy into and out of the spacecraft through reasonable thermal design during the spacecraft flight. In other words, the heat generated by the spaceborne equipment and the energy obtained by spacecraft from the outer space environment shall be balanced against the energy emitted by spacecraft to the outer space. Efforts should be made to ensure that the temperature of all onboard instruments and equipment and of the spacecraft structure itself is in the required range, so as to ensure the normal operation of the spacecraft in orbit.

The thermal control system needs to provide a good thermal environment for the onboard instruments and equipment during the launch of the spacecraft and its entire in-orbit life to ensure the reliable performance of the instruments and equipment. The thermal control design of the whole satellite should be unified by comprehensively considering the orbital conditions and configuration layout of the spacecraft, in combination with the required heat consumption, heat capacity, temperature and other parameters of each equipment.

The overall thermal control design covers all the structures and equipment on the spacecraft, that is, all equipment listed in the spacecraft product package. Usually, the temperature monitoring point of the equipment is used as the duty interface between the thermal control subsystem and the equipment. Generally, the temperature monitoring point is located on the outer wall of an instrument near its mounting plate. Special monitoring points are defined in the interface data sheet (IDS) table of the equipment or are arranged by the designer of thermal control subsystem according to the need and then specified in the thermal control document.

6.2.4.1.1 Analysis of Orbital Thermal Environment For a spacecraft in orbit, the factors with the greatest influence on its thermal environment are high vacuum, low universe temperature, solar radiation, Earth radiation and reflection. Without any thermal control measures, the components and equipment on a spacecraft can be exposed to temperatures ranging from less than −100°C to more than 100°C.

According to the spacecraft's input conditions (such as orbit parameters, flight history and flight attitude), the orbital thermal environment is analyzed comprehensively. The typical external heat flow conditions in each stage of the spacecraft flight history are sorted out to give the quantified results, which can serve as the basis for the subsequent thermal design and analysis.

6.2.4.1.2 Task of Thermal Control Design The task of thermal control design is to analyze the thermal environment necessary for the equipment and products in each subsystem to satisfy the above indexes according to the spacecraft's overall functional and performance requirements and then to put forward the corresponding temperature index requirements for the thermal control subsystem. Meanwhile, the constraints necessary for thermal control subsystem are put forward according to the weight, power, reliability and other information resources that the System Engineering can provide.

1. Demonstrate and analyze the thermal control requirements of a stand-alone device and put forward the index requirements for thermal control subsystem on the basis of analysis results.

   The determination of equipment temperature indexes should not only ensure the normal function and performance of the equipment but also consider the resource affordability of System Engineering and the feasibility of thermal control, so as to avoid insufficient design margin and over-design. The general temperature indexes of a stand-alone device are as follows:

   Temperature level: to maintain normal working condition, any equipment must meet a certain ambient temperature requirement. The temperature of general electronic equipment is kept at −15°C to +50°C. Some devices have special requirements. For example, the temperature range for batteries (usually Ni-CD, Ni-H or lithium batteries) is narrow, possibly from −15°C to +30°C. The specific temperature level may vary with the requirement of the spacecraft model.

   Temperature uniformity: for some special equipment, the thermal deformation caused by temperature difference may affect the equipment's function or performance. Therefore, to ensure normal equipment operation, the temperature difference of all or local equipment positions should not exceed a specific value. This is the temperature uniformity requirement. For example, for an optical camera, the excessive thermal deformation of its optical structure will affect its imaging quality. The thermal deformation must be controlled within a certain range, so the requirement of temperature uniformity is proposed.

   Temperature stability: some special equipment must meet not only the high requirements for temperature level and temperature uniformity but also the strict requirement for temperature stability (for example, the temperature level drift in a certain period).

   For example, an optical camera can achieve the best imaging quality by adjusting the focal plane position after obtaining a temperature field with the required temperature level and uniformity at the initial stage of orbit insertion. However, with the increase of in-orbit time, the temperature level may drift within the qualified range and may cause the change of the optimal focal plane position. To make it worse, it is impossible to focus the camera frequently in orbit. To avoid this phenomenon, the camera requires temperature stability.

2. Put forward the requirements for equipment-level thermal design and supervise their implementation.

   Usually, the temperature monitoring point of the equipment is used as the duty interface between the thermal control subsystem and the equipment. Generally, the temperature monitoring point is located on the outer wall of an instrument near its mounting plate. The thermal control subsystem is responsible for controlling the temperature of the monitoring point within the range specified by the IDS, while the equipment manufacturer is responsible for the internal thermal design of the equipment.

6.2.4.2 Design Principles of Spacecraft-Level Thermal Control

1. Inherit the proven design condition of a mature model as far as possible, and strictly follow the specifications and standards on the development of the satellite and its thermal control subsystem to ensure the high quality and reliability of thermal control subsystem.
2. Use the flight-proven thermal control materials, components and technologies as far as possible. Choose domestic components and raw materials to reduce costs while ensuring reliability, safety and effectiveness.
3. Focus on passive thermal design, achieve the best system performance through the design optimization, and minimize the active heating power.
4. Design focus: power supply, attitude/orbit control and propulsion, load.

6.2.4.3 General Idea on Thermal Control Design

1. The heat dissipation surface should be preferentially set in the direction where the arriving heat flow is small and stable. The heat dissipation capacity of outer cabin surface will increase with the decrease of the arriving heat flow. As the external heat flow becomes more stable, the temperature difference between high- and low-temperature conditions and the heating power to be compensated will decrease.
2. For a special device whose temperature index is stricter than most of the devices, its heat dissipation approach can be designed separately and can be insulated from other structures and equipment in the spacecraft.
3. For the equipment requiring high-temperature uniformity and stability, passive thermal control measures are generally used to control the temperature below the lower limits of the indexes and then active heating measures are taken to achieve stable temperature control.

6.2.4.4 Conceptual Design

Preliminary design of heat dissipation surface: according to the layout of all the onboard equipment and the distribution characteristics of external heat flow, the preliminary layout of heat dissipation surface is selected. According to the sunlight condition and life requirement, white paint or OSR (Optical Solar Reflector) coating is generally used as the condition of heat dissipation surface. The average periodic internal heat flow is calculated for every module. The area of each surface is preliminarily determined according to the heat dissipation capacity of each surface, and the exact area is determined and modified during the detailed analysis.

Determination of typical working condition: the determination of external heat flow of a typical radiating surface must consider the maximum and minimum solar incident intensities as well as the maximum and minimum sunlight angles (a smaller angle means a longer shadow period) and determine the maximum and minimum internal heat flows and the time of their occurrence. Through the combination of internal and external heat sources, the typical orbit model can be established as the extreme condition for design and analysis.

Design of active temperature control loop: according to the above design scheme and the requirements of equipment temperature, the configuration of active temperature control loop is preliminarily designed. For the equipment whose temperature may be lower than the lower limit of the index under low-temperature conditions, the heating measures are taken for temperature control and the heating power and temperature control threshold are initially set.

Establishment of thermophysical model: according to the configuration, structure & material and equipment layout of the spacecraft and the preliminary thermal control concept determined in the above steps, a thermophysical model is established for the spacecraft. The thermophysical model should be able to accurately reflect the conditions of heat exchange between the inside and outside of the spacecraft, including heat source, heat conduction and heat radiation process.

Thermal analysis and design iteration: by running the above thermophysical model, the temperature results of the spacecraft under typical working conditions are obtained. According to the temperature results, the layout of the radiating surfaces and the configuration of the heating loop can be adjusted continuously. If necessary, the installation positions of some equipment can also be adjusted through the coordination with the System Configuration. To obtain more ideal temperature results, the resource consumption of thermal control subsystem is reduced to finally obtain the optimal thermal control concept.

6.2.5 Overall EMC Design

6.2.5.1 EMC Design Requirements

EMC has been interpreted as a quantitative comparison between some unintentional electromagnetic interference (EMI) emission and the sensitivity of the circuits to which such interference is coupled. EMC of the system can be confirmed if the sensitivity threshold of a disturbed device in the system is higher than the interference emission level coupled to the device. Therefore, EMC safety margin is the difference between them. Obviously, the sensitivity threshold is higher than the interference emission level, and the margin represents a positive dB value; and vice versa. Therefore, EMC safety margin is considered to be a safety margin that allows for many uncertainties including aging.

The improvement of electronic system integration and the increase of high-frequency devices have become the development trend of modern spacecrafts. The main work of the spacecraft EMC is to ensure that all the RF transceivers on the spacecraft can work normally in the common electromagnetic environment. The prediction of RF EMC safety margin has become an important part of the early and medium-term development of microwave remote sensing satellite, communication satellite and navigation satellite. The electromagnetic interference safety margin (EMISM) is defined in the standard as the difference between the sensitivity threshold at a critical testing point and the EMI level coupled to the test point. The EMISM can be confirmed as the ultimate goal of the overall spacecraft EMC design, and all the control work of spacecraft EMC will be carried out with EMISM as the core.

So far, in the domestic and foreign standards for spacecraft EMC systems, the EMC safety margin is usually required to be 9 dB (20 dB for pyrotechnics) for the safety-class critical testing point, 9 dB for the mission-class critical testing point, and 6 dB for other testing points.

6.2.5.2 Contents and Process of Overall Spacecraft EMC Design

The main contents of overall spacecraft EMC design include the design of model EMC specifications (or EMC technical requirements), the design of equipment EMC test matrix, the analysis of spacecraft EMC frequency, the confirmation of EMC test conditions and results of model equipment products, the analysis of spacecraft EMC safety margin, the design of spacecraft EMC test program and the verification and summarization of spacecraft tests. The detailed process is shown in Figure 6.8.

At the beginning of the spacecraft system design, EMC safety margin analysis can be used as the first step of model EMC specification design. Through the preliminary analysis of such inputs as spacecraft development assignment, RF equipment frequency and preliminary spacecraft configuration layout, the anti-interference and coupling degrees of RF link can be preliminarily estimated, and the EMC safety margin requirements for key testing points of the system can be obtained. Then, the designer can preliminarily analyze the design requirements for system antenna layout (such as the degree of isolation between antennas), the design requirements for cable layout (special cable coupling) as well as the design requirements for the limits of specific EMC test items of a model product (including the requirements for conduction and radiation emission as well as conduction and radiation immunity) and include those requirements to the model EMC specifications.

In the middle and late stages of model development, model RM-satellite test verification (or antenna isolation analysis) and equipment EMC test verification should be arranged. The EMC test results of model equipment, the antenna isolation analysis results (or RM-satellite measurement results), the simulation analysis results of cable harness coupling (or cable measurement results) and the simulation analysis results of electromagnetic shielding degree of equipment shell are used to analyze the EMC safety margin of the whole satellite, and calculate and analyze whether the EMC safety margin of the key testing points of the system can meet the requirement of 6 dB.

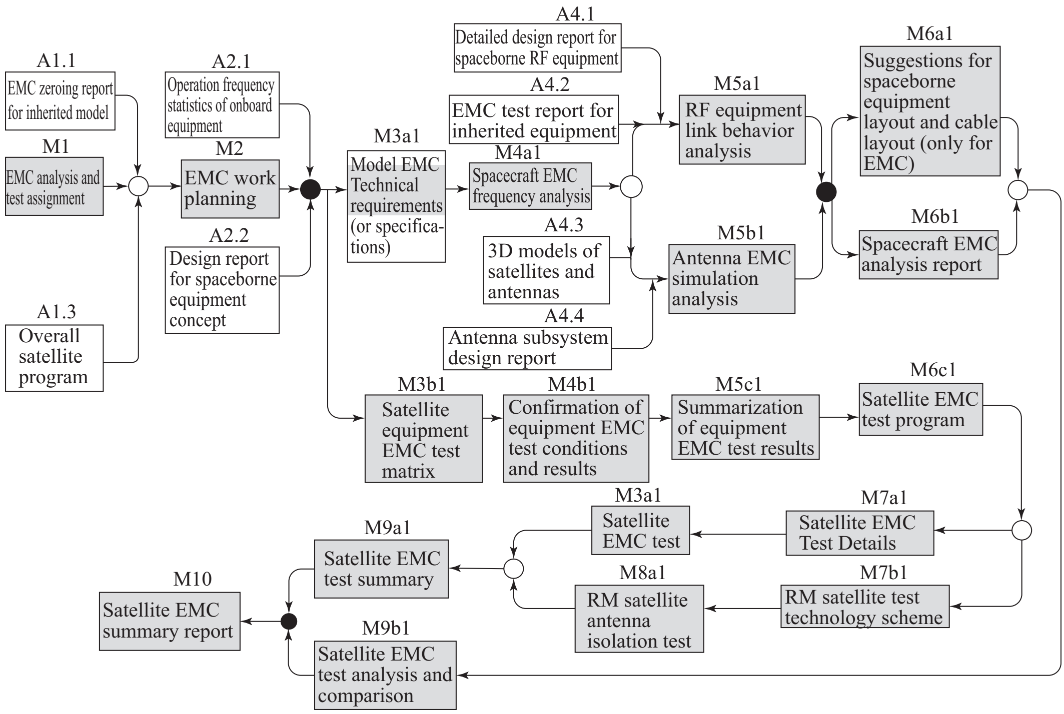

FIGURE 6.8 Contents and process of overall spacecraft EMC design.

In the later stage of model development, the interference pairs with small safety margin are obtained according to the analysis of whole-satellite EMC safety margin, so as to design the test items for satellite EMC test program. Finally, the satellite EMC safety margin is obtained through whole-satellite EMC test verification.

6.2.5.3 Introduction of the Analysis Method of EMC Safety Margin of Spacecraft System

As can be seen from the definition, the comparison of EMC safety margins is neither required in the frequency domain or time domain nor specified in terms of voltage, current, power or energy, so all the aspects need to be considered simultaneously. In addition, the margin requirement does not specify whether 6 dB is over-design. This requires the overall analysis by the System Engineering from the perspectives of development cost and cycle. Most importantly, the margin requirement does not require any accuracy but requires enough margin to ensure that the final system is compatible with the results. All these make the EMC analysis of the system more difficult and complicated.

Therefore, the determination of key testing points becomes the first and most important step in the prediction of system EMC safety margin. According to the development experience of spacecraft models, this work can be done by analyzing the three elements of EMI. First, potential interference pairs (or potential sensitive sources and interference sources) are identified through task analysis (analysis of inherited platform characteristics, payload mission characteristics, RF frequency, EMC test results of inherited equipment models, RF link anti-interference simulation analysis etc.). Second, the possible coupling paths between potential interference pairs are analyzed to further confirm the relevant analysis method of interference coupling paths. The testing points that need more attention are confirmed through antenna isolation analysis (analyzing the coupling degree between catalog RF antennas), cable harness coupling simulation analysis (analyzing the interference coupling degree of power supply and cable network in the low-frequency time domain or frequency domain), shell-specific electromagnetic shielding simulation analysis (analyzing the coupling between unintentional equipment radiation emission and sensitive antenna) and other means. Finally, the critical testing points are identified according to the importance of the disturbed object (primary load, onboard equipment or platform equipment etc.).

Next, the prediction process of RF EMC safety margin will be highlighted. The same analysis method is applicable to the prediction of low-frequency interference margins on the power line and data line. In addition, EMC safety margin and EMI margin are opposite to each other. For convenience, this calculation process is explained below by using the RF EMI margin.

At the beginning of the spacecraft system design, the system designer should perform frequency analysis to identify the spectrum compatibility between RF transmitter and receiver. The frequency analysis results can only preliminarily confirm the existence of frequency overlap between the RF devices in the spacecraft system, which often reveals many interference pairs and forms. After screening the actual possibility of these potential interference pairs, the interference margins of key interference pairs should be calculated. The interference margin $IM(f, t, d, p)$ of the coupling-type interference between antennas is calculated by the following equation:

$$IM = P(f_E) - L_{CE1} - L_t - L_{tr} - L_r - P_R(f_R) \tag{6.1}$$

where

$P(f_E)$ is the power corresponding to the center frequency point f_E of the signal transmitted by the transmitter;

L_{CE1} is the attenuation of the power of the transmitting signal at the center frequency point of the receiving device relative to the power at the center frequency point of the transmitting signal (dB);

L_t is the feeder loss between the transmitting device and the transmitting antenna at the center frequency point of the receiving device, dB;

L_{tr} is the isolation degree between the receiving antenna and the transmitting antenna (measured by vector network analyzer), dB;

L_r is the feeder loss between the receiving device and the receiving antenna at the center frequency point of the receiving device, dB;

$P_R(f_R)$ is the sensitivity threshold of the receiving device (dBm, the indicator of equipment design or acceptance test) when the response frequency is f_R; f_R is the center frequency of the receiving device.

The interference margin for the antenna coupling between the receiving and transmitting devices with potential interference pairs is obtained by the calculation formula. However, the interference or non-interference cannot be judged absolutely because the interference also involves the random characteristics of the signal or noise and some factors (such as the measurement accuracy deviation of antenna isolation, the out-of-band suppression control deviation etc.) are not considered. Therefore, the following engineering division criteria are given according to the actual situation:

1. Basic non-interference ($IM \leq -30\,\text{dB}$)
2. Quasi-interference ($-30\text{ dB} < IM \leq -10\,\text{dB}$)
3. Critical interference ($-10\text{ dB} < IM \leq 10\,\text{dB}$)
4. Possible interference ($IM > 10\,\text{dB}$)

According to the above criteria, the margin analysis results of potential interference pairs are confirmed, and the interference correlation matrix of RF subsystem is established. For possible interference and critical interference, the corresponding measures should be taken at the acceptance stage of the equipment and subsystem to avoid interference. Meanwhile, according to the interference correlation matrix, the key objects to be observed in the whole-satellite EMC test and verification can be confirmed.

6.2.6 Usability and Ease-of-Use Design

With the development of space technology, the user's requirements for spacecrafts gradually shift from desired function and performance to good user experience. Ease of use has become an important part of spacecraft design.

6.2.6.1 Definition of Concepts

Usability: the quality and application efficiency of the relevant data for ground applications (such as the image data of optical remote sensing, the calibration and measurement data of ground laboratory etc.).

Ease of use: the degree of operation convenience in the process of using the spacecraft, including TT&C/operation control task planning, routine maintenance, instruction preparation, uplink injection, instruction execution and optimization, operation state monitoring and control and other links.

6.2.6.2 Items of Usability and Ease-of-Use Design

1. **Usability design:** in order to design a user-friendly spacecraft, the designer should fully listen to the user's feedback on data usage requirements, identify the corresponding spacecraft design links and focus on the planning, design and verification of usability improvement efforts. The usability design items of a remote sensing satellite are shown in Table 6.2.

TABLE 6.2 Usability Design Items of a Remote Sensing Satellite

No.	Working Items	
1.	Improve the target positioning accuracy	Design and verification of the high-precision onboard time system based on second pulse
1.		Analysis and design of relative positioning accuracy within the scene and in-orbit geometric calibration
1.		Auxiliary-data optimization design
1.		Integrated installation and isothermal design of attitude sensor and payload
1.		Design and verification of high-frequency and high-precision attitude measurement technology
2.	Improve the effectiveness of radiometric calibration data	Improvement of the availability of calibration data in ground laboratories
1.		In-orbit yaw calibration
3.	Improve the image-blending effect	Development of the performance indicators related to out-of-band response and spectral response characteristics
1.		Testing and verification of ground spectral characteristics
4.	Optimize the dynamic range of images	Design and implementation of the setup of onboard autonomous imaging parameters
1.		Precise setting of optical satellite imaging parameters based on satellite model
5.	Improve the accuracy of atmospheric correction	Analysis and validation of atmospheric correction method
		Acquisition planning, simulation analysis and onboard experimental verification of satellite polarization atmospheric parameters
		Analysis of imaging atmospheric correction process and sharpness evaluation method

2. **Ease-of-use design:** the ease-of-use design of a satellite aims to reduce the labor intensity of user instruction programming and uplink injection, simplify the constraints on the use of onboard energy, storage resources and data transmission and make it easy for the ground system to complete the monitoring and maintenance of in-orbit satellite operation status. The ease-of-use design items of a remote sensing satellite are shown in Table 6.3.

6.3 DESIGN OF INTERNAL PHYSICAL INTERFACES

The internal physical interfaces of a spacecraft refer to the products with independent functions, complete structures and defined mechanical, electrical and thermal interfaces in the spacecraft subsystems. The design of a single device in a subsystem is restricted by other devices in the subsystem, other devices in other subsystems and even the internal and external environments of the spacecraft. These constraints are the inputs of equipment design and need to be standardized at the system level. This section will mainly present the design and construction specifications, IDS and other important equipment-level design specifications.

TABLE 6.3 Ease-of-Use Design Items of a Remote Sensing Satellite

No.	Working Items	
1.	Interfacing between satellite and operation control system	Mission planning and design
1.		Design of mission planning constraint
1.		Design of payload control interface
2.	Interfacing between satellite and TT&C system	Design of ground/relay TT&C link interoperability
1.		Relay TT&C design
1.		Ground TT&C design
1.		TT&C data link layer design
3.	Payload handling design	Payload mode design
1.		Payload instruction timing design
1.		Payload control instruction design
4.	Satellite attitude planning and design	Planning and design of ground attitude imaging
1.		Design of planning algorithm of onboard autonomous attitude imaging
1.		Design of planning software of onboard autonomous attitude
5.	Energy usage strategy design	Analysis of onboard solar incident angle and solar wing occlusion
1.		Analysis of the constraints on single-working mode
1.		Analysis of the combined constraints on typical working modes
1.		Constraint service design for satellite energy balance
6.	Payload data balance strategy design	Data storage requirement analysis
1.		Data transmission requirement analysis
1.		Satellite data balance design
7.	Design of in-orbit monitoring and fault handling	Health data package design for operation control system
		Health data package design for TT&C system
		In-orbit monitoring design
		In-orbit fault handling design

6.3.1 Design and Construction Specifications

The design and construction specifications of a spacecraft give the general requirements for the design of instruments and equipment on a spacecraft, especially the interface requirements for design and construction. In order to ensure the system consistency and performance, all subsystems and equipment on the spacecraft must meet these basic requirements to achieve the harmony among various instruments, equipment and subsystems during the component assembly, system assembly, testing, transportation, storage, launch and in-orbit operation.

6.3.1.1 Principles of Equipment Design

6.3.1.1.1 Mechanical Design

1. The structural design of a product shall meet the requirements of strength and stiffness, while taking into account the thermal design, EMC design and charged-particle-irradiation protection design to create an appropriate operating environment for the components.

2. Based on the analysis of the product structure, the product design should have a small size and weight, and enough safety margin.
3. The structural design shall undergo sufficient structural mechanics analysis and test verification in order to meet the mechanical environmental conditions and interface requirements.

6.3.1.1.2 Circuit Design

1. The circuit design of the product shall be as simple as possible, and the interface design shall use the recommended circuit as far as possible to ensure the design reliability and standardization.
2. The standard components (including microprocessors), standard unit circuits, standard electronic function modules and common parts that have been finalized or verified shall be used.
3. The circuit of a dangerous system shall be designed to be fail-safe, so that the system will not have catastrophic consequences caused by component failure or ground operation error.

6.3.1.1.3 Thermal Design

1. In the thermal design of an equipment product, low-power components should be selected, and the equipment structure should be used for heat transfer as far as possible.
2. Sufficient thermal analysis should be carried out for the whole equipment in consideration of its extreme operating conditions while meeting the temperature derating requirements of components.
3. If necessary, the thermal balance test of the equipment should be carried out to verify the correctness of thermal analysis and the rationality of thermal design, so as to provide a basis for improving thermal design.

6.3.1.2 Main Contents of Design and Construction Specifications

The contents of design and construction specifications cover all aspects of equipment design, involving many disciplines and a wide range. The main requirements are shown in Table 6.4.

6.3.2 Design of IDS

IDS is a control document that specifies and describes the status and parameters of the mechanical, electrical and thermal interfaces of a piece of equipment. The equipment here refers to an onboard product with relatively independent functions that can be delivered as a whole. The detailed information on mechanical, electrical and thermal interfaces can be written into IDS and countersigned by the corresponding subsystem designers during

TABLE 6.4 Main Contents of Design and Construction Specifications

No.	Category of Requirement	Description of Requirement
1.	General requirements	Product assurance requirements, design principles, as well as other common requirements of each spacecraft in terms of product identification, environment, EMC, inspection, testing, service life etc.
2.	Mechanical design requirements	Requirements for configuration and interface, coordinate system, digital model, size, quality characteristics, structural design, installation, alignment, design of mechanisms and moving parts, pipeline design, assembly component design, lifting, packaging, transportation, storage etc.
3.	Thermal design requirements	General thermal design requirements, and the requirements for the temperature, margin, interface data, installation and other aspects of the equipment and its components
4.	Electrical design requirements	Requirements for power-on/off, remanence, primary and secondary power supply and distribution, grounding and lap connection, electrical connector selection, cable design, use of key components etc.
5.	RF system design requirements	Requirements for RF links, RF components, passive intermodulation protection design, and microdischarge protection
6.	EMC design requirements	General requirements, as well as the requirements for EMI safety margin and low-frequency EMI control
7.	Signal interface design requirements	General design requirements for high- and low-frequency signals and the requirements for ground equipment interfaces, telecontrol and telemetry and various buses
8.	Final-assembly design requirements	Requirements for electrostatic protection, battery charging protection, high voltage devices, and redundancies

the design of each onboard equipment. The equipment information in IDS must meet the design and construction specifications.

The common specification-type IDS items and their sequencing are shown in Table 6.5.

6.3.3 Design of Interface Control Document (ICD)

The ICD can be used for equipment design and system interface design. The IDS highlights the intuitive tabular representation of mechanical, electrical and thermal interface parameters. When ICD is used as an equipment design document, the description of functions, ground equipment and other diversified information can be added. ICD is commonly used in foreign equipment design (Table 6.6).

6.4 FLIGHT PROGRAMMING

The flight program of a spacecraft usually refers to the main program flow from the prelaunch setting of initial satellite state at the launch station to the delivery of the whole spacecraft to its orbit and the routine program used in the normal flight phase. The flight program shall list the flight events in chronological order. The work contents of the spacecraft vary greatly during the in-orbit operation, especially in different phases of the initial orbiting period, so the flight program is generally divided into several stages for design. Flight programming is one of the most important tasks in overall satellite design.

TABLE 6.5 Main Design Contents of Spacecraft IDS

No.	Name	Content
1.	IDS A.0 Table of Contents	Document number, subsystem name, equipment name, equipment code, model spectrum code, cabin, version, development organization, stage mark, catalog list, template number, page number, number of pages etc.
2.	IDS A.1 Mechanical Properties IDS	Mass per unit, quantity, size, location of the center of mass, inertia through the center of mass, number of mounting holes, diameter of mounting holes, mounting contact area, mounting surface flatness, mounting surface roughness, mounting surface material, mounting surface treatment state, maximum outer dimensions, ground stud specifications, parameter relationship diagram, grounding method, mechanical properties and other special notes
3.	IDS A.2 Equipment Sketch IDS	3D model diagram, 3D view diagram, installation diagram, footprint diagram, as well as other information, special notes etc.
4.	IDS A.3 Thermal Properties IDS	Surface material, range of hemispherical emissivity, range of solar absorption ratio, mounting contact area, range of starting temperature, global flatness of mounting surface, range of operating temperature, range of storage temperature, heat capacity, relative temperature and humidity in orbit, heat consumption, diagram of temperature reference point location, working mode and other special notes
5.	IDS A.4 Heat Distribution IDS	Schematic diagram of mounting surface, special notes etc.
6.	IDS A.5 Circuit and Interface Schematic IDS	The electrical interface relationship between a device and other devices, the electrical schematic diagram of internal circuit of the device and its external interface and the relationship between the relevant function blocks inside the device, including at least the following contents: a. Main functional blocks of the device; b. Frequency and frequency-conversion process (such as local frequency, clock frequency, intermediate frequency, the switching frequency of DC/DC converter etc.); c. Internal backup and mode conversion relationship; d. Power supply and grounding diagram; e. Input and output interface signals; f. Necessary instructions regarding electrical connectors.
7.	IDS A.6 Power Supply IDS	Operating time after single non-long-time power-on, total number of devices, power supply properties, operating mode, power supply voltage, voltage stability, ripple voltage, starting-current characteristics, power consumption, curves of dynamic power consumption etc. (the operating time after non-long-time power-on as well as a further explanation of the above contents or the contents beyond the above items can be used as notes); the continuous power consumption, intermittent power consumption and peak power consumption of each flight phase (the intermittent power consumption and peak power consumption should be indicated with time and can be used as notes)
8.	IDS A.7 Telemetry Parameters IDS	Telemetry parameter category, parameter code, parameter name, parameter type, parameter description, output level, output impedance, special notes etc.

(*Continued*)

TABLE 6.5 (*Continued*) Main Design Contents of Spacecraft IDS

No.	Name	Content
9.	IDS A.8 Instructions IDS	Instruction category, code, name, type, instruction description, width, pulse amplitude, load impedance, verification, special notes etc.
10.	IDS A.9.1 Electrical Connectors IDS	The type, code, name and model of electrical connector, the type of pins/holes, the code of electrical connector for the corresponding device, and the information to be commented
11.	IDS A.9.2 RF Interface Management IDS	Code of electrical connector, model of electrical connector, signal content, heat consumption, connection and the information to be commented
12.	IDS A.10 Electrical Connections Distribution IDS	Code of electrical connector, model of electrical connector, cabin position, contact number/type/code, signal content, (contact load) nature, (contact load) voltage, (contact load) current, polarity, shielding requirement, twisted-pair requirement, the information to be commented, special notes etc.
13.	IDS A.11 Electrical Interface Properties IDS	Signal type, signal description, signal characteristics, interface circuit, the information to be commented etc.
14.	IDS A.12 Equipment Specifications IDS	The interface contents not suitable for IDS 1-IDS 12, the special requirements of equipment interface and the related additional information, such as the delivered state, operating requirement, storage requirement, transportation requirement, periodic testing requirement and safety requirement of the equipment and its components, and the important protection requirements or notes for special accessories, final assembly or electric logging
15.	IDS A.13 Equipment Grounding Diagram IDS	Equipment grounding diagram, the information to be commented etc.
16.	IDS A.14 Location of Electrical Connectors and Ground Studs IDS	The code, model, mounting surface, direction and coordinates of electrical connector/ground stud, and the coordinates (X, Y) of mounting hole point
17.	IDS A.15 Ribs and Lugs IDS	Basic size definition, lug unit definition, type, position, right angle etc.

6.4.1 Definitions Related to Flight Program

The spacecraft flight orbit is calculated based on the nominal orbital parameters from the nominal injection point of the launch vehicle (the satellite-rocket separation point). The actual trajectory will be different from the calculated trajectory due to the offset of the initial orbit and the orbit adjustment after injection. The flight program can be adapted to the actual orbit. To define the number of orbiting cycles, the arrival of the flying satellite at the N-th descending node after injection marks the end of its flight in the N-th cycle and the beginning of its flight in the $(N+1)$-th cycle.

The visible orbit segments are generally defined as follows:

1. **Satellite-ground TT&C segment:** the orbit segment where a communication link can be established between the satellite-borne TT&C antenna and the ground TT&C station to remotely control and meter the satellite during its in-orbit flight.

TABLE 6.6 Main Design Contents of Spacecraft ICD

No.	Name	Content
1.	General	Purpose and scope
2.	General Equipment Information	Name, function description etc.
3.	Mechanical Properties	Equipment identification, coordinate system, equipment diagram, mass characteristics, installation characteristics, vent hole etc.
4.	Thermal Properties	Temperature reference point, temperature range, temperature stability, heat dissipation information, heat capacity, radiating surface, thermal model, photothermal characteristics
5.	Electrical Properties	Allocation of functional interfaces (e.g. telecontrol and telemetry), electrical connectors, important power interfaces, command and control interfaces, RF interfaces, connection requirements etc.
6.	Ground Support Equipment	Explanation of ground support equipment such as test equipment and ground tools
7.	Ground Management	Explanation of cleanliness as well as various restrictions and constraints

2. **Relay TT&C segment:** the orbit segment where a communication link can be established between the satellite-borne relay antenna and the orbiting relay satellite to remotely control and meter the satellite during its in-orbit flight.

3. **Satellite-ground data transmission segment:** the orbit segment where a communication link can be established between the satellite-borne data transmission antenna and the ground data transmission station to transmit the satellite remote sensing data and platform data down to the ground during the in-orbit flight.

6.4.2 Constraints and Supporting Conditions

The flight programming originates from the spacecraft concept design. The design of the whole system and its subsystems plays an effective role in supporting the flight programming. In addition, the design boundary of the whole system and its subsystems also plays a certain role in constraining the flight programming.

6.4.2.1 Constraints

1. **TT&C constraints:** the visibility of TT&C station is one of the important constraints. The telecontrol and telemetry items of the spacecraft should be completed with enough time margin in the TT&C segment, which should also include the relay TT&C segment.

2. **Payload constraints:** the payloads on different satellites may vary greatly and have their own unique features, which need to be considered during the flight programming. The onboard payload operations generally follow a strict time sequence, including equipment power-on, power output and so on. The working time of different payloads at different modes is also different, possibly from a few minutes to dozens of minutes, or even around the clock. For some payloads in all-time rotation

with a large inertia (such as the microwave scatterometer and microwave radiometer of Chinese HY-2 satellite), their influence on the spacecraft attitude should be minimized by means of bidirectional counter-rotation during the flight programming.

3. **Data transmission constraints:** the rate and segment of data transmission are one of the constraints on flight programming. After completing the corresponding observation, navigation, communication and other tasks in orbit, the spacecraft should transmit the acquired data to the ground in time through the selection of an appropriate orbit according to the data volume to be transmitted and the capacity of the data transmission equipment. The flight programming should not only ensure the real-time data transmission but also avoid the delay of main task.

4. **Thermal control constraints:** the spacecraft thermal control should keep both the temperature of all the onboard instruments and equipment and the ambient temperature of the spacecraft structure itself within the required range, so as to ensure the normal operation of the spacecraft in orbit. The flight programming needs to be carried out within the spacecraft's thermal control capacity. If the working time of a payload is too long, its temperature may rise even beyond the required temperature range. In this case, the flight program needs to be adjusted according to the actual situation to ensure that the thermal requirements of the spacecraft are met.

5. **Energy constraints:** the contents of flight program are restricted by the spacecraft energy. The energy balance budget should serve as one of the constraints on flight programming, mainly considering the following two aspects: the power output of solar arrays should meet the requirements of payload power consumption and battery charging; the discharge depth of the batteries should always be controlled within the range required by the cycle life. The contents of flight program should be considered comprehensively according to the analysis results of energy balance in the current cycle or multiple cycles.

6. **Other constraints:** in addition to the above constraints, the different characteristics of different spacecrafts should be considered in the flight programming. If the orbital maneuver attitude of a satellite is not consistent with its flight attitude, special attention should be paid to the change in the starting & ending time and duration of the TT&C segment before and after the orbital transfer.

6.4.2.2 Supporting Conditions

1. **Support for usability and ease-of-use design:** to better serve the user, the usability and ease-of-use design are regarded as not only an important step in the spacecraft design process but also a strong support for flight programming. For example, in the mission planning of a satellite, the effective concentration of multiple tasks into a single data packet for uplink can reduce the complexity and risk of multi-packet uplink, shorten the time of task arrangement, increase the operational flexibility in each TT&C segment and provide convenience for flight program arrangement.

2. **Equipment capacity support:** strong equipment performance can provide convenience and reliability for flight programming, and even improve the in-orbit spacecraft performance. For example, the high dynamic capability of a new star sensor and its adaptability to rapid spacecraft maneuvering can obtain much higher attitude determination accuracy in the maneuvering arrangement than the previous infrared attitude determination. The high dynamic positioning capability of TT&C subsystem can ensure the continuous high-precision positioning of the spacecraft during the maneuver.

3. **"Six characteristics" support:** the common "six quality characteristics", namely reliability, maintainability, security, testability, safety and environmental adaptability, provide an effective support for the design of spacecraft flight program. For example, the maintainability designs, such as the component replacement and refueling at a space station, can increase the flexibility of flight programming.

6.4.3 Principles and Contents of Flight Programming

The spacecraft flight programming is subject to the requirements of application, delivery, TT&C and other aspects. For details, see the Rule for Formulation of Satellites Flight Program (GJB2498-95).

1. The design and arrangement of spacecraft flight program must meet the requirements of the application system.
2. The flight programming must consider the constraints on the launch facilities and launch vehicle and must be compatible with the pre-launch program of the launch vehicle.
3. The events requiring a time sequence must be arranged in strict accordance with their execution time without being reversed.
4. The important events being programmed should have the execution time margin, the execution mode redundancy, and the opportunities for repeated execution. When both onboard autonomous control and ground TT&C can enable their execution, ground TT&C is generally preferred.
5. It is necessary to reduce the risk of program execution, improve the safety of program execution, increase the probability of task completion and improve the efficiency of program execution.
6. The flight programming must consider the constraints on the function of ground TT&C network and must be compatible with the ground TT&C network.

The requirements for flight programming are as follows:

1. List the event items and their main functions in different flight stage programs.
2. List the time required to execute each event.

3. List the conditions required for the execution of each event.
4. List the parameters and criteria required for the judgment of each event result.
5. Arrange the sequence and time of events.

6.4.4 Analysis of TT&C Conditions

The TT&C segment analysis in flight programming mainly considers three aspects: ground TT&C station, relay satellite and ground instrumentation ship. After determining the launch orbit parameters through coordination with the rocket developer, the spacecraft developer will analyze the post-injection TT&C segment according to the injection point, the layout of ground TT&C stations and the application of relay satellite, and if required, will cooperate with ground instrumentation ship to complete the telecontrol and telemetry task at the corresponding time.

The requirement for TT&C segment time varies with the spacecraft tasks in different phases. The telecontrol and telemetry task in each segment shall be rationally allocated according to the available TT&C resources and the following key considerations:

1. The elevation angle of the spacecraft relative to ground TT&C station is within the range allowed by the equipment.
2. The angular velocity limit of the spacecraft relative to ground TT&C station.
3. The operating range of the measuring equipment.
4. The spacecraft TT&C antenna effectively covers the ground stations, relay segment and instrumentation ship.

In some cases, the satellite is required to perform attitude maneuver and orbital maneuver due to the task needed. For example, if there is an Earth-oriented waist pit in the radiation pattern of TT&C antenna array, part of the TT&C segment will be unusable. When the complete TT&C segment is discontinued in the middle, the uplink of commands and data blocks can be arranged only in the visible segment.

In addition, the TT&C link margin and the TT&C segment time margin must also be considered in the flight programming. Full consideration before the spacecraft flight control is helpful for making real-time decision during the in-orbit flight process.

6.4.5 Process of Flight Programming

The design of flight program varies with the spacecraft. For example, the HEO spacecraft and LEO spacecraft are different in orbital maneuver, and the remote sensing satellite, navigation satellite and communication satellite are different in payload settings. The main steps of flight programming are shown in Figure 6.9.

Next, we take a remote-sensing LEO satellite in the sun-synchronous orbit at an altitude of 700 km as an example to briefly describe the design steps of its flight program.

The orbit starting from the injection point (after the satellite-rocket separation) to the first descending point is the first orbit, whose cycle number is Q_1. The cycle number

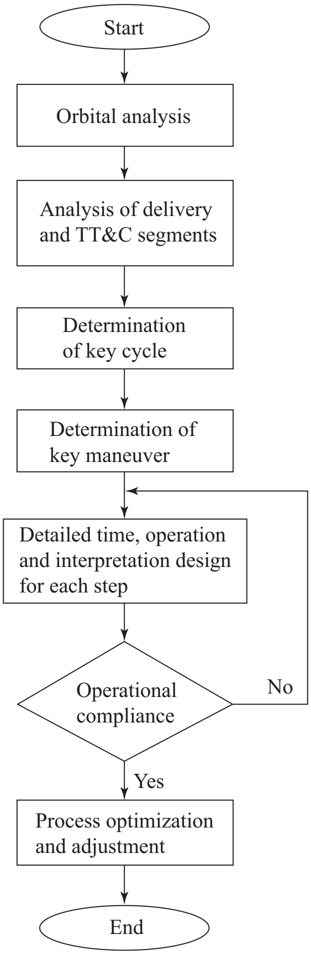

FIGURE 6.9 Flight programming process of a remote sensing satellite.

corresponding to the powered phase is Q_0. The unlocking of electric explosive device (EED) is an important event in the flight process. The cycle in which the unlocking of solar wings and various antennas takes place is set as critical cycle. In the critical cycle, a sufficient time margin should be left for the operation of ground uplink instruction and a complete troubleshooting plan should be made. For the low-orbit remote sensing satellite in this example, the time from the orbit insertion to the imaging data transmission lasts about 1 week. The initial orbit setting, platform setting, payload setting and other steps should be reasonably arranged according to the number of visible cycles per day.

6.5 DESIGN AND VERIFICATION OF EXTERNAL SYSTEM INTERFACES

The spacecraft engineering includes not only the spacecraft itself but also the launch vehicle, launch site, tracking and TT&C network and ground application system. In addition to the above items, the manned spaceflight engineering also includes the astronaut system, return landing site and so on. The application systems in different application directions are quite different, such as communication satellite application system, navigation satellite application system, meteorological satellite application system, remote sensing satellite application system etc. Even the same satellite can be used in different fields. Therefore, the interfaces between satellites and application systems are diverse and need to be designed in detail according to specific tasks. The launch vehicle, launch site, TT&C system and ground-receiving system have certain universality. This chapter describes the design and verification of the interfaces between the spacecraft and these four large systems.

6.5.1 Design and Verification of the Interfaces to Launch Vehicle

6.5.1.1 Overview of Launch Vehicle

The main task of a launch vehicle is to launch the spacecraft into the predetermined orbit according to the specified parameters and tolerances, and release and separate from the spacecraft according to the specified attitude and angular velocity. The launch vehicle capacity, the cowling envelope space as well as the vibration, heat and electromagnetic environment during launch are important constraints affecting the spacecraft design, so the selection of launch vehicle model and the design of launch vehicle interface are important work in the process of satellite development.

China's mature launch vehicles in service mainly include CZ-2C, CZ-2D, CZ-2F, CZ-3A/B/C and CZ-4B/C. Among them, the CZ-2C/D and CZ-4B/C are mainly used for launching the low-orbit satellites; the CZ-3A/B/C are mainly used for launching the spacecrafts into geostationary transfer orbit (GTO); and the CZ-2F is mainly used for launching large spacecrafts such as space ships and space stations. The rocket spectrum is shown in Figure 6.10, and their carrying capacities are given in Table 6.1.

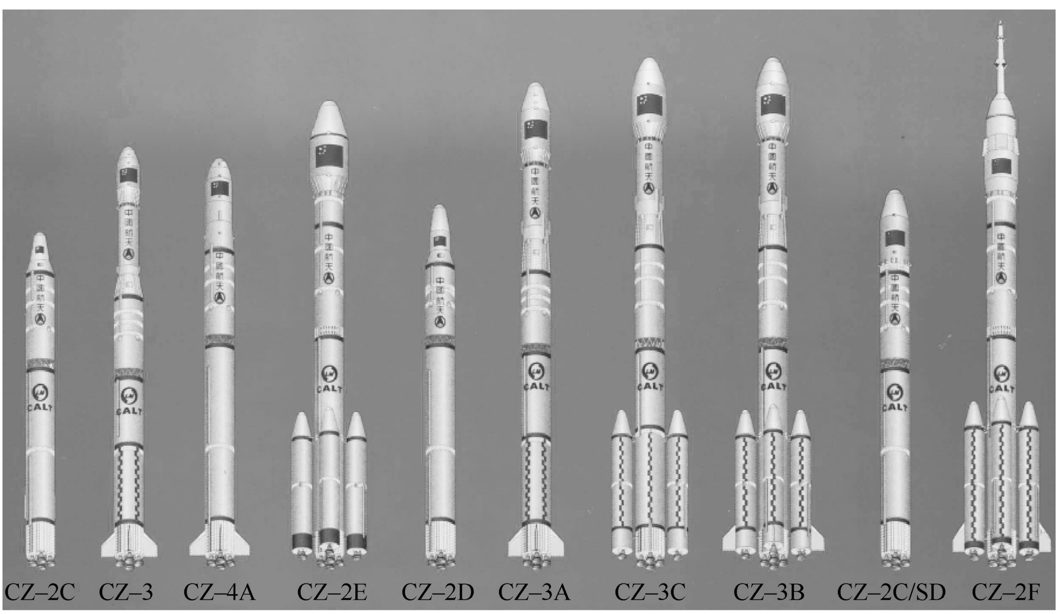

FIGURE 6.10 China's rocket spectrum in service.

In recent years, China's aeronautical institutes have developed new launch vehicles (such as CZ-5) to meet different needs, e.g. high reliability, large carrying capacity and low cost. With the use of these new vehicles, China's rocket spectrum will become more complete and diversified.

6.5.1.2 Design of the Interfaces to Launch Vehicle

6.5.1.2.1 Carrying Capacity and Model Selection The satellite-rocket interfaces refer to not only the mechanical, electrical and thermal interfaces in narrow sense between the satellite and the rocket but also all the technical conditions affecting the other side and even the agreements on the division of labor, mainly including orbital parameters, injection accuracy, satellite-rocket separation method, separation attitude orientation and accuracy, fairing, satellite-rocket docking/separation interfaces, operation space, electrical interface, cable branch, grounding, telemetry and telecontrol, electromagnetic environment and interface, dynamic environment, temperature and humidity, cleanliness environment and other conditions.

In the process of spacecraft development, the above technical conditions and management factors (such as cost) should be considered comprehensively when selecting the model of launch vehicle. Once the launch vehicle is selected, these technical conditions will in turn become the constraints on spacecraft design. Therefore, in the development process of medium and large spacecrafts, the model and basic interfaces of launch vehicle should be determined during the feasibility demonstration or project demonstration, and the satellite-rocket interfaces should be further refined and defined. However, the microsatellites are often launched through piggybacking or "multiple satellites with one rocket". Due to the piggybacking time limitation, the rocket model is often determined quite late, but the rocket capacity, interface and other factors should also be considered, designed and verified. Due to the uncertainty of early launches, the microsatellites should have a greater adaptability so as to minimize the dependence on specific carrying conditions.

The carrying capacity is the first factor to consider when selecting the rocket model and is mainly a function of such parameters as orbital altitude and inclination. Table 6.7 gives the main orbits of China's active spacecrafts and the corresponding carrying capacities, and Figure 6.11 shows the carrying capacities of China's CZ-4B rocket at different orbital inclinations. In practice, the launch capacity is also constrained by the factors such as launch site and launch impact point.

6.5.1.2.2 Mechanical Interface

6.5.1.2.2.1 Mechanical Configuration The satellite is connected to the launch vehicle by an adapter. The satellite configuration is influenced by the adapter form, the effective space in the fairing and the dynamic characteristics of satellite-rocket coupling.

The adapter generally includes an unlocking separation device. In the case of multi-satellite launch, the connection and separation between the satellites and the carrier rocket are realized through parallel separation, series separation or series-parallel separation, as shown in Figures 6.12–6.15. In the parallel separation, the satellites are connected to and

TABLE 6.7 Capacities of China's Carrier Rockets in Service

No.	Rocket Code	Carrying Capacities of Main Mission Orbits (kg)	Main Orbit	Propellant	Takeoff Mass (t)	Takeoff Thrust (kN)	Total Length (m)
1.	CZ-2C	1400	SSO	Full UDMH/N_2O_4	242.5	2,962	43
2.	CZ-2D	1150	SSO	Full UDMH/ N_2O_4	250	2,962	41
3.	CZ-2F	8100 (manned)	LEO	Full UDMH/N_2O_4	498	5,923	52
		8600 (target)			493	5,923	52
4.	CZ-3A	2600	GTO	Liquid hydrogen and oxygen for the third stage, and UDMH/N_2O_4 for other stages	243	2962	52.5
5.	CZ-3B	5500	GTO		456	5,923	56.5
6.	CZ-3C	3800	GTO		345	4,443	54.8
7.	CZ-4B	2370	SSO	Full UDMH/N_2O_4	250	2,962	47
8.	CZ-4C	2800	SSO		250	2962	47

SSO, sun-synchronous orbit; LEO, low Earth orbit; GTO, geostationary transfer orbit.

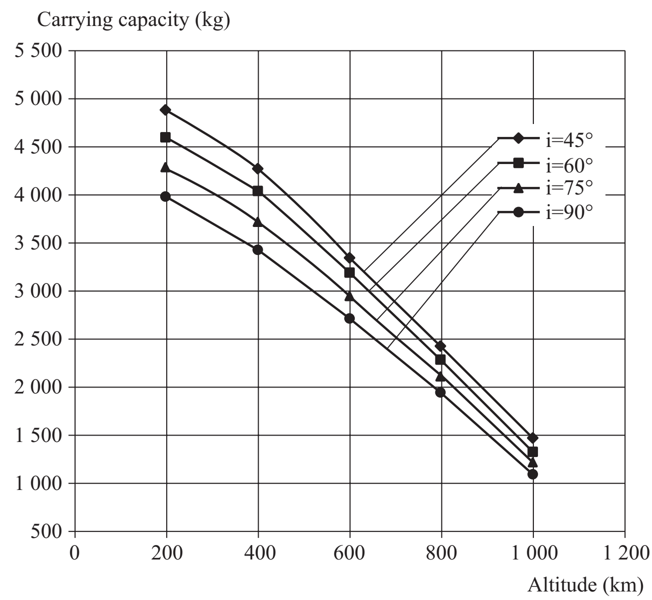

FIGURE 6.11 Carrying capacities of China's CZ-4B rocket in circular orbits with different inclinations.

separated from the rocket respectively, but they are not directly connected to each other. In the series separation, the bottom satellite is connected to the rocket, and the other satellites are connected to the bottom satellite or the lower support structure successively. The series-parallel separation is a combination of the above two patterns.

6.5.1.2.2.2 Fairing The fairing provides a favorable environment for the spacecraft and launch vehicle during the ground transportation, testing and atmospheric flight, and reduces the influence of external gas, heat, force and other factors. Although the fairing

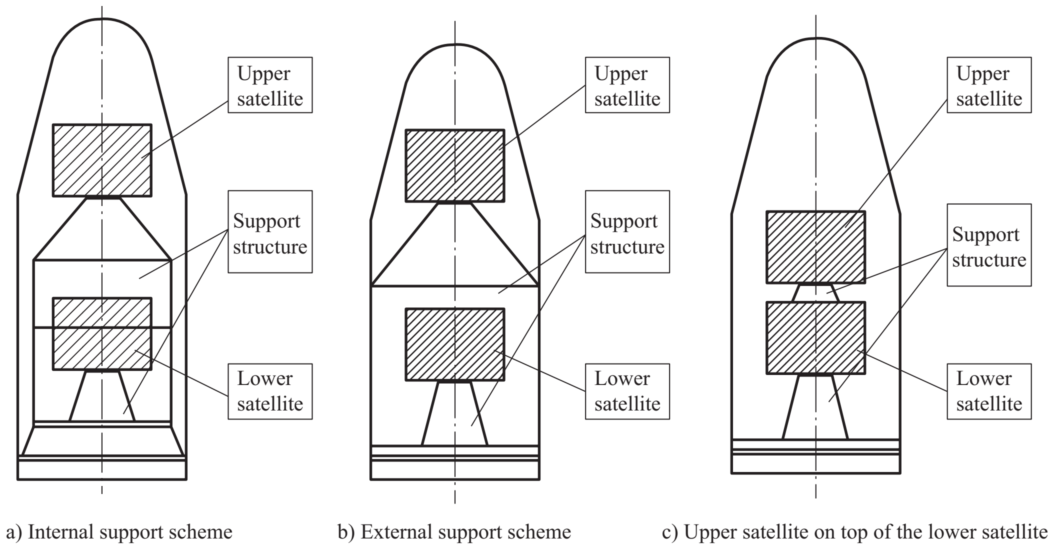

FIGURE 6.12 Two satellites in series.

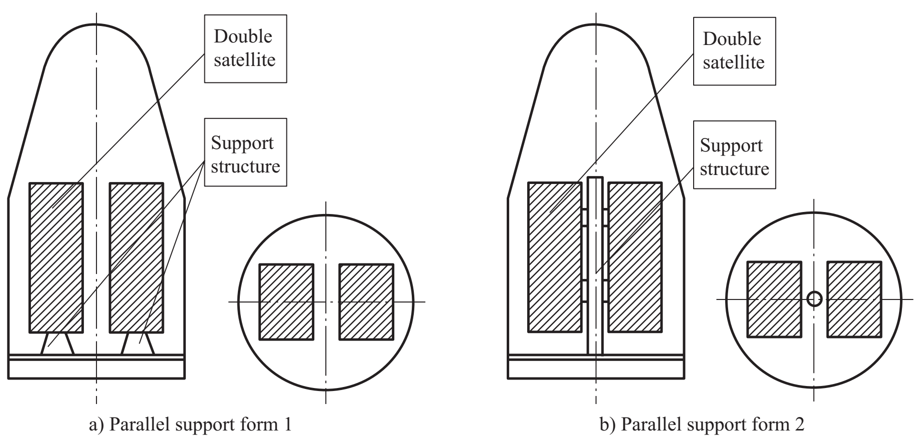

FIGURE 6.13 Two satellites in parallel.

has no direct contact with the spacecraft, its internal space size constrains the outer envelope size of the spacecraft. Each rocket model can generally be applied to several fairing sizes, and the fairing model can be selected according to the spacecraft size and launch form.

The main internal area of a fairing is typically a cylindrical space, which shrinks inward into a cone and an inverted cone in the upper and lower parts. Inside the fairing, there are often ringlike or longitudinal structural bulges, which should be avoided by the available satellite envelope. The available space provided by the rocket developer has generally considered the dynamic envelopes of the fairing and satellite, including the vibration caused

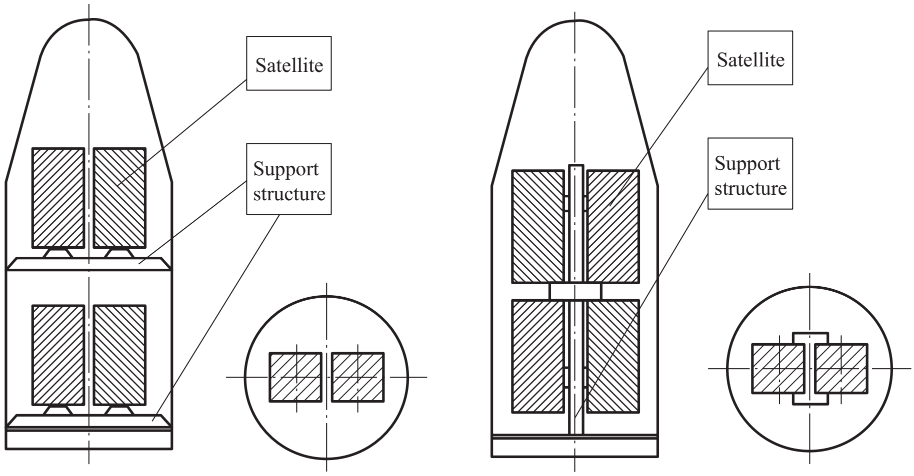

FIGURE 6.14 Four series-parallel satellites.

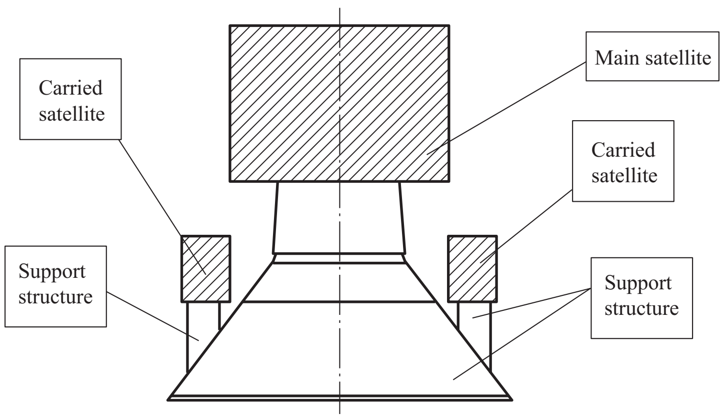

FIGURE 6.15 Piggyback scheme.

by the transmitting environment and fairing separation. Their dynamic envelopes should be reconfirmed during the satellite-rocket coupling analysis.

The fairing can provide an operating window and a wave-transmitting opening according to the satellite requirements. The wave-transmitting opening is made of a wave-transmitting material that can meet the requirements of on-tower satellite testing and launch telemetry. The operating window can be opened and closed, applying to the operations such as the cleaning of local satellite surface and the removal of protective cover. The positions of the wave-transmitting opening and operating window should be determined according to the position of the corresponding onboard product and avoid the load-bearing structure of the fairing.

When the satellite is docked with the fairing in the technical building and transported to the launching area as a whole, the fairing shall also ensure the satellite environment during transportation. Generally, the air with the required temperature, humidity and

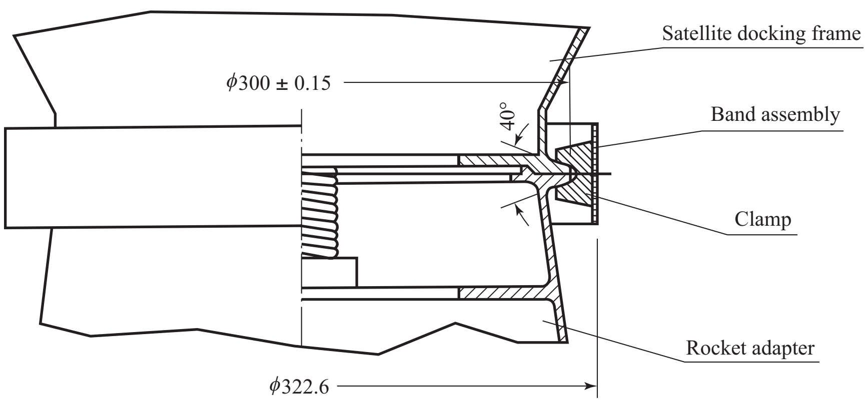

FIGURE 6.16 Band connection structure.

cleanliness shall be provided by an air-conditioning vehicle to the fairing. When the satellite rests on the launch tower, the air inside the fairing is provided by the air-conditioning system of the tower. The air supply into the fairing should avoid the high-speed airflow blowing the satellite directly.

The timing of fairing separation should consider the influence of carrying capacity and atmosphere on the spacecraft. The spacecraft design generally does not consider the environmental influence of dense airflow. It is required that the orbital altitude should be as high as possible and the atmospheric density should be small enough when jettisoning the fairing. In this case, the atmosphere has very small dynamic and aerodynamic heating effect on the spacecraft.

After being jettisoned, the fairing leaves the spacecraft and carrier rocket under the joint action of the separation driving mechanism and airflow. This process must be strictly analyzed and verified to avoid the collision between the fairing and the satellite/rocket. The available space of the spacecraft must avoid the movement space of the fairing in the jettison process.

6.5.1.2.2.3 Satellite/Rocket Separation Interface The spacecraft is mounted on the launch vehicle via a bracket (adapter). The lower end of the bracket is connected with the rocket through bolting, and its upper end is connected with the spacecraft through a separation device. The connection and separation between satellite and launch vehicle are generally realized by using the bands or points.

The satellite-rocket unlocking mechanism of the band connection (as shown in Figure 6.16) consists of bands, explosive bolts, V-shaped clamps, cylinders, limit springs, tension springs and other components. The structures near the contact surface between spacecraft and launch vehicle are wedge-shaped and are clamped together by V-shaped clamps. The bands produce a circumferential pretightening force, thus generating a radial pressure on the V-shaped clamps uniformly distributed in the circumferential direction. The radial pressure is then changed by a wedge-shaped structure into a longitudinal pressure pressing the satellite and rocket together. The role of the limit spring is to limit the rebound range of the bands after the initiation of explosive bolts and ensure a certain envelope of the bands. The role of the tension spring is to leave the unlocked band-locking device on the side wall of the adapter to avoid collision with the satellite. According to the interface diameter of the docking frame, the commonly used band interfaces are divided into Φ300, Φ660, Φ937, Φ1194, Φ1194A, Φ1194B and Φ1828 interfaces, which have been standardized.

In the mode of point connection and unlocking, the satellite and adapter are directly connected through multiple explosive bolts. Compared with band connection, this method has a higher reliability requirement as all the explosive bolts must be unlocked successfully before the separation of satellite and rocket. In addition, the forces on the chain joints are more concentrated, the requirements on local structure are higher, and the impact of the unlocking on the spacecraft is greater, so this method is often applied to small and medium-sized spacecrafts. However, owing to its flexible use and fewer constraints on the structural form of a spacecraft, its application is gradually increasing and begins to appear on large trussed spacecrafts.

After the satellite-rocket separation, the separation velocity is generated by the separation spring or the backthrust rocket. The separation spring is installed on the adapter bracket and acts on the spacecraft through a push rod. Like other mechanical interfaces between the satellite and the rocket, the interface to the separation spring also needs agreement.

6.5.1.2.3 Indicators of Satellite-Rocket Separation

6.5.1.2.3.1 Separation Status In addition to the conventional attitude with vertical axis along the flight direction, the attitude of the rocket and satellite during their separation can be adjusted until their vertical axes point to the ground according to the satellite requirements. The two sides should agree on the separation attitude, angular velocity and tolerance.

6.5.1.2.3.2 Anti-collision and Anti-passivation Measures after Separation After the satellite-rocket separation, the rocket orbit should be reversed so that the rocket is far away from the satellite to avoid the collision with the satellite. Meanwhile, in order to avoid the disintegration and explosion of the rocket body in long-term orbiting, the rocket also needs to discharge the remaining propellant to realize its passivation treatment.

6.5.1.2.4 Electrical Interface In the launch area, the spacecraft is electrically connected through the satellite-rocket electrical interface and the umbilical cable provided by the rocket design (the umbilical cables on some spacecrafts are not connected with the corresponding rockets) after being lifted and mechanically and electrically docked with the rocket. The ground TT&C support equipment can power, meter and control the satellite through umbilical cable.

The umbilical cable usually transmits three signals, including the signal of spacecraft power supply and wired TT&C, the satellite-rocket separation signal and the telemetry signal forwarded through the rocket. The latter two signals are transmitted after the umbilical cable is connected through the rocket.

The satellite-rocket unlocking device and the electrical separation connector are controlled by the rocket. The separation signal can be obtained by both the satellite and the rocket through the jumper wire state of the separation connector or through the micro-stroke switch on the separation surface.

6.5.1.2.5 Electromagnetic Environment and Interface The TT&C subsystem, data transmission subsystem (active during ground test and inactive during launch), navigation subsystem and other high-frequency products of both the satellite and the rocket will transmit wireless signals when working. The two sides shall define the RF points, check the EMC between the satellite and the rocket, formulate the radio management measures and carry out the EMC test.

6.5.1.2.6 Dynamic Environmental Conditions and Verification Due to the factors such as engine thrust fluctuation, aerodynamics, separation and unlocking, the satellite is affected by dynamic environment conditions (such as quasi-static state, low-frequency vibration and high-frequency vibration) during the launch process. The dynamic environment found during launch is a key factor to be considered in the process of satellite development.

6.5.1.2.6.1 Frequency Requirement To avoid the dynamic coupling between the satellite and the rocket and ensure the rocket controllability, the frequency of the overall satellite structure should be higher than the minimum lower limit allowed by the rocket. And after the completion of satellite structure design, the satellite-rocket coupling analysis should be carried out to verify the satellite-rocket coupling condition, the satellite's dynamic load conditions and the rationality of satellite structure stiffness.

6.5.1.2.6.2 Design Load The design load factor of a spacecraft is the sum of static overload factor and dynamic overload factor during the rocket flight. The lateral load on the satellite is maximized in the transonic phase and maximum dynamic pressure phase. The first-stage engine shutdown is the stage where the satellite's static load is the most serious. In the first-stage separation and second-stage sustainer shutdown, the longitudinal dynamic load is the most serious. After considering an appropriate safety factor, the design load factor is the basic input for the design and validation of the main satellite structure.

6.5.1.2.6.3 Vibration Environment The vibration environment in the launch process includes low-frequency vibration and high-frequency vibration, which are generally represented by sinusoidal scanning vibration and random vibration respectively. On this basis, the satellite is designed, tested and verified.

6.5.1.3 Verification Test of the Interfaces to Launch Vehicle

In the development process, the mechanical, electrical and EMC docking tests should be carried out on the satellite-rocket interfaces to verify the correctness of interface design and production:

1. The docking test between the docking stage of the spacecraft and the third-stage instrument module of the rocket, which is used to check the matching and correctness of mechanical interfaces between the spacecraft and the rocket.
2. The docking test between the spacecraft cable and the rocket cable, which is used to check the matching and correctness of the cable interface.
3. The EMC test between the satellite and the rocket, which is used to check the radio compatibility between them.

The above docking tests must be carried out in the case of flight modeling and can be tailored according to the inheritance and maturity of interface design in the case of concept design and prototyping. On the launch site, both sides also need to carry out the EMC test, joint operation inspection and other test verification.

In addition, the joint tests that can be arranged according to the design margin and the analysis accuracy requirement include joint dynamics test, fairing compatibility verification, operation window verification, on-site joint drill and other tests.

6.5.2 Design and Verification of the Interfaces to TT&C System

6.5.2.1 Overview of TT&C System

The main task of TT&C system is to track and measure the satellite orbit, determine and predict the orbit, receive and process the satellite telemetry data as required, monitor the working status of the satellite, send the telecontrol instructions and inject the telecontrol data as required, control and manage the satellite, and complete the Earth-satellite timing as required.

The ground TT&C network (as shown in Figure 6.17) consists of three basic parts, namely satellite TT&C center (SCC), multiple TT&C stations, and data communication system. The ground TT&C station is composed of a fixed TT&C station, a moveable TT&C station and an ocean-going instrumentation ship. The TT&C systems include uniform carrier TT&C system and spread-spectrum TT&C system. The functions of each part of China's TT&C network are shown in Table 6.8.

6.5.2.2 Design of the Interfaces between Large TT&C Systems

Due to the particularity of TT&C systems, there is an important interfacing relationship between the satellite TT&C system and the ground TT&C system. Because the satellite TT&C is realized through the cooperation between the two TT&C systems, the matching and compatibility between the satellite interface and the ground interface are very important. Therefore, the design of satellite TT&C system must rely on the conditions and capabilities of the existing ground TT&C system; otherwise, it will be a castle in the air.

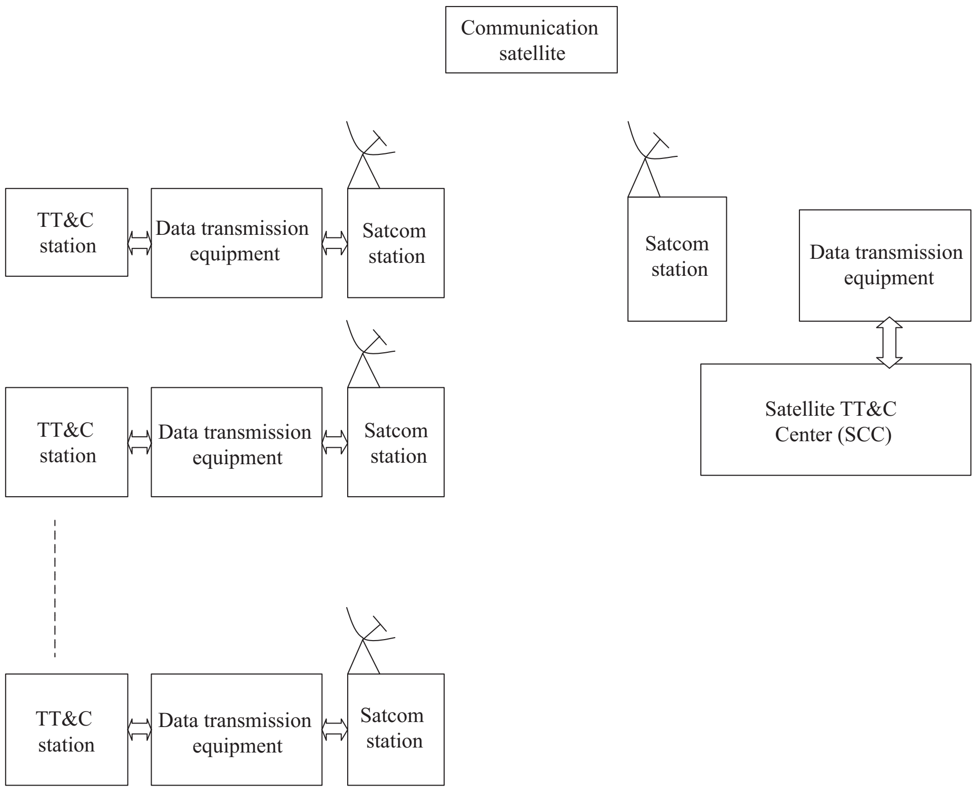

FIGURE 6.17 Basic composition of satellite-ground TT&C network.

Two kinds of interfaces are mainly found between the satellite TT&C system and the ground TT&C system, including RF channel interface and telecontrol & telemetry data interface. The design of satellite-ground TT&C interfaces should fully consider the layout of ground TT&C network, the telecontrol and telemetry capabilities (including the adaptability to subcarrier frequency, code rate, processing capability and data format), the capability of orbit tracking and measurement as well as other constraints, and define the mission requirements, TT&C regime and working mode, TT&C channel index requirements and telemetry/telecontrol interface requirements of satellite TT&C system and ground TT&C system. The main interface indexes should meet the requirements of the satellite-ground TT&C interface document. The correctness and matching of the interfaces should be verified through the experimental docking between the satellite TT&C system and the ground TT&C system.

The following main contents should be specified for the interface between the satellite TT&C system and the ground TT&C system:

1. **TT&C mission requirements:** clarify the mission requirements of satellite/ground TT&C systems in various stages from satellite launch preparation to long-term in-orbit operation.

TABLE 6.8 Functions of Each Part of China's TT&C Network

Name	Function	Task	Composition
SCC	Carry out the TT&C and management of near-Earth satellites/geosynchronous satellites	1. Automatically generate various flight plans for satellites; 2. Receive, record, process and display various telemetry information from multiple TT&C stations in real time; 3. Receive the orbit measurement information from multiple TT&C stations in real time, calculate the satellite's orbital parameters and forecast the orbit; 4. Calculate the orbit, attitude, rotation speed and return parameters of the satellite and make a control decision; send the decision telecommands to the satellite and calculate the injected data; 5. Remotely monitor the TT&C station.	Consisting of information processing system, monitoring and display system, front-end communication processor and communication system
Communication system	Ensure the data, voice and telegraph communications between SCC and TT&C stations (ships)/foreign satellite operation centers	1. Ensure the smooth and reliable command and scheduling of the TT&C network; 2. Realize the time synchronization of the TT&C network; 3. Transmit various TT&C and monitoring information in real time between the TT&C center and the TT&C station.	Consisting of satcom subsystem, ground-to-ground transmission subsystem, command and communication subsystem, timing subsystem and network management subsystem
TT&C station	Track, measure and control the near-Earth satellites/geosynchronous satellites, and receive and demodulate the telemetry signals	1. Track and measure the satellite orbit; 2. Receive and process the satellite telemetry data; 3. Send the telecommands and inject the telecontrol data as required.	Consisting of antenna, tracking and pointing subsystem, RF transceiving channel, baseband equipment, system monitoring console and time/frequency subsystem

2. **Satellite TT&C system, working mode and equipment composition:** determine the satellite/ground TT&C regime, TT&C frequency, working mode, equipment composition etc.

3. **RF channel interface requirements:** determine the uplink/downlink channel indexes for USB or spread-spectrum TT&C, the forward/backward channel indexes for relay TT&C and the performance indexes of equipment (TT&C transponder, antenna etc.).

4. **Requirements for satellite-ground telecontrol and telemetry data interfaces:** determine the telemetry channel (code type, code rate, modulation mode, bit error rate etc.), telemetry format protocol and data processing (protocol, processing method, spacecraft identification convention etc.), telecontrol channel (code type, code rate, modulation mode, bit error rate etc.), telecontrol format protocol (telecontrol channel description, instruction classification, remote frame format) and other interface requirements.
5. **Other special interface requirements:** satellite-ground TT&C encryption/ decryption (yes/no), encryption and decryption method, password type, requirements of satellite-ground time synchronization interface (onboard time, timing method, satellite-ground time synchronization accuracy etc.), special in-orbit application strategy etc.

The following main factors should be considered when designing the satellite-ground RF channel interface:

1. **Acquisition thresholds of satellite-borne and ground receivers:** determine the link budget according to the TT&C frequency, ERIPs of ground station and satellite, the maximum operating distance between spacecraft and ground TT&C station, satellite-borne and ground antenna gains, as well as a variety of channel losses. Calculate the minimum input levels at the input ends of satellite-borne and ground receivers. Determine the acquisition threshold of receivers after considering a certain margin. To prevent the EMI of other onboard equipment, the acquisition sensitivity of the receivers should not be too high.
2. The dynamic range of a receiver should meet the level changes at the input end of the receiver caused by various factors and have a margin of at least 3 dB.
   1. The changes in the maximum and minimum distances between the spacecraft and ground TT&C station caused by the change of spacecraft orbit, and finally the change of space loss.
   2. The gain change in the coverage area of TT&C antenna caused by the change of spacecraft attitude.
   3. The variation range of transmitter power.
3. The following two factors should be considered when determining the frequency acquisition range/rate and frequency tracking range/rate of satellite-borne and ground receivers:
   1. The maximum Doppler frequency shift and change rate determined by ground station position and spacecraft orbit.
   2. The receiver frequency shift caused by initial frequency inaccuracy, aging, temperature, power supply and other factors.

4. **G/T value:** confirm the G/T value requirements for satellite and ground reception according to the link budget results, and assign the corresponding indexes.

5. **EIRP value:** confirm the EIRP (Equivalent Isotropically Radiated Power) value requirements for satellite and ground transmission according to the link budget results, and assign the corresponding indexes.

The following main factors should be considered in the design of satellite-ground telecontrol and telemetry data interfaces:

1. The satellite-ground telecontrol and telemetry information rates should meet the requirements of satellite-ground link budget.

2. The satellite-ground operation process should be in accordance with the standard or agreed in advance.

3. The ground telecontrol mode, instruction frame format and uplink data block format shall be agreed between the satellite and the ground.

4. The downlink telemetry data format, encryption (yes/no), encryption mode and downlink data format shall be agreed between the satellite and the ground.

6.5.2.3 Verification of TT&C System Interfaces

To verify the correctness and matching of satellite-ground TT&C interfaces, a satellite-ground TT&C docking test is needed. The docking test is jointly completed by the overall TT&C system, satellite TT&C system and ground TT&C system, and is arranged according to the maturity of satellite TT&C system and the change of system state in the development process. The purpose and test items of TT&C system interface verification are shown in Table 6.9.

TABLE 6.9 Purpose and Test Items of TT&C System Interface Verification

No.	Purpose	Item
1.	Check the matching of satellite-ground joint efforts	TT&C channel docking; telemetry and telecontrol docking; simulated orbit-insertion flight
2.	Check the function and performance of TT&C systems	TT&C channel docking; telemetry and telecontrol docking
3.	Check the correctness and harmony of telecommand transmission and implementation;	Telemetry and telecontrol docking
4.	Check the correctness and matching of the downlink telemetry data for ground demodulation	Telemetry and telecontrol docking
5.	Check the time synchronization between satellite and ground	Timing function test
6.	Injection simulation exercise	Simulated orbit-insertion flight

6.5.3 Design and Verification of the Interface to Ground-Receiving System

6.5.3.1 Overview of Ground-Receiving System

The ground-receiving system is divided into satellite-ground microwave link receiving system and satellite-ground laser link receiving system, of which the former has been put into operation and the latter has been verified by satellite-ground experiments.

The main task of microwave ground-receiving system is to automatically track the satellite; receive, demodulate, decode and save the remote sensing data sent by the satellite; descramble, decrypt and decompress the demodulated data stream; and save and archive the received data. The main task of laser ground terminal is to cooperate with the onboard laser communication terminal to achieve the establishment, maintenance and communication of the satellite-ground laser communication link; send the uplink beacon light and modulation signal light as required; acquire and track the received downlink beacon light; and complete the photoelectric conversion, demodulation and data processing of downlink signal light.

The ground-receiving system in China consists of fixed ground-receiving stations and mobile stations. At present, there are fixed ground stations in Beijing, Kashgar, Sanya, Mudanjiang and Polar Regions, which can cover the whole territory of China and 70% of Asia. At present, the ground stations mainly use the X band. Because the X band is limited to 8025–8400 MHz by International Telecommunications Union, the upper limit of X-band data transmission capacity is restrained. By using the dual circular frequency polarization multiplexing technology, the 2*450 Mbps and 2*600 Mbps data-receiving capacities can be achieved at the X band. In order to improve the data transmission capacity, the ground stations are building a modulation system with a higher bandwidth availability ratio or choosing other bands, such as Ka, Ku and so on.

6.5.3.2 Design of Satellite-Ground Microwave Link Interface

Due to the particularity of data transmission system, there is an important interfacing relationship between the satellite data transmission system and the ground-receiving system. Because the satellite data transmission is realized through the cooperation between satellite/ground data transmission systems, the matching and compatibility between the satellite interface and the ground interface are very important. Therefore, the design of satellite data transmission system must rely on the conditions and capabilities of the existing ground-receiving system.

Two kinds of interfaces are mainly found between satellite/ground data transmission systems, including RF channel interface and baseband signal interface. The design of satellite-ground data transmission interfaces should fully consider the layout of ground data transmission stations, the data-processing capabilities (including subcarrier frequency, code rate, channel modulation characteristics, RF bandwidth, decompression and decryption), the capability of orbit tracking as well as other constraints, and define the mission requirements of satellite and ground-receiving system. The main interface indexes should meet the requirements of the satellite–ground-receiving interface document. The correctness and matching of the interfaces should be verified through the experimental docking between satellite/ground data transmission systems.

The following main contents should be specified for the interface between the satellite and the ground-receiving system:

1. **Data transmission mission requirements:** clarify the mission requirements of satellite and ground-receiving system in various stages from satellite launch preparation to long-term in-orbit operation.
2. **Satellite data transmission system, working mode and equipment composition:** determine the satellite-ground data transmission carrier frequency, channel modulation characteristics, working mode, equipment composition etc.
3. **RF channel interface requirements:** determine the number of data transmission channel configurations, channel performance indicators (carrier frequency, data rate, channel modulation characteristics, RF bandwidth, transmission frequency stability, transmission frequency accuracy, phase jitter, amplitude frequency characteristics, clutter suppression, harmonic suppression, satellite transmission EIRP, bit error rate etc.).
4. **Baseband signal-processing interface requirements:** define the interface requirements such as AOS format arrangement, scrambling requirement and channel coding requirement.
5. **Other special interface requirements:** satellite-ground data transmission encryption (yes/no), encryption method, password type, special in-orbit application strategy etc.

The following main factors should be considered when designing the satellite-ground RF channel interface:

1. **G/T value:** confirm the G/T value requirements for ground reception according to the link budget results, and assign the corresponding indexes.
2. **EIRP value:** confirm the EIRP value requirements for satellite transmission according to the link budget results, and assign the corresponding indexes.

The following main factors should be considered in the design of the satellite-ground baseband signal:

1. The satellite-ground data transmission rate should meet the requirements of satellite-ground link budget.
2. The satellite-ground operation process should be in accordance with the standard or agreed in advance.
3. The satellite-ground data transmission should be processed in accordance with the agreed data format arrangement, channel coding, data stream encryption, and data stream scrambling mode.

6.5.3.3 Verification of Satellite-Ground Microwave Link Interface

To verify the correctness and matching of satellite-ground data transmission interfaces, a satellite-ground data transmission docking test is needed. The docking test is jointly completed by the ground system and satellite data transmission system and is arranged according to the maturity of satellite data transmission system and the change of system state in the development process.

The main objectives of the satellite-ground data transmission docking test are:

1. Verify the consistency and correctness of the satellite and ground data transmission interfaces.
2. Verify the matching and compatibility of joint satellite and ground transmission efforts.
3. Check the functions and main technical indicators of the satellite and ground systems, including the correctness and matching of satellite/ground RF interfaces and baseband signal interfaces.

The main test items in the satellite-ground data transmission docking test include:

1. **Baseband data docking test:** dock the baseband data of data transmission system with the frame synchronizer, decryption device/descrambling device/decompressor and other equipment of ground station to verify the matching between ground data–processing devices and between them and satellite interfaces. The test contents mainly include data compression/decompression, interpretation/decryption, formatting and format interpretation, data recording/playback, plaintext/ciphertext switching etc.
2. **Wired channel docking test:** use all equipment (except antenna) of the data transmission system for limited docking to verify the matching of satellite/ground interfaces. The test contents mainly include the tests of carrier frequency, hybrid harmonic suppression capability, phase noise, spectral characteristics, transmission bit rate, system BER test, BER characteristic curve etc.
3. **Wireless channel docking test:** the docking is to establish a wireless link, and verify the matching of satellite/ground interfaces through the wireless docking between onboard data transmission system and ground station.

6.5.3.4 Design of Satellite-Ground Laser Link Interface

Because the satellite-ground laser link communication system belongs to precision opto-mechanical product, the satellite terminal has a strict interface relationship with the optical ground terminal. Otherwise, a minimal error or deviation may result in wide divergence. The completion of satellite-ground laser link test needs the cooperation between satellite system and ground system, so the matching and compatibility between satellite interface and ground interface cannot be ignored.

There are mainly two kinds of interfaces between satellite/ground laser link communication systems, namely optical interface and communication interface. The design of satellite-ground laser link communication interfaces should give full consideration to the optical parameter matching (including the compatibility of beam divergence angle, filter bandwidth, laser wavelength and line width) and communication signal matching (including the adaptability of code rate, processing ability and data format) of onboard terminal and optical ground terminal, and should define the task requirements, communication regime, working mode, index requirements and link interface requirements of onboard laser communication terminal and optical ground terminal. The main interface indexes should meet the requirements of the satellite-ground laser link interface document. The correctness and matching of the interfaces should be verified through the experimental outfield docking between satellite/ground laser link communication systems.

The following main contents shall be specified for the interface between the onboard terminal and optical ground terminal of satellite-ground laser link system:

1. **Mission requirements:** clarify the mission requirements of onboard terminal and optical ground terminal in the satellite operation period after the orbit insertion.
2. **Communication regime, working mode and equipment composition:** determine the communication regime, working mode and equipment composition of satellite-ground laser communication link.
3. **Link interface requirements:** define the optical interface (including beam divergence angle, filter bandwidth, laser wavelength, line width etc.) and communication interface (including code type, code rate, modulation mode, data format etc.).
4. **Other special interface requirements:** such as satellite-ground time synchronization interface requirements (onboard time, calibration mode etc.), special in-orbit application strategy etc.

The following main factors should be considered in the design of the satellite-ground laser link interface:

1. The wavelengths of the transmitting/receiving signal light and beacon light in the satellite-ground laser link shall be determined in accordance with the agreement.
2. The emitted light power and beam divergence angle of satellite-ground laser link should meet the requirements of satellite-ground link budget.
3. The operation process of satellite-ground laser link test should be in accordance with the standard or agreed in advance.
4. The working mode and data format of onboard terminal and optical ground terminal in the satellite-ground laser link should be determined as agreed.

6.5.3.5 Verification of Satellite-Ground Laser Link Interface

To verify the correctness and matching of satellite/ground laser link interfaces, the outfield docking test of satellite/ground laser links should be carried out during the ground development stage. The outfield docking test of satellite-ground laser link is completed jointly by onboard laser communication terminal and optical ground terminal. To avoid the possible environmental pollution of onboard laser communication terminal in the outfield environment, the qualification product of onboard laser terminal is used instead in the outfield docking test. This test is arranged according to the maturity of satellite-ground laser communication system and the change of system state in the development process.

For satellite-ground laser link system, the main objectives of its outfield docking test are as follows:

1. Verify the matching of onboard laser communication terminal and optical ground terminal in the joint operation.
2. Check the functions and performance of onboard laser communication terminal, including aiming, capturing, tracking and communication.
3. Verify the working mode of satellite-ground laser link and the working program of optical ground terminal.
4. Check the influence of atmospheric turbulence on the system function and performance through the outfield test, and correct the compensation effect of optical ground terminal.

The outfield docking test items of satellite-ground laser link system mainly include the following:

1. Verify the interfaces of satellite-ground laser link system.
2. Verify the targeting, acquisition and tracking functions of the system.
3. Verify the communication and data-processing functions of the system.
4. Verify the atmospheric compensation capability of optical ground terminal in the system.
5. Verify the work flow of this system.

6.5.4 Design and Verification of the Interface to Launch Site

6.5.4.1 Overview of Launch Site System

Launch site is a place for launching a carrier rocket with spacecraft and is also a place for their pre-launch final assembly, testing, refueling and other activities. The launch site system provides the site, electricity, gas, communication, environment, transportation, lifting, safety and other conditions for pre-launch final assembly and testing, provides weather forecast, telemetry and safety control for the launch, and carries out the organization, command, planning and coordination of the launch mission.

The telemetry and safety control system of the launch site is mainly responsible for the tracking, measurement, telemetry and safety control of the first, second and third rocket stages in the flight phase. Generally, the launch site has no direct interface to the satellite.

The selection of a launch site is generally considered along with the selection of launch vehicle. It should consider not only the launch site's ability to support the launch of the rocket and satellite but also the safety of launching trajectory, flight area as well as the ground area that receives the falling rocket debris.

China has built four launch sites.

1. Jiuquan Space Launch Center (42°N, 101°E), China's first launch site. As a launch site for manned spacecrafts, it can also launch other satellites in different orbits.
2. Xichang Space Launch Center (28°N, 102°E), mainly for launching the satellites into geosynchronous or low-inclination orbits.
3. Taiyuan Space Launch Center (39.3°N, 112°E), mainly for launching the satellites in polar orbit and sun-synchronous orbit.
4. Wenchang Space Launch Center (19°N, 111°E), mainly for launching geosynchronous orbit satellites, massive polar-orbit satellites, large-tonnage space stations and deep space exploration satellites.

6.5.4.2 Interface between Satellite and Launch Site

A launch site generally consists of a technical area (a building for testing the rockets and satellites), a launch area, a command and control center, a tracking and measurement system, a communication system, as well as a propellant refueling system and a weather forecast system.

In the technical area, the satellite will be transshipped, tested, assembled and stored. In the launch area, it will be tested on a launch tower to conduct the satellite-rocket joint tests (such as EMC test) and the joint general inspection of large systems. The typical working process on a launch site is shown in Figure 6.18.

A satellite can be docked with a launch vehicle in one of the following three types of opportunities and transfer modes:

1. The satellite is transferred separately and docked with the launch vehicle on the launching pad.

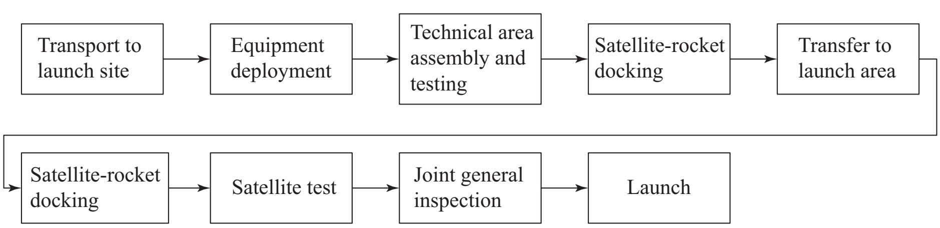

FIGURE 6.18 Typical working process on a launch site.

2. The satellite and the launch vehicle are docked in the technical area and then transferred to the launch area as a whole.
3. The satellite is docked with part of the launch vehicle in the technical area (for example, dock with the third-stage instrument module and covered with fairing), and then the combination is transferred to the launch area to dock with the rest part of the launch vehicle.

Launch site stage is the key stage of a satellite development project, and is the final stage before launch. Every act and move at this stage will affect the work on all large systems in the whole project. The working process at the launch site must be determined in advance, and various interfaces (for site connection, power supply, communication, refueling and so on) between the satellite and the launch site must be coordinated.

The chambers available for satellite testing include transfer chamber, assembly test hall, system test chamber, unit test chamber, gas distribution chamber, product storage chamber and so on. Each chamber must meet the requirements for satellite environment, power supply, gas supply and communication, as well as the testing requirements (such as satellite transportation channel and operation space). In addition, special rooms should be allocated for EED storage and testing, battery storage and other special requirements.

The satellite test power should be separated from power grid and should be provided by an uninterruptible power supply. The type, quality and power of power supply should meet the requirements of satellite testing. Special high-power devices, such as solar wing lighting, should be powered separately. The ground wires are classified into signal ground wires, protective ground wires, neutral ground wires, antistatic ground wires, and equipotential ground wires. The signal ground wire of the satellite system should be separated from the signal ground wires of other systems.

The facilities for propellant storage and filling include special water firefighting system, explosion-proof plug, shower and eye bath. According to the satellite requirements, the launch site can provide nitrogen, high-purity nitrogen and anhydrous hydrazine.

After the satellite is transferred to the launch area, remote testing is generally pursued. Only the satellite power supply and the front end of test system are moved to the launch area (specifically, under or on the launch tower), and are connected to the satellite by umbilical cables. The environment, power supply and gas supply of the launch area shall meet the satellite requirements. In the remote testing, the satellite-ground wireless signal can communicate with the technical area through wireless frequency-conversion relay or optical fiber relay.

6.5.4.3 Experimental Verification

When coordinating the interface between the satellite and the launch site, the relevant personnel should comb the launching site process and define the work items, division of labor, condition guarantee requirements and other items in the whole working process of the launching site. If necessary, the satellite-rocket joint exercises shall be carried out on the launching site, and the interface matching, system coordination and comprehensive exercises shall be conducted for the satellite, rocket and ground equipment. According to the purpose of joint exercises, the joint exercises on the launch site can be done at the system level or subsystem level.

When the satellite is on the launch site, the whole site and equipment to be used shall be jointly inspected by the satellite designer and the launch site operator to confirm good environmental condition and technical safety. The inspection certificates are also needed for the anhydrous hydrazine, nitrogen and other materials provided by the launch site. After the satellite transfer, several joint general inspections, including electrical logging, EMC test and launch day drill, will be conducted by the three parties (satellite, rocket and launching site).

CHAPTER 7

Design of Spacecraft Configuration and Assembly

Zu Jiaguo, Xiao Pengfei, Shi Haitao,
Zhang Xiaofeng, and Hao Ganggang

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CONTENTS

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The design of spacecraft configuration, layout and assembly is an important part of spacecraft system design, also known as the overall mechanical design of a spacecraft. The configuration and layout design is the first step for a spacecraft to develop from theoretical and logical design to product design. It is the basis of mission implementation and directly relates to the feasibility of spacecraft mission implementation. Layout design is to complete the layout of equipment and components on the spacecraft through the comprehensive consideration of multiple factors and requirements on the basis of configuration design, so as to ensure the completion of spacecraft system mission. The general assembly design is to further refine the configuration and layout design and transform the relevant design into the detailed requirements for structural design and spacecraft AIT (assembly, integration and test) implementation, and finally complete the cabin assembly and system-level assembly. It is a system-level engineering implementation.

In this chapter, the design criteria, design contents and analysis verification methods for the spacecraft configuration, layout and assembly are systematically combed and specifically illustrated.

7.1 DESIGN CRITERIA AND CONTENTS OF SPACECRAFT CONFIGURATION

7.1.1 Design Criteria

See Figure 7.1.

7.1.1.1 Meet the Mission Requirements

7.1.1.1.1 Orbital Requirements (e.g. Special Space Environment, Light Pressure, Atmospheric Resistance) Among different mission orbits, some orbit environments have certain particularities. To analyze the influence of these particularities on spacecraft configuration, we should first consider the orientation of main payloads (pointing to sub-satellite point, the Earth, other spacecrafts or other celestial bodies such as the sun) in different orbits. Second, we should arrange the equipment in consideration of the external heat flow in different orbits. Some equipment is sensitive to high temperature, while the others are sensitive to low temperature. Third, we should consider the constraints (such as the center of mass and inertia) related to spacecraft control and stability, which also include the influence of low-orbit atmospheric resistance on a mission, the possibility of reducing the thrust faces of a configuration and the aerodynamic shape required for the return mission.

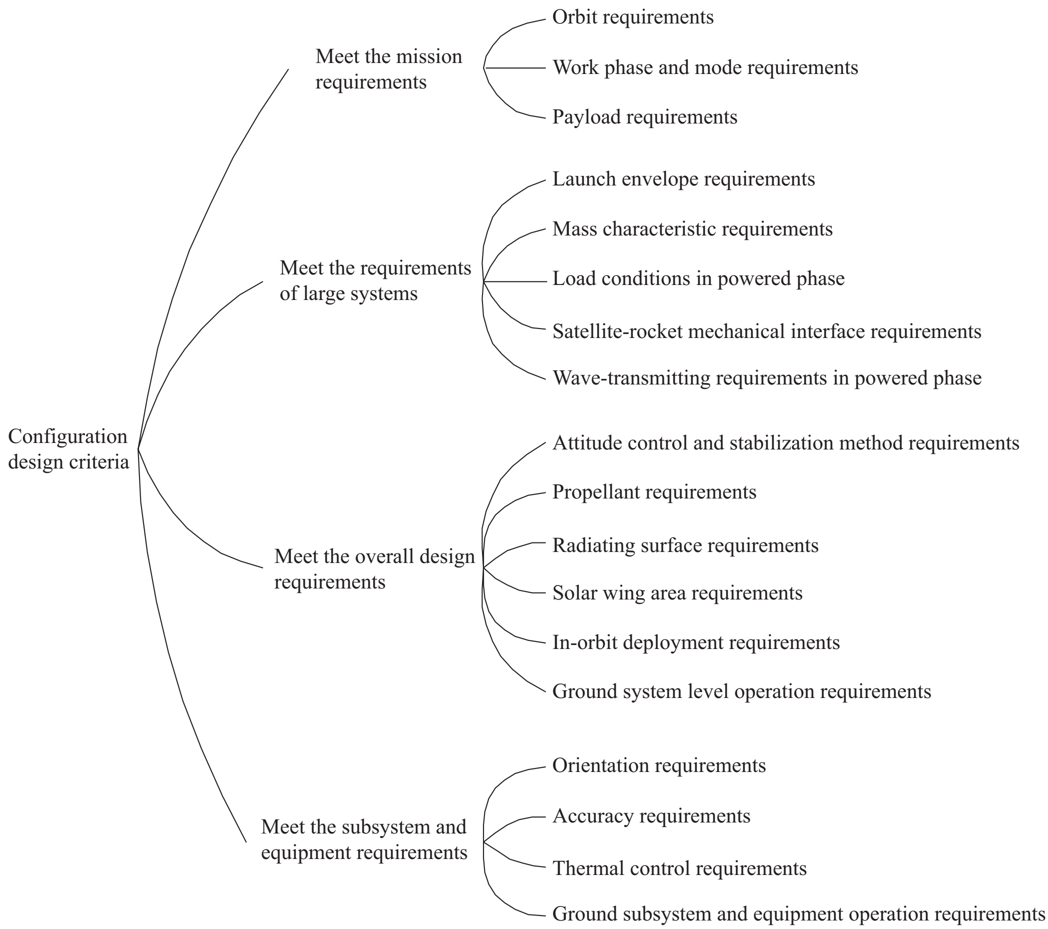

FIGURE 7.1 Classification of configuration design criteria.

7.1.1.1.2 Requirements of Development Phase and In-orbit Operating Mode The spacecraft will be affected by different environments in various phases from development, test, launch to orbiting, due to a series of actions and processes such as ground transportation, launch, spacecraft-rocket separation, spacecraft docking, deployment of antennas and mechanical arms, attitude adjustment and orbital transfer, and changing the load point orientation. These processes and special requirements decide whether to divide a spacecraft into several function modules, for example, resource module (mainly including energy and propulsion equipment), payload module (mainly including payloads and data transmission equipment) and reentry module (mainly including astronauts, lunar materials and other equipment that needs to return).

7.1.1.2 Meet the Requirements of Large-System Interfaces

7.1.1.2.1 Requirements of Rocket Envelope The envelopes include a static envelope and a dynamic envelope. The static envelope refers to the available space that the rocket provides to the spacecraft. The dynamic envelope refers to the influence of spacecraft-rocket separation, clamp band unlocking, spacecraft-rocket docking and other motions on the spacecraft, which should be considered in the spacecraft configuration design.

7.1.1.2.2 Rocket Load Conditions Including the rocket's requirements for spacecraft fundamental frequency and centroid, center of mass of the spacecraft, the design load of spacecraft structure, the load conditions of spacecraft vibration test and noise test, and the spacecraft-rocket coupling analysis.

7.1.1.2.3 Spacecraft-Rocket Mechanical Interface Requirements Including the coordination of interface type, docking orientation, interface size and connection mode between the satellite and the rocket, as well as the procedure and mode of spacecraft-rocket separation.

7.1.1.2.4 Coordination of Wave-Transmitting Window between Spacecraft and Rocket The position and microwave transmittance of wave-transmitting fairing port should consider the feasibility of wireless spacecraft Telemetry, Tracking, and Command (TT&C) on the launch tower during the large-scale closure or opening, and the feasibility of wireless spacecraft TT&C in the powered phase through the wave-transmitting window in the case of multi-satellite launch.

7.1.1.3 Meet the Subsystem and Equipment Requirements

7.1.1.3.1 Mass Characteristics The mass characteristics of a spacecraft include center of mass, mass and moment of inertia. The requirements for mass characteristics mainly come from the attitude control subsystems of both the rocket and the spacecraft, the latter of which has more and higher requirements. Those requirements, in turn, determine the spacecraft shape as well as the way of equipment distribution. It is important to note that the mass characteristic requirements vary with the mission profile. Therefore, the analysis of mass characteristics should be carried out according to the mission profile.

7.1.1.3.2 Operating Requirements for Large Deployable Components The installation, fixing, unlocking, separation and deployment of deployable and movable components (such as solar wings, cameras, antennas and pull rods) – all require certain space. The configuration and layout design should not only accommodate the original installation space of these components but also check whether the dynamic envelope of the components under deployment will interfere with the primary structure or other equipment.

7.1.1.3.3 Orientation Requirements The orientation requirements include the requirements for thrust vector, equipment field of view (FOV) and antenna beam, which mainly affect the configuration of the equipment on spacecraft surface. In principle, there should be no obstacles within the thrust vector, equipment FOV and antenna beam, and no reflected light, thermal radiation and other influences in the FOV of antennas, attitude sensor and remote sensor.

7.1.1.3.4 Accuracy Requirements (Position, Angle) The accuracy requirements mainly affect the layout, and sometimes also affect the configuration or local structure. According to the normal work flow, the equipment with accuracy requirements should be installed in the place with good structural stiffness, accuracy measurement space and AIT stability. If such a place does not exist according to the AIT state analysis, inverse requirements (e.g. increasing the load-bearing points of the primary structure) should be proposed for the structure to strengthen local stiffness or change the configuration.

7.1.1.3.5 Thermal Control Requirements (Such as the Heat Balance Requirements of Equipment) According to the power consumption of onboard instruments, the spaceborne heat dissipation channels and their positions can be determined. Local overcooling or overheating should be avoided, and local thermal control measures should be taken when necessary.

7.1.1.4 Meet the Maintainability and Operational Accessibility Requirements

These requirements are mainly considered from two dimensions, including the requirements of the operation space itself and the environment, site and state of the operation. The types of operations include large-parts docking, equipment installation, cable plugging and unplugging, paving, pipeline welding, photography, fine-metering light path, various measurements and tests, docking with ground equipment, pre-launch state setting etc.

7.1.2 Contents of Configuration Design

See Figure 7.2.

7.1.2.1 Design of Flight Attitude and Orientation

The flight attitude of a spacecraft is described in a pre-defined coordinate system. Typical flight attitudes include orientation to the Earth, orientation to the sun, orientation to the target celestial body and inertial orientation. The specific requirements are as follows:

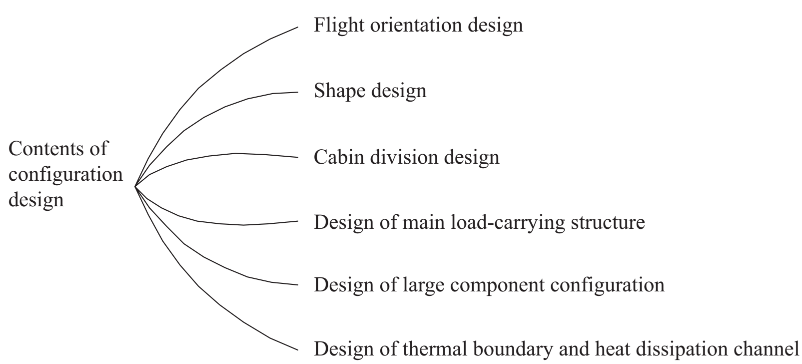

FIGURE 7.2 Contents of configuration design.

1. **Earth-oriented remote sensing mission:** to meet the Earth orientation requirement of remote sensing payload or the orientation requirement of other observation targets.
2. **Navigation mission:** to meet the Earth orientation requirement of navigation payload antenna.
3. **Reentry return mission:** to meet the requirement of reentry trim AOA (angle of attack) in the pitching direction and the lift control requirement (half ballistic) in the rolling direction, and to ensure that the ballistic trim AOA is basically zero.
4. **Rendezvous and docking mission:** to ensure that the docking axes (mechanisms) point to each other when the active and passive spacecrafts approach to each other.
5. **Deep space exploration mission:** to meet the orientation requirements of cruise flight phase, orbital transfer phase and target-body detection phase, the orientation requirements of landing navigation payload, and the orientation requirements of roving observation payload.
6. **Space science exploration mission:** to meet the specific orientation requirements of space exploration payload.

7.1.2.2 Shape Design

7.1.2.2.1 Shape Design of a Reentry Spacecraft The shape design of a reentry spacecraft must meet the special requirements of aerodynamic characteristics. Due to high reentry speed and obvious start-up heating, the shape design of a reentry spacecraft whose reentry environment is filled with atmosphere (the Earth, Mars etc.) usually takes into account the influence of heat protection and aerodynamic force.

1. **For a reentry spacecraft returning to the Earth by ballistic reentry:** the reentry capsule should have a good aerodynamic shape to meet the overload requirements of the equipment in the reentry capsule, minimize the aerodynamic heat and ensure a sufficient static stability margin. Figure 7.3 shows the shape of a Chinese reentry satellite.
2. **For a reentry spacecraft returning to the Earth by semi-ballistic reentry:** during the shape design, an appropriate shape and center-of-mass offset position should be selected in accordance with the range and overload constraints to ensure a certain lift-to-drag ratio. Meanwhile, the spacecraft shape should ensure the aerodynamic heating of the whole spacecraft and adjust the heat flux distribution reasonably. In addition, the spacecraft should have good static and dynamic stability to ensure that a proper trim AOA is maintained in the returning process. The return of the reentry capsule of a manned spacecraft is generally realized in a semi-ballistic way to obtain a small overload factor and ensure the safety of astronauts.

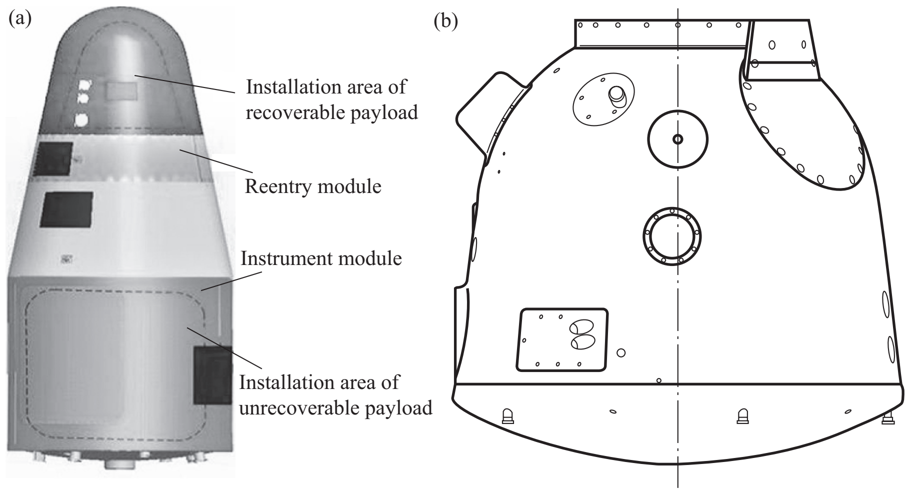

FIGURE 7.3 Outside view of return-type spacecrafts. (a) Ballistic return satellite. (b) The reentry module of Shenzhou spacecraft.

3. **For a reentry spacecraft returning to the atmosphere at high speed:** the aerodynamic heating and overload environment is more severe during reentry. The spacecraft shape design needs to have better aerodynamic and thermal characteristics and minimize aerodynamic overload, aerodynamic heating and heat flux.
4. **For a spacecraft entering an extraterrestrial body:** an appropriate shape that meets a certain ballistic coefficient should be selected according to the atmospheric-model characteristics of the target body and the requirements of spacecraft size. It is necessary to ensure that the spacecraft is able to effectively use the atmosphere to achieve aerodynamic deceleration, so as to meet the requirements of parachute deployment/landing.

7.1.2.2.2 Shape Design of a Planet-Landing Spacecraft For a planet-landing mission, the spacecraft shape design mainly considers the landing mode requirements, including the requirements for landing leg configuration and the special requirements for landing payload. The landing mission focuses on the requirements for landing buffer stability, impact load and landing attitude. Based on the above consideration, the configuration of landing leg is mainly to determine the ratio of the span to the longitudinal center of mass of the spacecraft in order to maintain the landing stability, meet the attitude requirements and ensure the resistance of onboard equipment to mechanical environment. A certain ratio of the folded size to the deployed size shall be maintained to adapt to the installation position in the main instrument module and the maximum rocket envelope. The expansion of the buffer mechanism should ensure that the bottom protrusion will not interfere with the landing surface in various attitudes and should leave a certain margin. Figure 7.4 shows the shape of the Chang 'e-3 (CE-3) lander.

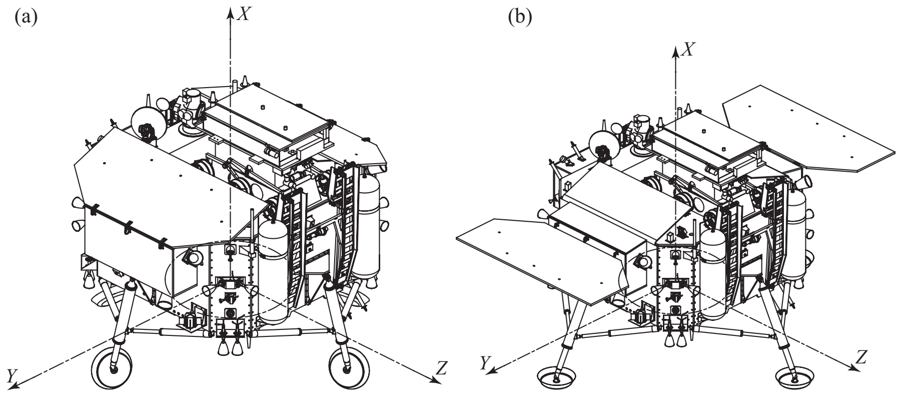

FIGURE 7.4 Outside view of CE-3 lander. (a) Folded. (b) Deployed.

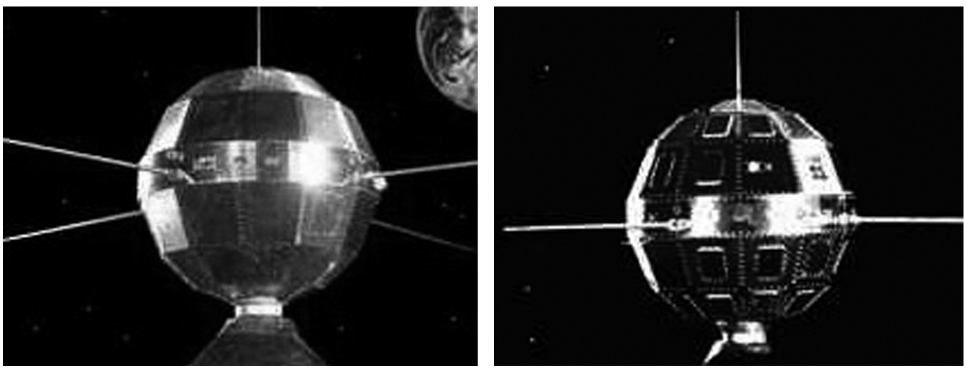

FIGURE 7.5 Single-spin stabilized spacecrafts. (a) Dongfanghong-1. (b) Shijian-1.

7.1.2.2.3 Spacecraft Shape Design Based on the Control and Stabilization Requirements

1. The single-spin stabilized spacecraft under simple single-spin stabilization control can be selected as a sphere or polyhedral sphere. In order to ensure the spacecraft attitude stability, the ratio of the spacecraft rotation inertia around the spin axis to that around any horizontal axis should be greater than 1. China's first man-made Earth satellite Dongfanghong-1 and China's first space exploration and technology experiment satellite Shiji-1 are spherical 72-hedral single-spin stabilized spacecrafts (as shown in Figure 7.5).

2. The double-spin stabilized spacecraft has a strict requirement for dynamic balance, generally using passive nutation damping. Its shape can be a short thick cylinder. The ratio of its rotational inertia around the spin axis to that around any horizontal axis should be greater than 1. The double-spin stabilized spacecraft with the ratio of the rotational inertia around the spin axis to that around any horizontal axis less than 1 should have a slender cylinder shape with active nutation damping. The Dongfong-2 satellite shown in Figure 7.6 is just a double-spin stabilized spacecraft.

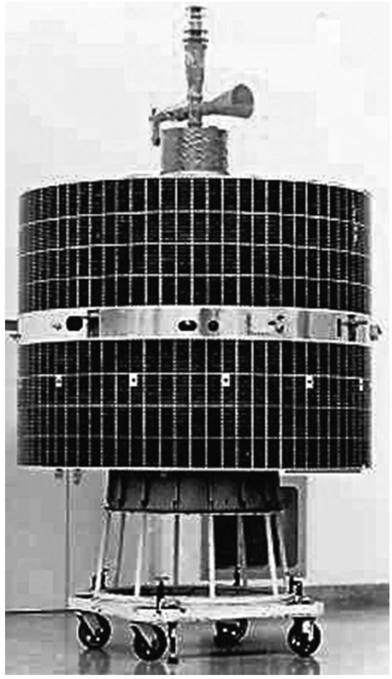

FIGURE 7.6 Double-spin stabilized spacecraft (Dongfong-2).

3. To obtain enough control torque, the gravity-gradient stabilized spacecraft generally has a gravity gradient pole with a certain mass on its top. The pole length should be more than three times the spacecraft height to ensure that the vertical and horizontal inertia ratio of the spacecraft in orbit is far less than 1. This spacecraft is generally provided with body-mounted solar arrays.

4. There is no special requirement for the shape of a triaxial stabilized spacecraft, which is usually a regular box or a multi-faceted prism. GSO (Geostationary Otbit) communication satellites and meteorological satellites are mostly cuboids, while large remote sensing satellites often adapt their shapes to the payload shapes. Figure 7.7 shows the shape of Gaofen 4, the first GSO remote sensing satellite in China.

7.1.2.3 Cabin Division Design

For a spacecraft with cabin separation requirement during the in-orbit flight, its configuration design should consider the design of cabin connection and separation. The in-orbit cabin separation is generally realized by some release and separation mechanisms. The typical release and separation mechanisms include the release and separation mechanism of clamp band spring, and the pyrotechnic connection, release and separation devices such as explosive bolts and pyrotechnic lock.

Generally, a large and complex spacecraft is divided into several modules to facilitate the common use of the platform and the parallel development of the spacecraft (final assembly, testing etc.). Therefore, the connection and separation design between modules needs to be considered and is generally realized by conventional connection fasteners that can be disassembled and assembled for several times.

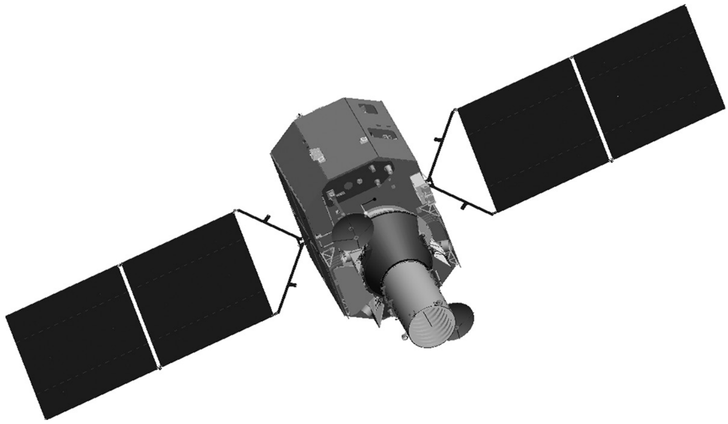

FIGURE 7.7 Gaofen-4 triaxial stabilized spacecraft.

7.1.2.4 Design of Main Load-Bearing Structure

When being launched, a spacecraft needs to undergo the mechanical environment during the powered phase of the launch vehicle. Therefore, the design of main load-bearing structure is generally required for a spacecraft to meet the requirements of powered-phase mechanical environment.

The typical load-bearing structures of spacecrafts can be divided into five types: bearing cylinder type, box plate type, truss type, shell type and mixed type.

7.1.2.4.1 Bearing Cylinder Type With cylindrical shell as the main bearing structure, this type of bearing structure has good strength and stiffness against torsion, bending and shear and good load transfer. A propellant tank with large mass can be easily installed in the cylinder. The bearing cylinder can be matched with the circular docking structure of launch vehicle easily. It is widely used by GSO spacecrafts (such as navigation satellites, communication satellites and meteorological satellites) and large- and medium-sized low-orbit remote sensing satellites, such as resource satellite platform. To reduce weight, modern bearing cylinders are made of carbon fiber composites. Their structures can be corrugated type, truss skin type or honeycomb sandwich type. The truss skin structures can be divided into aluminum alloy truss skin structure and carbon fiber truss skin structure according to their materials. Figure 7.8 is the exploded view of Spacebus4000 satellite platform designed with a bearing cylinder as main load-bearing structure.

7.1.2.4.2 Box Plate Type Due to the continuous payload increase and the limitation of rocket fairing diameter, the satellite payload can only develop to the height direction so that the disadvantage of central bearing cylinder is exposed. In comparison, the box-plate bearing structure is a box with a certain space shape composed of structural panels. As the main load-bearing structure of a spacecraft, it can provide a better installation surface for onboard equipment and can be modularized easily. Its disadvantage is the poor performance of the concentrated load-bearing force.

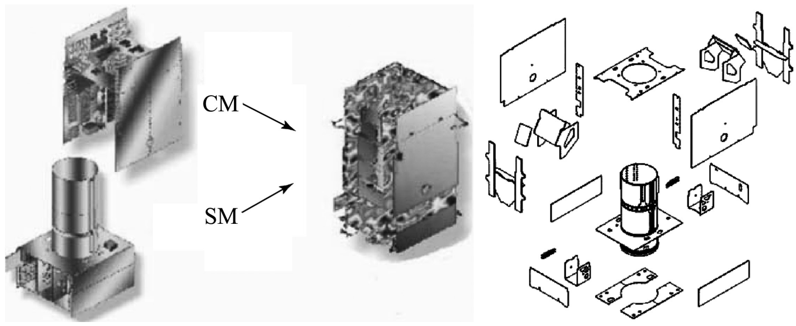

FIGURE 7.8 Spacebus4000 platform structure with central bearing cylinder.

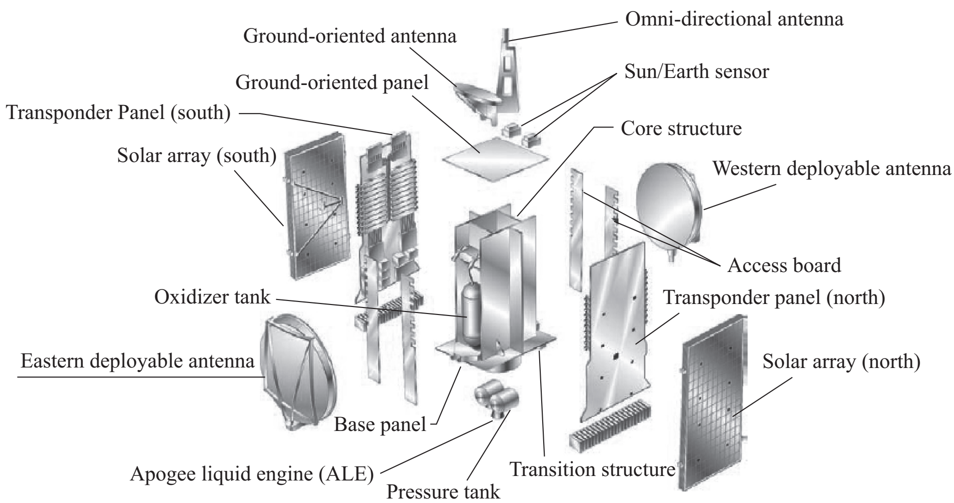

FIGURE 7.9 A2100AX platform with box-plate bearing structure.

The A2100 communications satellite of Lockheed Martin uses the box-plate structure (as shown in Figure 7.9). The connection between the platform cabin and the socket is a structural component that can ensure that the spacecraft-rocket docking ring is evenly stressed. The payload module consists of northern and southern panels and ground-oriented panels. The eastern and western panels are installed after the common platform and payload are connected. The batteries are installed on the bottom plate of the common platform for easy assembly/disassembly and thermal control. All panels are made of the carbon fiber composites with aluminum honeycomb sandwiches.

7.1.2.4.3 Truss Type The truss structure is a main load-bearing structure composed of truss members. The typical representatives of truss structure are the BSS-601 platform and GEM platform made by Boeing Company. The main advantages of truss structure are relatively light weight, direct force transfer, easy implementation of a large span of

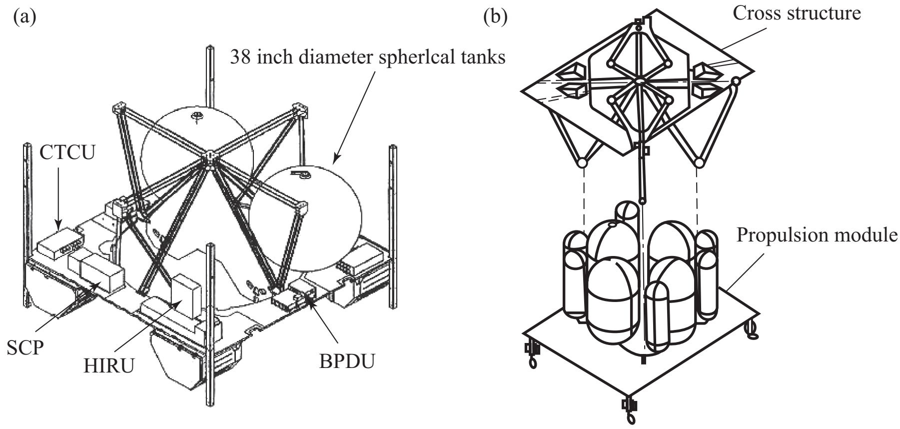

FIGURE 7.10 Truss platform structures. (a): BSS-601 platform. (b) GEM platform.

main structural body, good spatial environment stability and good structural openness that facilitates the transfer of concentrated load. Its disadvantages are the complex forces and installation of joints as well as the dynamic characteristics inferior to the bearing cylinder. The schematic diagram of BSS-601 platform structure is shown in Figure 7.10. The satellite platform is unlocked by clamp band, and four tanks are placed on the docking ring side by side. The cross-shaped structure is the primary structure carried by the platform, consisting of an east-west cross plate and two north-south vertical shear plates. The cross plate and the vertical shear plates are all sandwich panels, using high-modulus graphite fiber composites as face panels and aluminum honeycombs as the core. According to the load-bearing requirement, a composite reinforcing sheet is added into the cross plate, and oblique rods are added beside the vertical shear plates. The lower end of aluminum-alloy thrust cylinder is connected with the carrier rocket through the clamp band, and its upper end is connected with the above cross structure. The four corner columns are made of aluminum alloy. The bottom plate of the platform is an aluminum honeycomb sandwich panel with embedded heat pipes. It is connected with the thrust cylinder and supported by brackets on the four corner columns. The eastern and western panels are the aluminum honeycomb sandwich panels made of high-modulus carbon fiber composites. The tanks are supported by the trusses composed of high-modulus carbon fiber composites.

7.1.2.4.4 Shell Type The shell-type structure uses shell as the main load-bearing component and is generally used for a recoverable spacecraft. Because the spacecraft is exposed to aerodynamic force and aerodynamic heat during the rocket launch and the reentry into atmosphere, the shell itself should have high strength and stiffness. The shell is the main load-bearing component of the whole recoverable spacecraft. Moreover, it is directly connected with the launch vehicle, without the need for a special spacecraft-rocket adapter. As the shell has a large diameter, a cross beam (such as a cruciform or checked beam) shall also be designed to transfer loads from the instrument and equipment to the shell. Figure 7.11 is the sketch of a recoverable satellite.

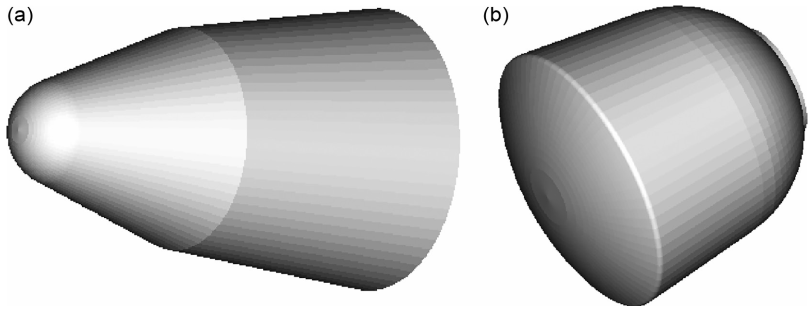

FIGURE 7.11 Shell-type structure. (a) ESA CTV. (b) Russia's Soyuz capsule.

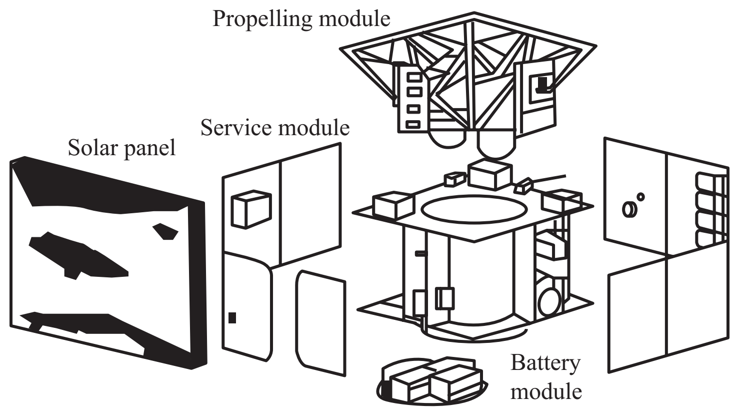

FIGURE 7.12 Exploded view of SPOT (Systeme Probatoire d'Observation de la Terre) satellite platform.

7.1.2.4.5 Mixed Type According to the requirements of payload configuration, the two or more structures mentioned above (for example, bearing cylinder + truss, bearing cylinder + box plates and box plates + truss) can be combined as the main load-bearing structure of a spacecraft. Figure 7.12 shows a platform structure with a central bearing cylinder plus trussing, and Figure 7.13 shows a platform structure with a bearing cylinder plus box plates.

7.1.2.5 Configuration Design of Large Components

The configuration design should meet the support and precision maintenance requirements of large payloads and the compaction/release requirements of large antennas. The overall dimensions of large components in the launching state should be confined to the net envelope space of the rocket fairing. If some components are beyond the envelope, the envelope requirement can be met by enabling the component folding through a mechanism during launch and the component deployment after entering the orbit.

The configuration design requirements of typical large components are as follows:

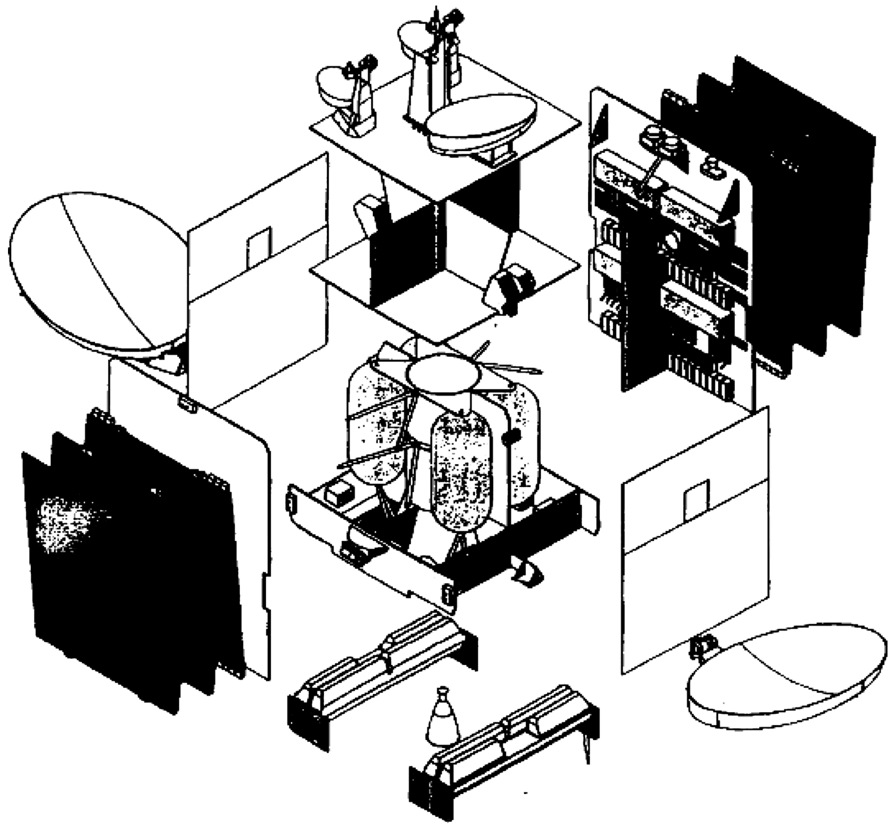

FIGURE 7.13 Exploded view of Eruostar3000 satellite platform.

7.1.2.5.1 Design Requirements of Solar Wing Configuration The configuration design of a solar wing should first determine its required area according to its functional requirements (i.e., output power), and then decide the number of base plates considering the size of solar wing. Its configuration design requirements are as follows:

1. According to the analysis of orbital light conditions and spacecraft flight direction, the orientation of solar arrays is selected.
2. The solar wing area is determined according to the orbital light conditions and solar array layout.
3. According to the spacecraft mission requirements and the requirement of solar wing area, the working mode of solar wings is selected and the configuration design of solar wings is completed (fixed type or deployable type, deployment method, single-axis rotation or double-axis rotation and the size and number of solar wing base-plates etc.).

FIGURE 7.14 Solar wing deployment configuration (a: series type, b: parallel type).

4. When a solar wing is folded, its maximum external profile (mainly the total height of the folded solar wing) should be within the dynamic envelope of the rocket fairing.

For a spacecraft in sun-synchronous orbit, the layout of solar wings can be determined according to the local time at the orbital descending node. In a noon-day orbit, the solar wings can be perpendicular to the orbital plane. In a 10:00 orbit at the descending node, the solar wings can be tilted by 30° so that their normal will point to the sun.

For the satellites with ordinary agility, solar wings are often deployed in series connection. However, for the satellites with high agility, solar wings are generally deployed in parallel connection in order to reduce the moment of inertia of the whole spacecraft. In Figure 7.14, series solar wings are given on the left and parallel solar wings on the right.

7.1.2.5.2 Design Requirements for the Configuration of Large Deployable Components

1. According to the mission requirements and functional requirements, those components should be arranged in an appropriate position on the spacecraft and should meet the requirements of orientation and FOV.
2. According to the envelope constraints of the rocket fairing, the dimensional requirements are put forward for the large components that are either deployed or folded.

7.1.2.6 Design of Thermal Boundary and Heat Dissipation Channel

The thermal control measures (such as heat dissipation surface) depend on the design of the whole spacecraft configuration, which, in turn, needs to consider the design of thermal boundary and main heat dissipation channel.

The onboard heat dissipation channels are mainly of the following types:

1. Direct heat dissipation through heat conduction.
2. Indirect heat dissipation through thermal radiation.
3. Heat dissipation over long distance or equipment is thermality, which is achieved through heat pipe, loop heat pipe and fluid loop system.

According to the condition of heat dissipation channel, the configuration design should consider the following aspects:

1. According to the conditions such as spacecraft orbit, descending node time and flight attitude, the configuration design should consider the selection of main heat dissipation channel.
2. In the case of direct heat dissipation by heat conduction, the configuration design should consider the thermal conductivity of materials and structures as well as the characteristics of structural surface. In case of indirect heat dissipation by thermal radiation, the configuration and equipment layout should consider the avoidance of shielding and should use the structural surface materials that can enhance radiant heat exchange. For the arrangement of heat pipe, loop heat pipe and fluid loop system, the configuration design should consider the installation space and strength of the structures.
3. For large payloads with high thermal stability requirement and other special equipment, consideration should be given to the inverse requirements of their boundary conditions on configuration.
4. For some spacecrafts with large heat dissipation demand, large heat dissipation surfaces or radiant heat exchangers (such as annular radiator, expandable radiator) should be set.

7.2 LAYOUT DESIGN CRITERIA AND DESIGN CONTENTS

7.2.1 Design Criteria

See Figure 7.15.

7.2.1.1 Meet the Mission's Requirements for Mass Characteristics

The equipment layout should meet the requirements for spacecraft mass characteristics to achieve flight control in the launch, orbit insertion, orbital transfer, in-orbit normal operation and other phases. The design value of inertia product of a spin-stabilized spacecraft around its spin axis should be zero. The layout of reentry capsule should ensure that its center of mass is in front of the pressure center by a certain margin. The mass characteristics of a GSO spacecraft should ensure that the solar pressure center coincides with the spacecraft center of mass as much as possible. Large-mass components should be arranged as close to the longitudinal spacecraft axis as possible. For the components with mass consumption (such as tanks and cylinders), they should be arranged symmetrically along the horizontal axis of the spacecraft or their center of mass should pass the longitudinal axis. The instrument layout should minimize the deviation of the center of mass of the spacecraft from its longitudinal axis.

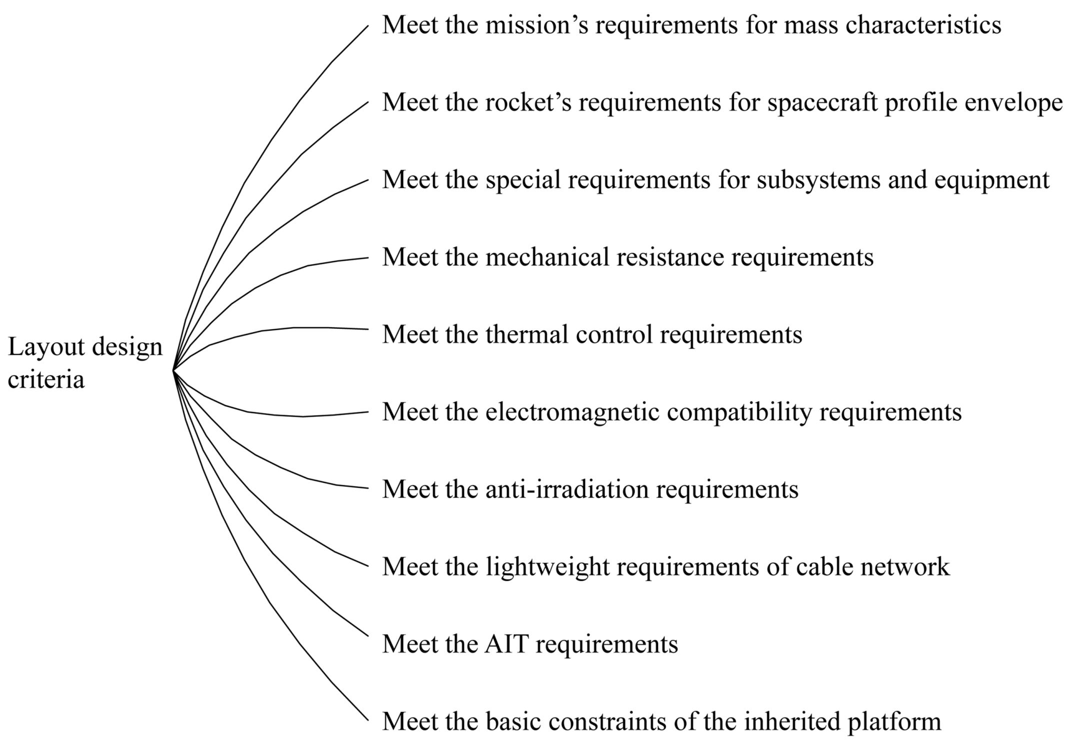

FIGURE 7.15 Classification of layout design criteria.

7.2.1.2 Meet the Rocket's Requirements for Spacecraft Profile Envelope

The layout of onboard equipment should meet the rocket's requirements for mechanical envelopes, mainly including static envelope and dynamic envelope.

7.2.1.3 Meet the Special Requirements for Subsystems and Equipment

The special requirements for subsystems and equipment mainly include the requirements for orientation, FOV, accuracy and the connection of power/signal cables (low frequency, high frequency, waveguide etc.). The equipment with orientation requirement (such as thrust vectors, sensors) should have accurate orientation. The equipment with FOV requirement (such as thrusters, sensors, antenna beams) should have no obstructions, reflected stray light and thermal noise radiation in their FOV. The equipment with accuracy requirement (such as sensors, gyros) should meet the installation accuracy requirements of the subsystems in terms of position and angle and should be arranged in the areas that are rigid, less affected by different AIT states and accessible to the measurement channel.

7.2.1.4 Meet the Mechanical Resistance Requirements

For the whole spacecraft structure, the structural mass borne by each cabin should be reasonably allocated. The large-mass equipment should be located in the spacecraft area with strong carrying capacity. The mechanical sensitive equipment should be reasonably positioned according to the structural response.

7.2.1.5 Meet the Thermal Control Requirements

The instrument and equipment should be reasonably laid out according to their power consumption and the requirements of radiating surfaces and heat dissipation channels across the whole spacecraft to avoid local overcooling or overheating. If necessary, local thermal control measures should be taken. Batteries, discharge regulators and shunts are generally arranged in the spacecraft areas with relatively stable heat flows. Considering the requirements of spacecraft temperature control, the equipment with large heat consumption should get close to the radiating surface and have a large radiation angle coefficient. The equipment with high/low heat consumption should be alternated. The equipment layout should be coordinated with thermal control design.

7.2.1.6 Meet the Electromagnetic Compatibility Requirements

Instruments and cables should be rationally arranged to maximize the circuit performance and reduce their remanent and electromagnetic interference with the whole satellite.

7.2.1.7 Meet the Anti-irradiation Requirements

In the layout design, an appropriate installation position with small radiation dose should be selected for each instrument according to its radiation resistance and the shielding effect between the structure and the instrument.

7.2.1.8 Meet the Lightweight Requirements of Cable Network

While meeting the requirements of electrical performance, the layout design should optimize the length and weight of the onboard cable network by adjusting the position and direction of the equipment. The suggestions on the layout of electrical connectors should be provided when necessary.

7.2.1.9 Meet the AIT Requirements

The layout design should meet the requirements for AIT test status, operating convenience, operating space (such as large-component docking, equipment installation, cable plugging/unplugging and installation, pipeline welding and radiography, precision-measurement light path, all kinds of measurements and tests, and ground equipment docking, pre-launch state setting) and operating safety.

7.2.1.10 Meet the Basic Constraints of the Inherited Platform

When laying out the equipment on the inherited platform, the designer should fully understand the relevant information on the panels and cabins where the equipment is installed, and on the original equipment (including the panel itself) in those areas.

In addition to meeting the general and special requirements for the new equipment, its interference effect (the obstruction of FOV, accuracy test channel and cable channel) on the original peripherals should be fully demonstrated.

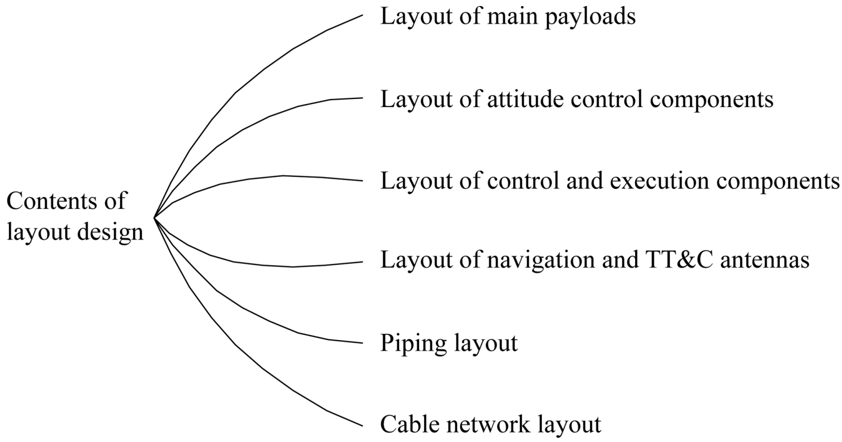

FIGURE 7.16 Contents of layout design.

7.2.2 Contents of Layout Design

See Figure 7.16.

7.2.2.1 Layout of Main Payloads

The layout of large components (fuel tanks, air cylinders, engines, solar wings, antennas) and main payloads often determines the spacecraft configuration, so the layout of main payloads has been completed in the configuration stage. The layout of main payloads mainly considers the following factors:

7.2.2.1.1 Layout of Optical Payloads Focus on the orientation, FOV, stray light suppression, installation accuracy and its maintenance (in the position with better rigidity), heat dissipation channel, thruster plume pollution, in-orbit perturbation/vibration avoidance, installation and operation convenience, maintainability etc.

7.2.2.1.2 Layout of Microwave Payloads Focus on the payload orientation and the clutter interference caused by the spacecraft body. For the large payloads deployed in orbit, attention should also be paid to the influence of their deployment on the spacecraft stability.

7.2.2.2 Layout of Attitude Control Components

7.2.2.2.1 Layout Criteria for Attitude Control Sensors The layout of attitude control components mainly considers the layout of attitude sensors and actuators and is implemented according to the general layout rules.

1. The layout design should ensure that the attitude control components and actuators meet the requirements of installation polarity and orientation.

2. The layout design should meet the FOV requirements of optical measurement components. If the required FOV is hard to achieve, the System Engineering should coordinate with the Control Subsystem Department and determine the layout after the relevant analysis and demonstration.
3. The layout design should meet the installation accuracy requirements of high-precision equipment. The equipment should be installed in the areas that are rigid, less affected by different AIT states and accessible to the measurement channel.
4. According to the mission requirements, some sensors (such as star sensor and main payloads) are integrated.
5. The layout design should avoid the dynamic coupling between the executors, such as momentum wheel and control torque gyro, and the main payloads.

7.2.2.2.2 Layout of Attitude Sensors Attitude sensors can be divided into the following five types according to different reference orientations:

1. Oriented to the Earth: infrared horizon sensor and Earth light sensor
2. Oriented to a celestial body: sun sensor, star sensor
3. Oriented to inertial space: gyro, accelerometer
4. Oriented to ground station: RF (Radio Frequency) sensor
5. Others: such as magnetometer (oriented to geomagnetic field) and landmark sensor (oriented to ground features)

Attitude sensors can be divided into the following four types according to different transducers:

1. **Optical sensors:** sun sensor, infrared horizon sensor, star sensor, Earth light sensor etc.
2. **Inertia sensors:** gyro, accelerometer
3. **Radio sensors:** e.g. RF sensor
4. **Others:** e.g. magnetometer

The layout design of the most commonly used attitude sensors (sun sensor, infrared horizon sensor, star sensor and gyro) will be described below.

7.2.2.2.2.1 Sun Sensor The sun sensor is a sensor which can measure the angle between the line of sight of the sun and an axis of the spacecraft by using its sensitivity to solar radiation. The sun sensors can be divided into analog sun sensor, digital sun sensor and 0–1 sun sensor.

1. Analog sun sensor

   Usually, a pair of analog sun sensors is installed on a spacecraft, that is, on the sunny side of the spacecraft. The slits of the two sensors are vertically arranged. The sensors should have no shade in the FOV and meet the requirements of thruster plume.

2. Digital sun sensor

   The digital sun sensor is arranged on the sunny side of the spacecraft. It should have no shade in the FOV and meet the requirements of thruster plume.

3. 0–1 sun sensor

   The FOV of a 0–1 sun sensor is a hemisphere. Generally, two 0–1 sun sensors are used to obtain a complete spherical FOV. If two 0–1 sun sensors cannot obtain a complete spherical FOV, three 0–1 sun sensors can be used while meeting the requirements of thruster plume.

7.2.2.2.2.2 Infrared Horizon Sensor At present, the infrared horizon sensor most commonly used in LEO spacecrafts is conical scanning infrared horizon sensor. Generally, two sensors, either vertically or coplanarly installed, must be used to meet the attitude measurement requirement:

1. Vertical installation is to set two identical conical scanning horizon sensors in the spacecraft's rolling axis and pitching axis (90° apart) to directly measure the roll angle and pitch angle.
2. Coplanar installation means that the two scanning axes are installed on the same plane as the local perpendicular line. The pitch deviation is still measured from the ground, while the rolling attitude is obtained by comparing the two chord widths of the Earth measured by the two scanning mechanisms crossing the Earth. In other words, when the two chord widths are identical, the roll angle will be zero.

7.2.2.2.3 Star Sensor A spacecraft is generally equipped with two to three star sensors to meet the attitude measurement requirements. The layout of star sensors should meet the following requirements:

1. The FOV ranges of star sensors do not coincide.
2. No occlusion exists in the FOVs of star sensors.

3. No stray light (such as sunlight, Earth atmospheric light and Earth surface reflection) exists in the FOVs of star sensors.
4. The heat dissipation requirements are met.
5. The requirements of thruster plume are met.

7.2.2.3 Layout of Control and Execution Components

The actuators carried by a spacecraft mainly include thruster, flywheel, magnetic torque and solar array driving actuator (SADA).

1. Thruster
   a. The thruster layout should consider the relationship between the thrust vector and the spacecraft center of mass to save fuel and reduce the disturbance to the spacecraft.
   b. The completion of thruster layout should be followed by thruster plume analysis to avoid the impact of the plume on the spacecraft in terms of pollution (especially for optical equipment), heat, interference torque and electromagnetism.

   A typical thruster layout is shown in Figure 7.17.

2. **Flywheel:** a typical flywheel layout is shown in Figure 7.18.
3. **Geomagnetic torquer:** three orthogonal geomagnetic torquers are generally installed.
4. **SADA:** the layout of SADAs shall ensure that the solar wings point to the sun.

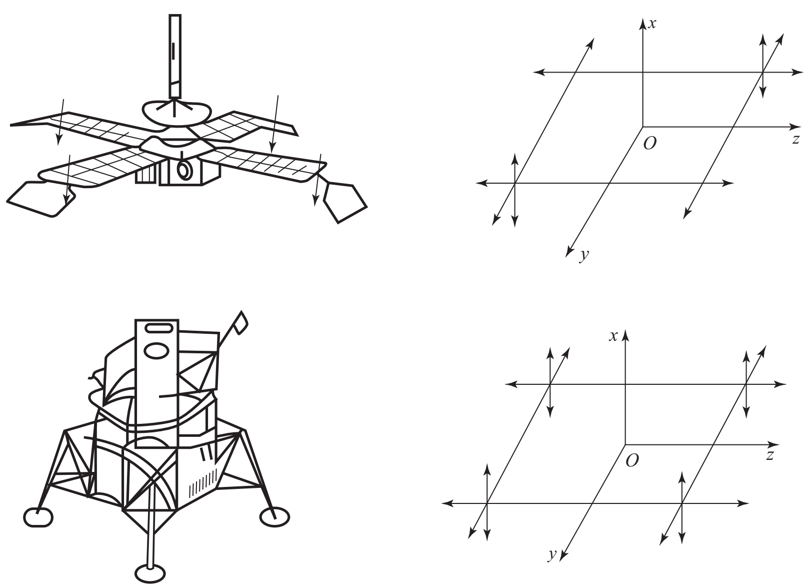

FIGURE 7.17 Thruster layout.

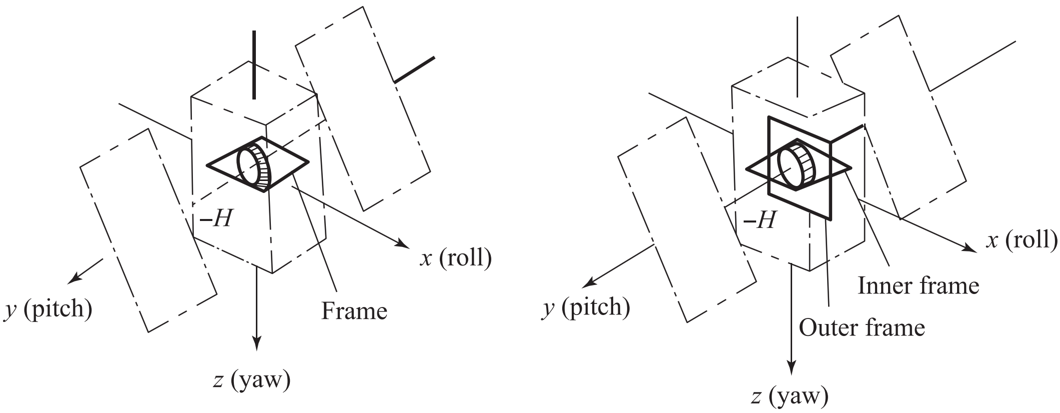

FIGURE 7.18 Flywheel layout.

7.2.2.4 Layout of Navigation and TT&C Antennas

7.2.2.4.1 Basic Requirements

1. According to the various flight attitudes of the spacecraft in orbit, the requirements of main coverage area and the influence of radiation and receiving levels outside the antenna main lobe are considered, and the electromagnetic compatibility interference between different antennas is effectively isolated in space. The antennas should be as far away from other protruding devices on the spacecraft surface as possible. The completion of antenna layout should be followed by the simulation analysis of directional diagram.

2. To meet the insertion loss requirements, the direction and length of the high-frequency cable running from the antenna to the transponder should be given priority.

3. The TT&C antenna layout should consider the wave-transmitting opening position of the fairing.

7.2.2.4.2 Navigation Antenna The navigation antenna is located on the sky-oriented side of a LEO spacecraft or the ground-oriented side of a HEO spacecraft. Its layout should meet the requirements of both FOV and antenna pattern while considering the thermal effect of thruster plume.

7.2.2.4.3 TT&C Antenna The TT&C antennas include a ground-oriented TT&C antenna and a sky-oriented TT&C antenna. The TT&C antenna layout should first meet the requirements of TT&C antenna pattern while considering the thermal effect of thruster plume.

7.2.2.4.4 Relay TT&C Antenna The relay TT&C antenna is located on the sky-oriented side of the spacecraft. Its layout should meet the requirements of TT&C antenna pattern and consider the thermal effect of thruster plume.

7.2.2.5 Piping Layout

The layout design of satellite piping system should be completed according to the layout and principle diagram of the system. The following factors should be considered during the design:

1. The piping system should be arranged on the structural components that will no longer be dismantled after the partial spacecraft structure assembling.
2. The piping layout should consider all operation requirements in the full life cycle (such as welding, post-loading inspection, propellant filling).
3. The design of piping direction and shape mainly considers the requirements for welding tooling and fixture, cleaning, photographic inspection, thermal control cladding and the safety distance to the structure.
4. For the pipes disconnected for process reason, special consideration should be given to the error compensation during docking, the position of process welding joints and the operation space.
5. The piping layout must not interfere with the operation of other onboard equipment (such as assembly/disassembly, screw fastening, fastener force measurement).
6. The component layout, piping direction and connection design should minimize the number of welding spots on the spacecraft.

7.2.2.6 Cable Network Layout

At the early stage of satellite layout design, the cable layout of the whole satellite should be planned in a unified way. The detailed cabling design is carried out at the detailed final-assembly design stage. The factors to be considered in the cable layout are as follows:

1. The pre-design of cable layout and channels, including cable trunk routing and cable hole layout, should be completed according to the equipment layout.
2. The cable density of the whole spacecraft should be estimated, and multiple channels should be designed and laid in the dense area to facilitate the cable harness fixation.
3. The cable path planning should consider both the path simplicity and the operation convenience.

7.3 FINAL-ASSEMBLY DESIGN CRITERIA AND DESIGN CONTENTS

7.3.1 Final-Assembly Design Criteria

See Figure 7.19.

7.3.1.1 Meet the Subsystem Requirements

The final-assembly design should meet the requirements of each subsystem, such as the requirements for general equipment installation (fastening, electrostatic resistance, grounding and heat insulation), the special installation requirements (installation process, status) for large primary payloads, the requirements for equipment polarity and propulsion piping installation in the attitude and orbit control subsystem, and the requirements for solar-wing and battery assembly in the power supply and distribution subsystem.

7.3.1.2 Meet the AIT Requirements

The final-assembly design serves the final assembly, testing and large-scale experiment of a spacecraft. It must meet the requirements for final assembly in each stage of AIT, reasonably formulate a final assembly plan and an assembly technique process, and specify the implementation status of final assembly at each stage.

7.3.1.3 Meet the Piping and Cable Channel Layout Requirements

The layout of pipelines and cable channels is completed at the early stage of configuration layout. The detailed design of piping and cable network path is completed at the stage of final-assembly design and must not contradict with the equipment layout.

7.3.1.4 Meet the Requirements of Operability and Maintainability

In the final-assembly design, attention should be paid to the operability of equipment disassembly and assembly, the operability of pipeline welding/connection/flaw detection, the operability of electrical connector plugging/unplugging, the operability of ground

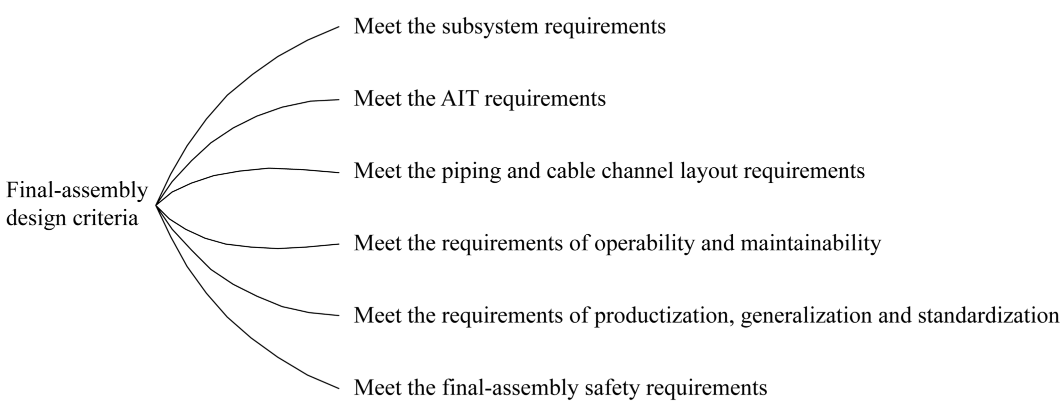

FIGURE 7.19 Classification of final-assembly design criteria.

simulator disassembly/assembly, the accessibility of accuracy test equipment and channels, and the accessibility of grounding implementation.

7.3.1.5 Meet the Requirements of Productization, Generalization and Standardization

The components directly under final assembly should be designed according to the productization principle. The grounding wires, counterweights, adjusting shims and other products should be managed according to the type spectrum. The ground support equipment should be generalized and standardized to reduce costs and improve efficiency.

7.3.1.6 Meet the Final-Assembly Safety Requirements

The impact of final assembly, testing, packaging, loading and unloading, transportation, maintenance, storage and other processes on product safety should be considered. Necessary safety protection capabilities and safeguards should be provided. Special attention should be paid to the requirements for electrostatic protection, pyrotechnics installation, battery pack installation, redundancy control and high-voltage device maintenance.

7.3.2 Contents of Final-Assembly Design

See Figure 7.20.

7.3.2.1 Assembly Scheme Design

1. Define the main work items and states in the AIT process of the spacecraft.
2. Define the main work items, main stages and main technical conditions of each stage in the final assembly process according to the technical development process and

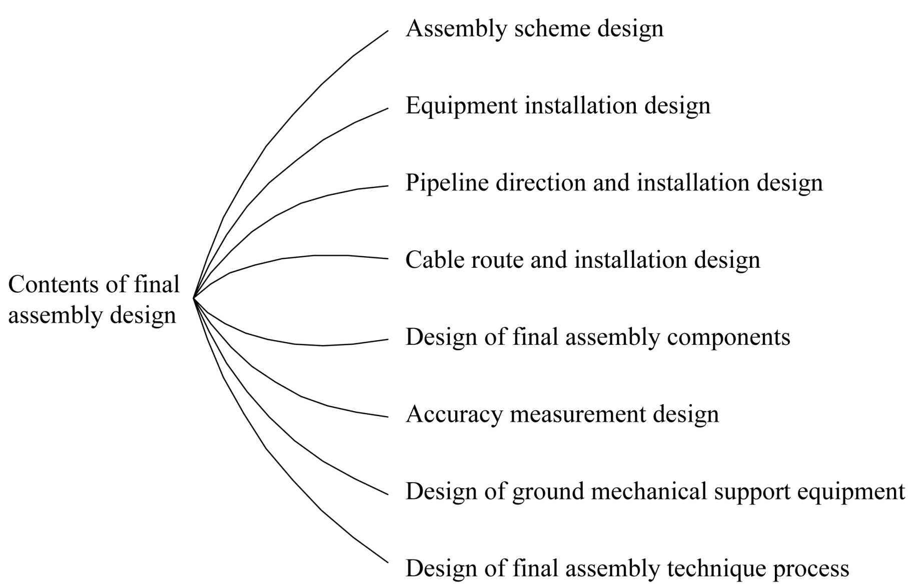

FIGURE 7.20 Contents of final-assembly design.

electrical performance test technology of the spacecraft, and plan the final-assembly design accordingly.

3. Preliminarily plan the process, stages and required states of final assembly, including the assembly and testing states of the spacecraft and its cabins, in the final assembly, testing, experiment and transportation processes. On this basis, establish the preliminary assembly process and put forward the safeguard requirements.
4. Identify and extract key technical links, and determine the solutions and approaches.
5. Analyze the special requirements for spacecraft configuration, equipment layout and key equipment installation, and fully understand the test status and large-scale experimental status of each stage. Extract key assembly points and validation items that have an important influence on the performance and safety of the spacecraft. Determine the solutions and measures. If necessary, analyze the failure mode.
6. Identify the required ground mechanical tooling through preliminary coordination with technicians to meet the demands for spacecraft/cabin assembly, parking, lifting, overturning and transportation.
7. Identify important output files/drawings/models.

7.3.2.2 Equipment Installation Design

1. **Installation type design:** for the equipment (such as a large payload) that cannot be directly connected with the primary structure or the equipment with special requirements like pointing accuracy requirement and motion requirement, their support and adapter types should also be considered to propose the design requirements for secondary structures.
2. **Installation direction design:** the installation direction should be specified by means of reference holes according to the requirements for equipment function and polarity, as well as the equipment layout and orientation.
3. **Installation safety design:** the equipment installation design should consider the installation space of high and low frequency cables outside the equipment, and a safe distance should be kept between two pieces of equipment and between equipment and structure. For the equipment with moving parts (such as control moment gyro and deployable antenna), the motion of those parts must not interfere with the surrounding cables.
4. **Installation process design:** attention should be paid to the equipment assembly sequence and installation accessibility during the equipment installation design. For example, in some areas with limited installation space, the mounting/dismounting sequence of adjacent equipment should be followed to make reasonable arrangements and avoid repetition.

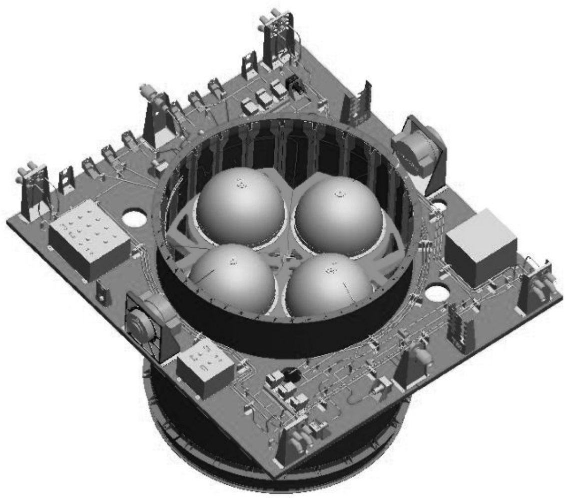

FIGURE 7.21 Piping route of monopropellant propulsion system.

7.3.2.3 Pipeline Direction and Installation Design

According to the requirements for final assembly in the propulsion subsystem (and the life support subsystem for a manned spacecraft), the pipeline route begins to be designed after the main components and valves of the piping system have been laid out at the configuration layout stage. After the pipeline layout is finalized, the pipe installation design is started. The typical propulsion system consists of a monopropellant propulsion subsystem and a bipropellant propulsion subsystem. The pipeline layout of a monopropellant propulsion system is relatively simple (as shown in Figure 7.21), generally including liquid/gas supply and discharge valves, pressure sensors, self-locking valves and filters. The equipment in a bipropellant propulsion system includes tanks (including oxygen tank and fuel tanks), air cylinders, thruster, orbit control engine and various valves. The fuel includes combustion agent and oxidant. Helium is generally used as pressurizing medium, and the balanced emission is required, so the pipeline layout is complicated (as shown in Figure 7.22).

The pipeline route design is mainly to identify the panels for propulsion pipeline arrangement based on the position of thruster and tanks, and then reasonably install the pipeline valves onto the corresponding structure according to the principle diagram and configuration layout of propulsion system. For a typical high-orbit spacecraft platform, the valves of propulsion system are placed primarily on the central panel, and the valves with operation requirements are installed near the spacecraft surface. For a spacecraft with shell structure, the pipeline values are usually arranged on the side wall, and the corresponding operation windows are opened at the valves that need to be operated. After the pipeline route design is completed, the pipe “branches” are connected via fittings to the appropriate positions of the pipeline. There are two typical design methods of pipeline layout: three-dimensional pipeline design and simulated pipeline assembly. The three-dimensional pipeline design is to complete the digitalized assembly of a pipeline using the design tool software, while the

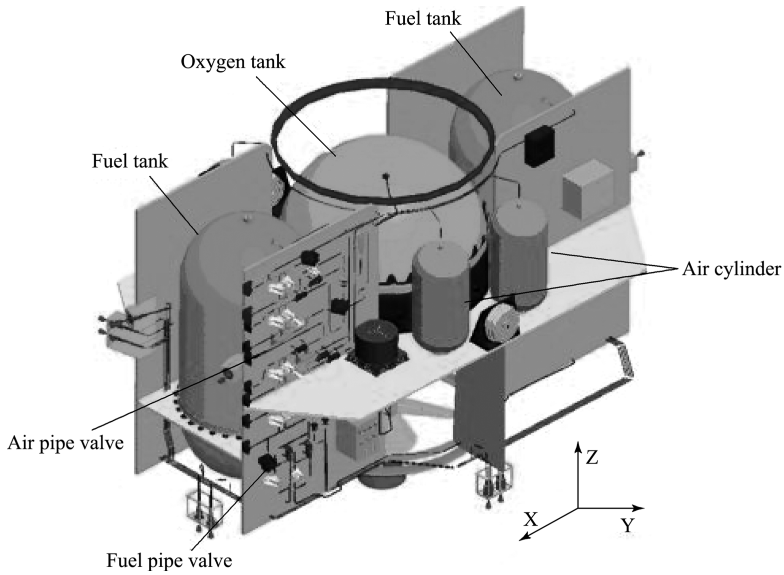

FIGURE 7.22 Piping route of bipropellant propulsion system.

simulated pipeline assembly is to physically simulate the pipeline layout and assembly on the simulated structure (usually on a complex spacecraft).

The pipeline installation design is mainly to connect and fix the propulsion components, valves and pipes to the structure. Typical propulsion components include attitude control thruster, orbit control engine, tanks and air cylinders. The propulsion valves and pipes are generally connected and fixed to the structure through supports (directly belonging to the final assembly).

7.3.2.4 Cable Routing and Installation Design

The spacecraft cables are divided into low-frequency cables and high-frequency cables. Cable routing design is a digital cabling design based on 3D model (as shown in Figure 7.23). According to the characteristics of spacecraft configuration and layout, the raceways of cable harnesses are generally set up in the whole spacecraft model, and each cable harness is routed separately. Depending on the cable complexity and source, high-frequency cables are generally put into production by means of branch length, while low-frequency and bus cables are mainly put into production by means of branch length or template cable.

Cable installation design is a cable routing design at the implementation layer, mainly including the 3D model selected for cable bundling and turning, which should be consistent with the 3D model used for spacecraft opening design. The 3D model needs to be accurately rechecked and detailed before the formal cabling design. The contents to be rechecked include the location of panel holes, the code, installation direction and position of equipment, and the installation position of nylon base. The contents to be detailed include the name and position of electrical connectors on the equipment.

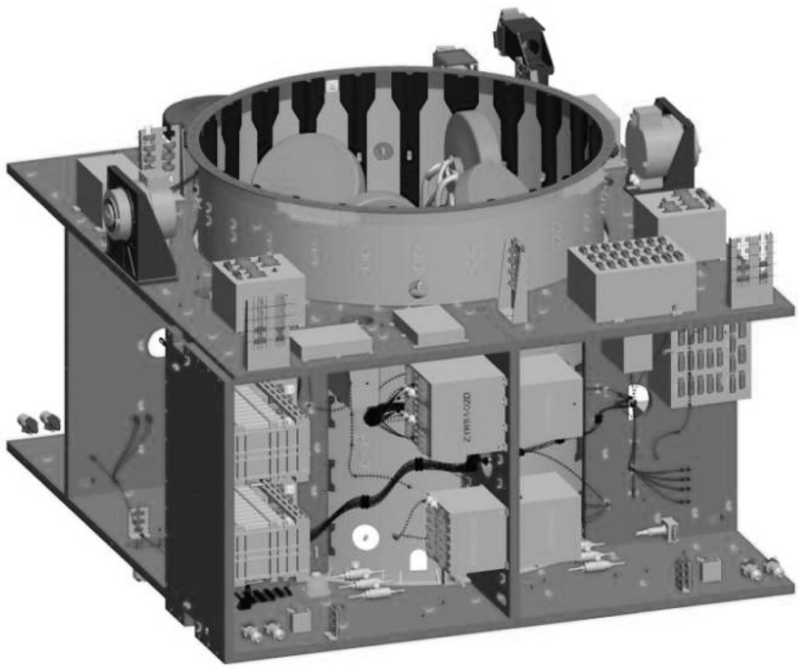

FIGURE 7.23 Spacecraft cable routing design.

7.3.2.5 Design of Final Assembly Components

Final assembly component is a special component used on the satellite to meet certain installation and positioning requirements (such as position, angle, precision) or other requirements of the equipment.

The final assembly components generally include instrument installation supports, pipeline supports, cable supports, ground wires, adjusting shims, heat insulation gaskets and counterweights, as shown in Figure 7.24.

The design of final assembly components can refer to the relevant standards. The following factors should be considered when designing those components:

1. The final assembly components had better be the products approved through productization or parameterization.
2. The cable clips, adjusting shims, pipe clamps and transition plug supports, which are widely used on the spacecraft, should meet the requirements of universality, interchangeability and parameterization.
3. In addition to function, the strength and rigidity of final assembly components should be guaranteed. The first-order fundamental frequency of an equipment support on the spacecraft is usually required to be greater than 100 Hz, and the fundamental frequency of the equipment combination installed on the support is generally required to be greater than 70 Hz.

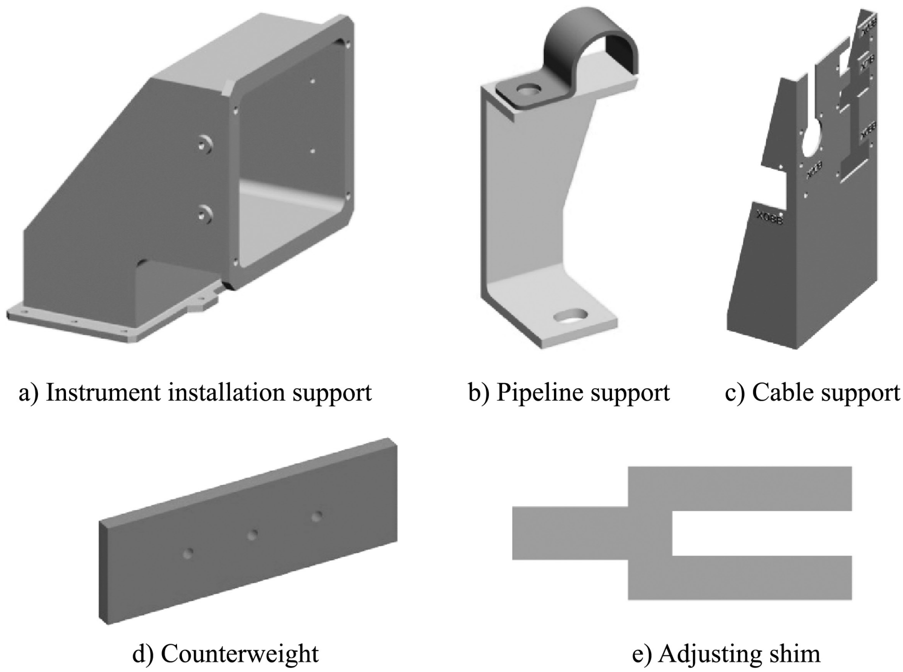

FIGURE 7.24 Spacecraft final assembly component.

4. According to the assembly relationship between final assembly components and the equipment (and spacecraft), the equipment and final assembly components can be assembled with enough assembly space, measurable fastener force, electrical-connector plugging/unplugging space and wiring space.

7.3.2.6 Accuracy Test Design

The accuracy measurement process of a spacecraft is shown in Figure 7.25. Its design should follow the following principles:

1. The accuracy measurement datum should be selected at the position less affected by no-load and full-load conditions to prevent the change of no-load and full-load measurement datum from affecting the measurement accuracy.
2. Attention should be paid to analyzing the influence of different states on accuracy measurement datum and zero gravity datum, such as the influence of the pressure of inflatable structure cabin. If necessary, a third-party datum should be derived to ensure the consistency of measurement datum between the two states.
3. When setting the accuracy test state, the structure state and equipment state should also be considered to prevent the obstruction of accuracy measurement path or the large change between the first and second measurement states of the equipment, which may result in the data incomparability.

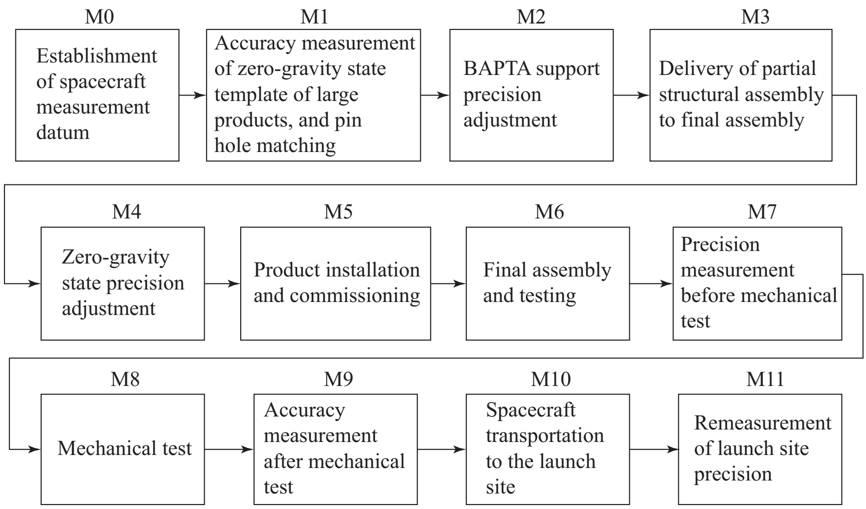

FIGURE 7.25 Typical spacecraft accuracy measurement process.

4. Attention should be paid to the relationship between the coordinate system of the measured equipment and the coordinate system of the whole spacecraft when combing the accuracy test requirements.
5. For the equipment requiring the transformation of the spacecraft coordinate system, the development organization of the equipment under measurement should provide the "transfer matrix of reference mirror coordinate system relative to the spacecraft coordinate system".

7.3.2.7 Design of Ground Mechanical Support Equipment

7.3.2.7.1 Packing Case The packing case (as shown in Figure 7.26) is mainly used to transport a spacecraft from the final assembly plant to the launch site. It is designed by using automobiles, trains and airplanes as the carriers and fully considering the state of spacecraft transportation.

In addition to providing the functions of load bearing, environmental monitoring (temperature and humidity, pressure, vibration, shock etc.) and data collection, the packing case should also have the function of leak detection. Its design concept should meet the need for generalization. The packing case should also meet the requirements for sealing, waterproof and pressure balance.

7.3.2.7.2 Two-Axis Turntable The design of two-axis turntable is mainly based on the deployment of the relevant antenna payload, the docking of solar wings, the relevant accuracy measurement and other final assembly operations.

Based on the final assembly requirements of the spacecraft, the design of two-axis turntable includes not only the design of carrying capacity, grounding and outer envelope size but also the requirement for two-axis rotation capacity. According to the requirements of

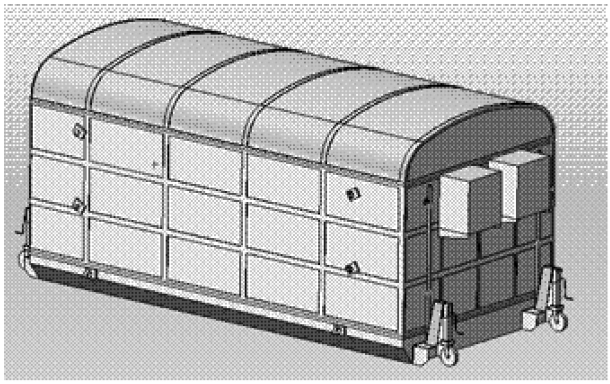

FIGURE 7.26 Typical packing case for spacecraft use.

antenna payload deployment and solar wing docking altitude/mode, the turntable should have a rotation angle range of 0°–360° around its axis (vertical to the ground) and an inclination range of 0°–90°. Meanwhile, it should be able to provide vertical and horizontal (front, back, left and right) adjustment.

7.3.2.7.3 Spacecraft/Cabin Parking Vehicle The spacecraft/cabin parking vehicle is mainly used for packing the spacecraft or an individual cabin under development in the process of final assembly and electrical performance test. Its design should mainly consider the interface between the vehicle and the spacecraft, the carrying capacity and anti-overturning capacity of the moving or stationary vehicle, the height adjustment ability and operation convenience of the vehicle, as well as the design of its lifting point to facilitate its transfer.

7.3.2.7.4 Spacecraft/Cabin Hoisting Tool The lifting mass, center of mass and lifting-point position under different working conditions must be defined in the design of a hoisting tool used for lifting a spacecraft or cabin. Compared with the parking vehicle, the hoisting tool has higher safety requirement (Figure 7.27).

7.3.2.8 Design of Final Assembly Technique Process

The process of final assembly is generally divided into several stages, including partial assembly, cabin assembly, spacecraft assembly, mechanical test assembly, thermal test assembly and launch site assembly.

7.3.2.8.1 Partial Assembly The partial assembly is mainly to complete the assembly of the main spacecraft frame (structure). In order to achieve the machining accuracy of part of the structure, some of the final assembly work needs to be completed, such as the datum derivation, the installation and commissioning of SADA bracket, and the commissioning of camera installation template.

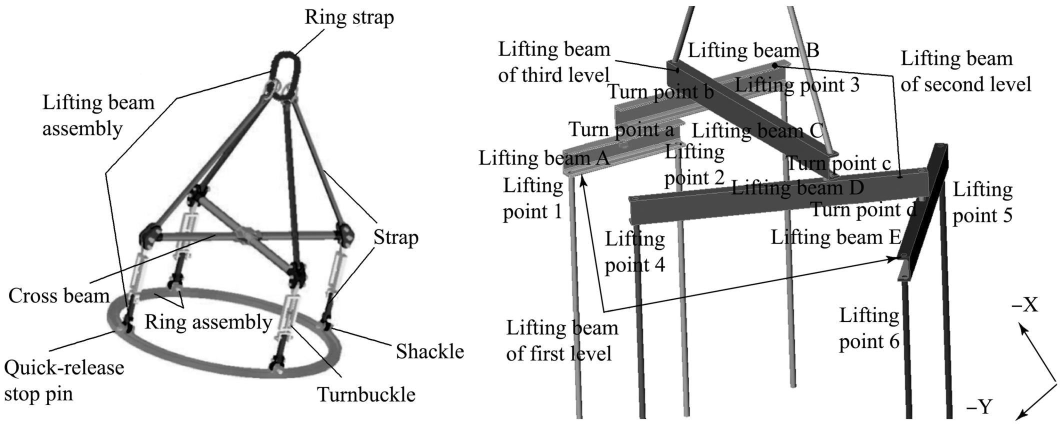

FIGURE 7.27 Typical spacecraft (cabin) hoisting tool.

7.3.2.8.2 Cabin Assembly The cabin assembly mainly refers to the cabin-level (or module-level) final assembly after the delivery of partial structural assembly. It mainly includes the final assembly of pipelines, cables and equipment in each module and the commissioning of the equipment with precision requirement. Due to a large number of electrical performance test items and a long testing time, the transition between final assembly and electrical performance test needs special attention.

7.3.2.8.3 Spacecraft Assembly The spacecraft assembly mainly refers to the process of assembling and integrating independent modules into a spacecraft. It mainly includes cabin docking, (part of) outer panel installation, large components (equipment) installation and various large-scale special tests (such as antenna unlocking and deployment test, spacecraft mass characteristics test and spacecraft accuracy measurement).

7.3.2.8.4 Mechanical Test Assembly The assembly work at this stage mainly includes the adhesion of mechanical sensor to spacecraft surface, the installation of spacecraft-rocket unlocking device, the installation and fixation of spacecraft-rocket transition cable (optional), the removal of spacecraft surface equipment and multi-layer protective cover, the unlocking of spacecraft-rocket unlocking device after mechanical test, the deployment of solar wings after test, and the accuracy measurement and leak detection after test.

7.3.2.8.5 Thermal Test Assembly The thermal test assembly will focus on the installation and implementation of onboard thermal control materials and simulated external heat flow, as well as the routing and outgoing of ground test cables from the vacuum tank. It mainly includes the adhesion and cabling of thermocouples to cabins, the installation and routing of short cables of special equipment, the installation of multi-layer protective cover and external heat flow on spacecraft surface, and the routing and connection of ground test cables from the vacuum tank.

7.3.2.8.6 Launch Site Assembly The final assembly on launch site includes the final assembly in the technical area and that on the tower. The assembly work in the technical area is similar to that in the assembly plant. The assembly operations on the tower mainly include electrical performance test and pre-launch onboard state setting, which need to fully consider the matching and coordination with other mechanical and electrical interfaces on the launch vehicle and launch site.

7.4 ANALYSIS OF CONFIGURATION AND LAYOUT

7.4.1 Large-System Compatibility Analysis

7.4.1.1 Spacecraft-Rocket Compatibility Analysis

After the completion of spacecraft configuration design, the rationality of the configuration layout should be verified from three aspects: the space interference between the spacecraft and the fairing, the correctness of spacecraft-rocket docking and the safety and reliability of the unlocking separation, and the rationality of the fairing opening.

7.4.1.1.1 Analysis of Space Compatibility The cylindrical section and inverted cone section of the fairing should not interfere with the spacecraft structure or equipment. The spacecraft-rocket unlocking separation device in motion should not interfere with the spacecraft. In particular, more attention should be paid to the interference of spacecraft surface protrusions in the final assembly state (considering the thermal control implementation, cable routing and other factors). When necessary, the dynamic interference should be analyzed and checked by the spacecraft-rocket coupling analysis.

7.4.1.1.2 Confirmation of Spacecraft-Rocket Mechanical Interface Check the matching of docking orientations, sizes and tolerances between spacecraft and rocket, and the matching between the spacecraft-rocket unlocking separation device and cable and the rocket.

7.4.1.1.3 Analysis of Fairing Opening Rationality Analyze the openings on the fairing such as air vent, ground operation port and TT&C wave-transmitting opening to check whether they can meet the operation requirements.

7.4.1.2 Spacecraft-Rocket Coupling Analysis

In order to ensure that the spacecraft configuration can successfully withstand the harsh dynamic environment of powered launch stage and safely enter the orbit, the spacecraft-rocket coupling analysis should be carried out after the configuration design. The result of spacecraft-rocket coupling analysis is the premise of determining the overall spacecraft concept (structural concept) and is also the basis of establishing environmental specifications and mechanical conditions. Generally, the spacecraft-rocket coupling analysis needs to go through three stages: conceptual analysis, prototype analysis and flight model analysis, the requirements for which will be proposed by the spacecraft developer. For a mature platform, part of the work can be carried out according to the specific situation. Some work can also be done depending on the spacecraft development stage, the schedule arrangement and the available input information. This, however, requires the agreement with the rocket developer. The main analysis contents are as follows:

1. The modal analysis of rocket/spacecraft combined structure provides the parameters related to structural dynamics for the design of rocket attitude control system.
2. The vibration response analysis of the spacecraft structure under the most severe longitudinal and lateral loads of the carrier rocket gives the time histories of the relative displacement, acceleration and force at the position required by the user as well as their maximum and minimum values.
3. The analysis of the dynamic clearance between the carrier rocket and the spacecraft during launch confirms the launch safety margin.
4. The load on the rocket/spacecraft interface is determined to provide a reference for the design of spacecraft vibration environment conditions under test. The analysis results need to be confirmed by both the spacecraft developer and the rocket developer.

7.4.2 Mission Adaptability Analysis

7.4.2.1 Frequency Response Analysis

The spacecraft launch will be affected by harsh mechanical environment in the powered phase. To ensure safety and meet the requirement for spacecraft stiffness specified in the spacecraft-rocket interface code, the modal and frequency response of the spacecraft need to be analyzed to simulate the state of spacecraft launch. The analysis of modal and frequency response can be divided into the following phases: simplified analysis, preliminary analysis, detailed analysis, and modification and verification analysis. The frequency response analysis generally includes the following three parts:

1. Modal analysis and calculation of the whole spacecraft: natural frequency and the corresponding vibration mode.
2. The local modals and vibration modes of large equipment (including primary payloads) and special equipment (such as the gyro sensitive to mechanics).
3. The analysis and prediction of responses (displacement, acceleration, strain etc.) of large payloads and key equipment and areas.

The correctness of the modal and frequency response analysis results is verified by spacecraft mechanical environment test. If the response is too large during the test, the test magnitude can be concavely controlled based on the results of the spacecraft-rocket coupling analysis in order to avoid damage to the spacecraft or onboard equipment. After the test completion, the analysis model is corrected according to the test results, and the local configuration layout of the equipment with large response is optimized. Based on the corrected model, the modal and frequency response can be analyzed again until the requirements are met.

7.4.2.2 Analysis of Main Structure Accuracy and Thermal and Mechanical Stability

A small deformation of the spacecraft structure will cause the camera's ground target to deviate by a large distance (the 1″ altitude change of a 500-km orbit means a 2.5-m deviation). With the improvement of the positioning accuracy of a spacecraft (especially Earth observation satellite), the analysis of main spacecraft structure and mechanical and thermal stability is needed after the configuration layout. The analysis contents generally include the following:

1. The geometric precision index that the main structure needs to meet.
2. The deformation caused by gravity and assembly in ground environment.
3. The deformation caused by temperature change and stress release in orbital environment.

7.4.2.3 FOV Occlusion Analysis

After the spacecraft configuration design, the FOV of payloads and sensors (such as remote sensing payload, Earth sensor, sun sensor) and the propagation direction of the antenna's electric axis need to be specially analyzed to check whether they will be occluded by the satellite body and other components. The main analysis work is as follows:

1. Sort out and clarify the orientation and FOV requirements of all equipment to avoid ambiguity in the definition of FOV.
2. Recheck the consistency between the model and the interface data sheet, and check the FOV of each device one by one.
3. Check whether the FOVs of remote sensor and optical sensor are occluded by other equipment or structures.
4. Analyze and check the occlusion of equipment FOV by the motion space of solar wings and rotatable antennas.
5. The diffraction and transmission exist in the microwave FOV. If there is local occlusion, it should be provided to the RF designer for electrical performance occlusion analysis, as shown in Figure 7.28.

If the FOV occlusion is serious, the analysis accuracy of 3D modeling method will be poor. Therefore, the 3D modeling method is not applicable for accurately calculating the FOV occlusion. However, the sensor FOV can be accurately analyzed according to the principle of graphic correlation, as shown in Figure 7.29. This method can not only visually display

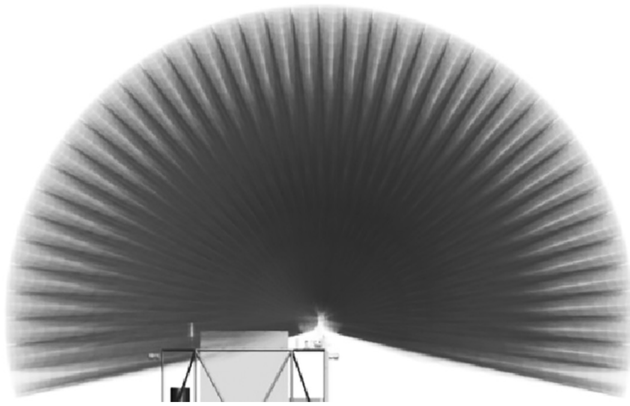

FIGURE 7.28 FOV occlusion analysis by 3D modeling.

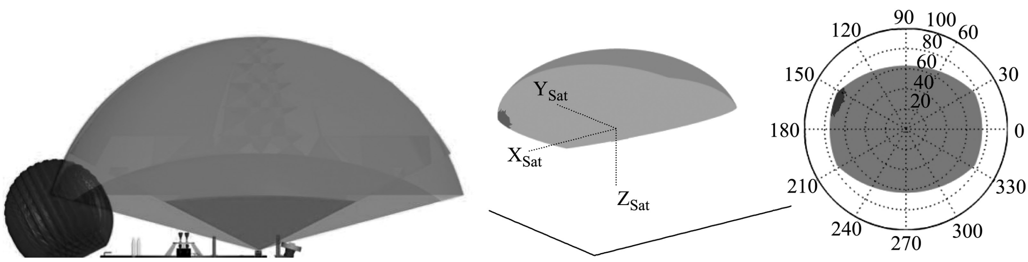

FIGURE 7.29 FOV occlusion analysis by numerical calculation.

the condition of FOV occlusion but also accurately calculate the size and position of the occluded area. The relevant designer should check whether the analysis results meet the requirements. If not, the configuration layout should be optimized and analyzed again.

7.4.2.4 Analysis of Quality Characteristics

The purpose of mass characteristics analysis is to obtain the mass characteristics (including mass, center of mass, rotational inertia and inertia product) of a spacecraft through software simulation, so as to check whether the configuration layout meets the overall requirements and to provide a basis for the adjustment and optimization of equipment layout and for the design of spacecraft control scheme. The configuration layout design should be followed by the estimation of mass characteristics (such as the weight, center-of-mass position and rotational inertia of the spacecraft) and the optimization of equipment layout (if required), so that the mass characteristics of the spacecraft can meet the relevant requirements of the carrier rocket and control subsystem. Different spacecraft missions will have different flight phases. In general, the mass characteristic parameters should be analyzed for the spacecraft during launch and orbit insertion, before and after the deployment of solar wings and antennas, and at the beginning of life and end of life. For some spacecrafts, the mass characteristic parameters before and after rendezvous and docking, assembly and separation in orbit need to be considered. The analysis results should be tabulated.

7.4.2.5 Analysis of Solar Wing Occlusion

Due to the constraints of the spacecraft's configuration and operating mode, the solar wings are often occluded by the spacecraft body and its surface equipment when the spacecraft is flying in orbit. This will directly affect the power generation capacity of solar wings and then affect the energy supply of the spacecraft. Therefore, solar wing occlusion analysis is needed in spacecraft design to provide a data support for spacecraft configuration design, energy system design and in-orbit energy evaluation. The solar wing occlusion analysis involves spacecraft geometric model, overall mission parameters, solar position and other factors, so the designer should formulate a three-level occlusion analysis document, define the inputs/outputs and analysis methods, standardize the analysis process and make the analysis task clear and scientific, so as to provide a strong guarantee for successful model development. The analysis process is shown in Figure 7.30.

(Any tool software, especially foreign tool software, must be deleted!!!)

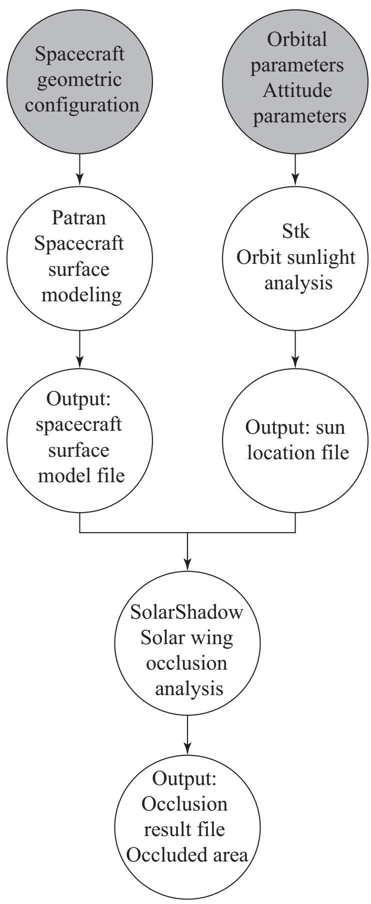

FIGURE 7.30 An example of solar wing occlusion analysis.

7.4.2.6 Stray Light Analysis

Stray light is produced by opto-mechanical structures, light sources outside FOV, imperfect optical parts, or thermal radiation of the optical or lighting system itself. For a payload such as optical camera, it will be directly irradiated by the Earth atmospheric light. After entering the system, such light will become non-imaging light through the reflection of optical element surface and the scattering of mechanical components and then reach the detector surface to form stray radiation noise that can affect the imaging performance of the camera system. After the configuration design, stray light analysis should be carried out for the camera. In particular, when there is an occlusion in the FOV of the camera, the corresponding stray-light suppression measures should be taken according to the result of stray light analysis (such as extending the outer lens hood, applying black paint on the surface of the shielding device).

7.4.2.7 Analysis of Antenna Electrical-Performance Occlusion

After the design of spacecraft configuration and layout, the FOV occlusion of each part of the antenna should be analyzed to confirm whether the antenna index is obviously affected by the spacecraft configuration and layout, and the suggestions on antenna layout adjustment and improvement should be given according to the analysis results. The analysis work mainly includes: analyzing the coverage gain in the antenna coverage area, and analyzing the influence of satellite occlusion on the polarization characteristics of the antenna with dual polarization requirement (such as spot-beam data transmission antenna); analyzing the phase center stability of the antenna with phase accuracy requirement (such as navigation antenna) and quantitatively analyzing the influence of unavoidable occlusion on the antenna performance.

7.4.2.8 Thruster Plume Analysis

The plume generated by the thruster will cause interference torque to the spacecraft attitude, or pollution to the optical devices, or thermal damage to the spacecraft surface equipment, finally resulting in the degradation of spacecraft performance. Therefore, the thruster plume analysis is needed after the completion of spacecraft configuration design. The detailed analysis methods and procedures are described in Chapter 9.

7.4.2.9 Analysis of Moving-Parts Disturbance

When the spacecraft is working in orbit, the disturbance of the onboard moving parts (such as the deployment of large solar wings, large antennas and radars, spatial flexible manipulators and large space truss structures and the movement of all kinds of multi-body mechanisms) will affect the spacecraft attitude control and normal payload operation and will degrade the imaging quality of remote sensor on a remote sensing satellite. Therefore, the disturbance analysis of the moving parts in orbit must be completed. The method and workflow of the disturbance analysis of the moving parts are detailed in Chapter 9.

7.4.2.10 Flexible Dynamics Analysis

Establishing a spacecraft attitude dynamics model approximate to the truth through the analysis of attitude coupling dynamic characteristics of various large-scale flexible components (such as solar wings, antennas) on an orbiting spacecraft and the calculation of coupling coefficient matrix is the basis of the conceptual design and simulation of spacecraft attitude and orbit control subsystem. It is vital to ensuring the attitude stability and pointing accuracy of a spacecraft. The method and workflow of flexible dynamics analysis are detailed in Chapter 9.

7.5 FINAL ASSEMBLY TESTING AND VALIDATION

7.5.1 Pipeline Leak Detection

7.5.1.1 Leak Detection Methods and Timing

The commonly used leak detection methods include helium mass spectrometry, bubble method, suction gun method and pressure drop method. Helium mass spectrometry is often used for the leak detection of air circuit system, liquid propellant transporting system and spacecraft system. The bubble method is usually used for the leak detection of air supply and exhaust valves and air test interface before and after filling. The suction gun method is often used for the leak detection of the welding and screwing points. The leakage of single pipeline point and solenoid valve nozzle is generally detected by using helium mass spectrometry, while the system leakage is generally detected by using non-vacuum collector mass spectrometry. The pressure drop method is often used in the leakage detection of a sealed structure.

In the final assembly process, four leakage detections are generally arranged for the propulsion pipeline:

1. The first leakage detection is to check the single-point leakage and the total leakage rate of the system after the pipeline of the propulsion subsystem has been welded. As a means of checking the installation quality and sealing effect of the pipeline system, the first detection can get rid of the problems caused by the early installation not in place, the equipment defects and the weld defects.

2. The second detection is to check the leakage of the screwing points and the total leakage rate of the system after the completion of spacecraft-level mechanical environment test. Its purpose is to examine the influence of mechanical test on the pipeline tightness, including whether the anti-looseness measures are effective in the vibration environment and whether the stress on the pipeline seals will cause plastic deformation.

3. The third detection is to check the total leakage rate of the system before the spacecraft delivery. This detection is generally arranged after the thermal test and before the delivery. The detection results will, on the one hand, provide support for the quality control of the delivered spacecraft. On the other hand, they can help evaluate the influence of thermal test on the pipeline tightness.

4. The fourth detection is to check the total leakage rate of the system after the spacecraft is transferred to the launch site. Its main purpose is to examine the influence of spacecraft transportation on the pipeline tightness.

7.5.2 Final-Assembly Accuracy Test

7.5.2.1 Contents of Final-Assembly Accuracy Test

The accuracy testing during the final assembly of a spacecraft mainly includes the following contents:

1. The measurement of main structure size and geometry with accuracy requirement, such as spacecraft docking clearance
2. The important interfaces of main structure with accuracy requirement, such as solar wing interface and optical payload mounting interface
3. The accuracy measurement and adjustment between the equipment and structure with accuracy requirement, such as the accuracy between attitude actuator and spacecraft datum
4. The accuracy measurement and adjustment between the devices with accuracy requirement, such as the accuracy between attitude measurement component and optical payload

7.5.2.2 Accuracy Test Method and Timing

The measurement methods of spacecraft assembly accuracy are mainly optical sighting measurement and mechanical measurement.

7.5.2.2.1 Optical Sighting Measurement The optical sighting measurement is based on theodolite plus processing software. The software is divided into general software and special software. This method is applicable to the non-contact measurement of large-sized structures and is widely used in spacecraft assembly measurement.

7.5.2.2.2 Mechanical Measurement The mechanical measurement tools mainly include Coordinate Measuring Machine (CMM) and Computer numerical control (CNC) machine tool. This measurement applies to the small-sized structures allowing contact.

7.5.2.2.3 Spacecraft Accuracy Test Generally, it is divided into five tests:

1. The first test is carried out in the stage of partial spacecraft structure assembly to measure the items such as assembly hole, overall machining and solar wing interface.
2. The second test is carried out during initial equipment installation, mainly for equipment installation and commissioning.

3. The third test is carried out before the mechanical test to measure the accuracy.
4. The fourth test is carried out after the mechanical test (before delivery) to measure the accuracy and ensure the consistency of the data before and after the test through comparison.
5. The fifth test is carried out at the launch site to remeasure the accuracy of the equipment with high accuracy requirement (6′ in general) so as to verify the consistency of the data before and after the equipment transportation and the consistency between the final data and the requirements.

7.5.3 Quality Characteristic Testing and Balancing

The tests of spacecraft mass characteristics mainly refer to the mass characteristics tests of the spacecraft body, including the measurement of envelope mass, center of mass and inertia. The purpose of these tests is to obtain the mass characteristics parameters of the spacecraft body and adjust the spacecraft center of mass into the required range by means of counterweight. The mass characteristic tests of a spacecraft also include the mass characteristic tests of general components, large components and deployable components, as well as the single-cabin weighing.

Both the mass and center of mass of the spacecraft are tested on the center-of-mass test bench by using a dynamic balancing machine. The rotational inertia is tested on a torsional pendulum table. At present, the CMM-based measurement of spacecraft mass characteristics has become a development trend.

CHAPTER 8

Spacecraft Reliability Design

Wang Jingyan, Cheng Zhuo, Gong Ying, Zhao Haitao, Xiong Xiao, Cheng Yao, Lv Xinqi, and Zhang Zhaoguo

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CONTENTS

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Reliability, safety, maintainability, testability and supportability—the general quality characteristics of a spacecraft—are important contents of spacecraft product design at all levels and are closely related to each other. This chapter will present the basic theories, methods, procedures and relations of the five properties.

The spacecraft reliability design is rich in contents nowadays. It belongs to a generalized category of reliability and overlaps with the context in this chapter. Therefore, from the perspective of the reliability in narrow sense, this chapter will present the general reliability design contents such as modeling, allocation, prediction, margin design, derating design, fault-tolerant design, failure mode and effects analysis (FMEA), fault tree analysis (FTA), event tree analysis (ETA), probabilistic risk assessment (PRA), sneak circuit analysis (SCA), worst case analysis (WCA), outage analysis, reliability mathematical simulation and reliability evaluation. In addition, the safety design contents of a manned spacecraft are quite different from those of an unmanned spacecraft. This chapter will focus on the methods of safety design, analysis and evaluation common to all spacecrafts.

8.1 RELIABILITY DESIGN ANALYSIS METHOD

8.1.1 Basic Theories of Reliability

The basic definition of spacecraft reliability is the ability of a spacecraft to accomplish the specified functions under the specified conditions within the specified time. The “specified conditions and time” should be determined after the mission profile analysis, while the “specified functions” should be decided by the spacecraft mission and determined after the functional analysis. Reliability is the probability that the product accomplishes the specified functions under the specified conditions within the specified time.[1]

The reliability is characterized by time dependence, statistics and comprehensiveness. The reliability of a product is a monotonically decreasing function of time, that is, it will decline over time. The reliability parameter is a measure that describes the reliability characteristics (such as reliability degree and life) of a product from different perspectives. The reliability characteristic quantity is the general term of various quantitative indexes that are used to describe the overall reliability of a product. Its theoretical value, namely, expected value, is unique. The reliability characteristic quantity can:

1. Express and judge the reliability, maintainability and effectiveness of a product simply and clearly by using numerical values.
2. Express the relationship between characteristic quantities mathematically to obtain the required results conveniently.
3. Reveal various factors affecting the product reliability, and describe their influence.
4. Make full use of various data of the product.

Confidence is the extent to which the reliability itself is credible. It is expressed by a certain confidence interval and is commonly used in reliability evaluation. Specifically, for the test sample of a product, confidence is the credibility of its test result. It is a probability expressed in percentage, representing the chance that the truth value of reliability degree appears in the confidence interval.

Life is the length of time in which a product can normally perform the specified functions. It is a random variable. Depending on the specific situation, life may refer to average life, reliable life, medium life or characteristic life. For a non-repairable product such as spacecraft, life refers to the service time before its failure.

The traditional reliability theory is the probabilistic reliability based on probability theory and mathematical statistics theory. With the development of science and technology, new reliability theories have penetrated into this field, and gradually developed into fuzzy reliability theory and gray reliability theory.

Since the 1990s, the reliability design has been improved from the "design in reliability" in the engineering development stage to the "design for reliability" in the conceptual design stage. The purpose of "design in reliability" is to improve the inherent reliability of a product and control and kill the faults in the engineering stage through sufficient reliability design and analysis, by using the derating design, thermal design, FMEA, FTA, reliability prediction, software component quality control and other technical means. The purpose of "design for reliability" is to achieve the synchronous integrated optimization of reliability and performance through the integrated design of reliability and performance.

8.1.2 Reliability Requirements and Allocation

The reliability requirements of a spacecraft product include qualitative requirements and quantitative requirements. The qualitative requirements are generally implemented by putting forward various design principles.

The formation of spacecraft feasibility demonstration scheme should be synchronized with the definition of system reliability index. Before the determination of final system design, the quantitative reliability requirements of the spacecraft system are determined according to the changes of the user's requirements and the technical level of the equipment, and then are allocated to the subsystems, equipment and components from top to bottom.

8.1.2.1 Reliability Index Demonstration and Determination

When putting forward the quantitative requirements for spacecraft reliability at the spacecraft project demonstration stage, the designer should made clear:

1. Life and mission profile;
2. Failure judgment criteria;
3. Index verification method;
4. Constraints and assumptions.

The confidence level and pass/fail criteria should be included for the products whose reliability needs to be evaluated with the experimental validation data.

The preconditions for spacecraft reliability index determination are the identification of basic technical conditions of the spacecraft (function, performance and main interface), the acquisition of empirical reliability data of products and components, the establishment of reliability model and the prediction analysis. Correct modeling and prediction is the main basis for determining the reliability index of a spacecraft, and is also the main basis for verifying the final reliability index of the spacecraft.

8.1.2.2 Reliability Allocation

The allocation of spacecraft system reliability indexes is to assign the defined reliability indexes to the specified levels (subsystem, equipment/components etc.) according to certain principles and methods. The reliability allocation and the reliability prediction are mutually iterated and constantly improved. The reliability prediction results can be used as a reference for reliability index allocation.

The purpose of reliability index allocation is to assign the quantitative reliability requirements to the specified lower product levels to ensure the realization of system reliability requirements.

The general principles for reliability index allocation are as follows:

1. The products with high complexity are assigned with low reliability indexes.
2. The products with mature technology and good inheritance are assigned with high reliability indexes.
3. The products in a harsh environment are assigned with lower reliability indexes.
4. High importance products are assigned with high reliability indexes.
5. The design optimization and comprehensive trade-off are carried out in consideration of other indexes.
6. After the reliability allocation, a certain margin must be reserved.

The steps of reliability index allocation are as follows:

1. Design (change) information: product design or design change.
2. Determine the index value to be assigned: the system reliability index value to be assigned.
3. Establish (modify) the system reliability model: establish the model according to the need for assignment, or correct the reliability model according to the design change.
4. Select an appropriate assignment method to allocate the indexes to each subsystem and equipment step by step: allocate the system reliability indexes to the required product levels.
5. Reliability indexes at all levels: obtain the reliability allocation results of the products at different levels.
6. Whether the user's requirement is met: analyze and compare the assigned reliability value with the user's requirement. If the requirement is met, the assignment will be stopped; otherwise, the reassignment will be carried out until the requirement is met.

The reliability assignment is needed in each stage of the spacecraft. In the conceptual design stage, the quantitative reliability requirements of each subsystem should be determined by system designer through the reliability allocation, and a certain margin should be left for the allocated reliability indexes. Then the subsystem reliability requirements are assigned to the equipment or components. In the prototype development stage, the assigned reliability value can be adjusted with the deepening and improvement of the design and the change of the reliability model and reliability prediction results. In the stage of flight model development, the implementation of assigned reliability value should be checked.

The reliability assignment methods commonly used for spacecrafts include weighted assignment method, expert score assignment method and minimum workload method.

8.1.2.2.1 Weighted Assignment Method (AGREE) This approach considers both the complexity and the importance of each subsystem. It applies to the electronic equipment which obeys exponential distribution and consists of K subsystems in series. The minimum mean time-to-failure of the i-th subsystem is shown in Equation (8.1):

$$\theta_i = \frac{N W_i t_i}{n_i \left[-\ln R_0\right]} \tag{8.1}$$

The reliability of the i-th subsystem is shown in Equation (8.2):

$$R_i\left(t_i\right) = \exp(-t_i / \theta_i) \tag{8.2}$$

where

t_i—Mission time of the i-th subsystem;

W_i—Weighting factor of the i-th subsystem, representing the probability that the failure of the i-th subsystem will lead to the failure of the system;

n_i—Number of components in the i-th subsystem;

N—Total number of components in the system;

R_0—System reliability index;

$R_i(t_i)$—Reliability assigned to the i-th subsystem;

θ_i—Mean time-to-failure assigned to the i-th subsystem.

8.1.2.2.2 Expert Score Assignment Method This method is based on human's experience to score the reliability by several factors and then assign a reliability index value to each subsystem according to the score.

Four factors are mainly considered (each with a score of 1–10):

a. Complexity. Each subsystem is scored according to the number of components that make up the subsystem and the difficulty in their assembly. The most complex subsystem is 10 points and the simplest subsystem is 1 point.
b. Maturity. Each subsystem is scored according to its current technical level and maturity. The lowest technical level is 10 points and the highest level is 1 point.
c. Working time. Each subsystem is scored according to its working time. 10 points are given to those that have been working all the time, and 1 point is given to those with the shortest service time.
d. Environmental conditions. Each subsystem is assessed according to its environment: 10 points are awarded to those that will experience extremely harsh environmental conditions in the operating process, and 1 point is awarded to those with the best environmental conditions.

Thus, the failure rate λ_i allocated to each subsystem is shown in Equation (8.3):

$$\lambda_i = C_i \lambda_0 \tag{8.3}$$

where C_i—scoring coefficient of the i-th subsystem, as shown in Equation (8.4);

λ_0—Failure rate required by the system.

$$C_i = \omega_i / \omega \tag{8.4}$$

where ω_i—Score of the i-th subsystem, as shown in Equation (8.5);

ω—System score, and $\omega = \sum_{i=1}^{n} \omega_i$.

$$\omega_i = \prod_{j=1}^{4} Y_{ij} \tag{8.5}$$

where Y_{ij}—score of the j-th factor of the i-th subsystem.

The score for each subsystem is given by expert scoring or by engineering team voting based on the practical knowledge and experience of the design engineer or reliability engineer.

8.1.2.2.3 Minimum Workload Method In this method, the system is assumed to be composed of n subsystems in series. This method requires a great improvement in the reliability of the subsystems with low reliability. If the estimated reliability values of n subsystems have been obtained and arranged in the non-reducing order $R_i\left(i=1,\ldots,n\right)$, that is, $R_1<\ldots<R_n$, then the reliability of the system will be $R_s=\prod_{i=1}^{n} R_i$.If the system reliability index is R_s^*, it indicates that the system reliability does not meet the index requirement when $R_s<R_s^*$, and that at least one R_i is needed to improve the system reliability to R_i^* enabling $R_s \geq R_s^*$. Therefore, a certain workload or cost, including the cost such as person, money, material, is required. If each subsystem has the same workload or cost function $F\left(R_i, R_i^*\right)$ which represents the workload or cost required for improving the subsystem reliability from R_i to R_i^*, then the total workload or cost required will be $\sum_{i=1}^{n} F\left(R_i, R_i^*\right)$.The minimum workload method is to ensure $\sum_{i=1}^{n} F\left(R_i, R_i^*\right)=\min(\text{minimum})$ under the condition of $\prod_{i=1}^{n} R_i^* \geq R_s$.

Then this optimization problem has a unique solution (Equation 8.6):

$$R_i^* = \begin{cases} R_0, \text{ if } i \leq k \\ R_i, \text{ if } i > k \end{cases} \tag{8.6}$$

Even if the reliability of the first k subsystems is improved from R_i to R_i^* (i.e., R_0), the reliability of other subsystems will remain unchanged. It can be seen that the improvement of weak system reliability links can minimize the workload or cost while meeting the requirement of system reliability index. In Equation (8.6), k is the largest j satisfying Equation (8.7):

$$R_j < \left(R_s^* \Big/ \prod_{i=j+1}^{n+1} R_i \right)^{1/j}, \ j=1,2,\ldots,\ n \tag{8.7}$$

where $R_{n+1}=1$. R_0 is Equation (8.8):

$$R_0 = \left(R_s^* \middle/ \prod_{i=k+1}^{n+1} R_i \right)^{1/k} \tag{8.8}$$

In this case, the system reliability meets the index requirement, that is $R_s = \prod_{i=1}^{n} R_i^* = R_s^*$.

8.1.3 Reliability Modeling and Prediction

8.1.3.1 Reliability Modeling

A system is an organic whole composed of several interacting and interdependent components with specific functions. The reliability model refers to the system reliability logic relationship and its mathematical model.

The reliability model of a spacecraft includes the reliability block diagrams and mathematical models of a system and its subsystems and functional components. The reliability block diagram should be consistent with functional block diagram and technical conditions. The product names and codes used in the reliability block diagram should be the same as those used in the functional block diagram and product specification.

8.1.3.1.1 Basic Reliability Models The basic reliability model includes a reliability block diagram and the corresponding reliability mathematical model. The basic reliability model is a series model in which the units in redundant or alternative operation modes are connected in series to estimate the maintenance and logistics requirements caused by the product and its constituent units.

8.1.3.1.2 Mission Reliability Model The mission reliability model includes a reliability block diagram and the corresponding reliability mathematical model. This model should be able to describe the intended use of each unit of the product during the mission completion. In the model, the units intended for redundant or alternative operation modes should be represented by a parallel structure. The names and logos of the product units used in the mission reliability model should be consistent with those used in the basic reliability model.

The basic reliability model can be used to estimate the mission reliability of a product only if the product has neither redundant nor alternative operation mode.

The spacecraft reliability modeling generally follows the following principles:

1. The reliability modeling shall be preceded by mission profile analysis and functional analysis to comb all the events and environmental profiles experienced by the product from delivery to EOL.
2. The reliability model shall be modified with the changes of development progress, mission requirements, operational constraints and product technology state and with other information obtained from the relevant tests, and shall pass the review.
3. The reliability block diagram shall correctly reflect the logical relationship between units, and shall be consistent with the functional block diagram and technical

conditions. The product names and codes used in the reliability block diagram shall be consistent with those used in the functional block diagram and product specification.

4. The system-level and subsystem-level reliability models shall be built to cover the equipment-level or independent functional units, and the equipment reliability models shall be built to cover at least the module circuits or components—and elements or parts if necessary.

8.1.3.2 Reliability Prediction

The reliability estimation is to estimate and predict the system reliability level by level based on the reliability data and models of the elements, components, equipment and subsystems that comprise the system.

The reliability prediction of a spacecraft system can be divided into basic reliability prediction and mission reliability prediction.

The basic reliability prediction is to add the estimated failure rates of all units in the basic reliability models of the spacecraft, subsystem and equipment levels, namely, full series models (all redundant or replacement equipment is treated as series connection).

The mission reliability prediction is to predict, according to the mission reliability models of the spacecraft, subsystem and equipment levels, the probability that the whole spacecraft, subsystems and equipment successfully accomplish the specified functions within the mission profile. Mission reliability prediction is a necessary task in spacecraft reliability engineering.

The general principles for spacecraft reliability projections are as follows:

1. For the reliability prediction of electronic products, the latest version of GJB/Z299 can be used for domestic components, and MIL-HDBK-217F can be used for imported components.

2. The reliability of a non-electrical product can be predicted according to its test information and reliability evaluation data, the information of similar products and the data of foreign non-electrical products.

3. An appropriate reliability prediction method should be selected according to the development stage and data of the product.

4. The mission reliability model of the product should be established to predict the mission reliability of the spacecraft, subsystem and equipment levels within the mission profile.

5. The reliability projection should start from the conceptual stage to determine the feasibility of the reliability index assigned to each functional level.

6. Once the design is changed in the development process, the reliability should be re-predicted. The early prediction focuses on the feasibility and reliability of the concept. With the deepening of design work, different prediction methods should be adopted.

7. For the functional levels operating in different conditions and environments, their reliability should be predicted under different conditions and environments.

The reliability prediction methods commonly used for spacecrafts include component counting method and stress analysis method.

8.1.3.2.1 Component Counting Method The component counting method is to add the failure rates of all the components contained in the equipment to obtain the failure rate of the entire product. The prediction is made from the bottom up, which is applicable to the early prototyping stage. The latest version of GJB/Z299 can be used for domestic components, and MIL-HDBK-217F[2] can be used for imported components.

8.1.3.2.2 Component Stress Analysis Method The stress analysis method for components is detailed in the latest GJB/Z299 version, and is generally applied in the prototyping stage. Its purpose is to determine the failure rate of equipment as a function of the failure rates of all components while considering the type, working stress level and derating characteristics of each component.

Basic steps of component stress analysis:

1. List all component types of the projected equipment down to the lowest component level specified in GJB/Z299 and MIL-HDBK-217F to facilitate the accurate use of failure rate models or the related diagrams and tables.
2. Count the number of components in each type and point out their role in the circuit (such as diode switch and voltage regulation).
3. Define the working environment and temperature T of the components, as well as the environmental coefficient.
4. Give the quality grade and quality coefficient of the components.
5. Give the electrical stress ratio S of the components.
6. Give various π coefficients and model parameters required by the prediction work.
7. Give (calculate) the basic failure rates λ_b of the components under the specified working conditions by table look-up or according to the basic failure rate model.
8. Calculate the failure rate λ_P of each component according to the stress analysis formula of the component.
9. Add the failure rates of all components in the predicted module/unit to obtain the total failure rate of the module/unit.
10. Calculate the reliability of the system and equipment according to the reliability mathematical model and mission time.

The emphasis of reliability prediction is different in each development stage of a spacecraft. The conceptual design stage is used to support the reliability index demonstration and concept optimization of the system and subsystems as well as the reliability design optimization of the equipment. The prototyping stage is mainly used to verify the system and subsystem design and support the reliability analysis, and to check whether the equipment design meets the specified reliability index requirements. When the technical condition is changed in the development stage of flight model, the reliability should be re-predicted to verify whether the reliability index meets the requirements.

8.1.4 Margin Design

Margin, also known as allowance, is a design allowance with respect to the index requirement, which is reserved to adapt to the boundary conditions, engineering implementation errors and other uncertain factors of a spacecraft product. It is an index describing the difference between the ultimate operating condition that can be achieved by a certain characteristic of a product and the actual condition. The margin is generally represented by percentage or absolute difference. The margin (allowance) should be managed and weighed at the highest level of a mission or system. If it is held locally or by a low level, it can cause manmade constraints or unproductive work and unnecessarily weaken the system's ability to achieve the mission objectives.

The design and analysis of spacecraft system margins (allowances) is to design the margins (allowances) according to the known system margin (allowance) requirements, and verify whether the design results meet the margin (allowance) requirements. The margin (allowance) design should focus on the key system characteristics and parameters that affect the mission success or spacecraft life, as well as the resources under overall control and allocation, and should check whether the margin (allowance) requirements are satisfied in the worst flight mode and under the limiting conditions of special environments.

The margin (allowance) requirements (such as mass, power, budget and schedule) of design resources and overall resources should be established early in the project. Part of the margin management plan is to use these margins to effectively solve problems and reduce risks. The workflow of spacecraft product margin design is shown in Figure 8.1.

The margin design items should be determined according to the spacecraft product type, mission characteristics and other specific information. The key characteristic parameters that affect the success of the mission should be included in the margin design items. After the analysis and evaluation, the designer should determine whether to design the margins of non-critical characteristic parameters.

The margin is generally achieved by decreasing the requirement value of an item and increasing its design value. The decrease of the requirement value usually requires the conceptual optimization of the product at the higher level, while the increase of the design value usually requires the increases of resource occupation.

In the prototype development stage, all the margin design results of the spacecraft product are verified. The verification of margin design results should meet the margin requirements in different environments and test conditions.

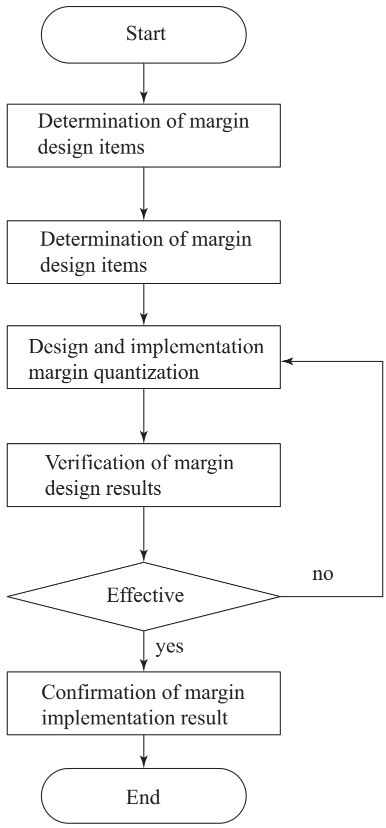

FIGURE 8.1 Workflow of spacecraft product margin design.

8.1.5 Derating Design

The derating design is intended to reduce the electrical, thermal and mechanical stresses of the parts so as to reduce their failure rates and improve the product reliability.

The main working procedures of spacecraft electronics derating include:

1. Determining the derating criteria

   Derating criteria are the design basis of derating. Generally, the component derating design is carried out in accordance with the relative national military standard. For the spacecraft parts used in special environments, the derating standards of enterprises are generally applied.

2. Determining the derating level

   The level-I derating is generally required for spacecraft parts and may be degraded to the level II in special cases, where the corresponding approval procedures should be handled.

3. Determining the derating parameters

   The derating parameters of spacecraft parts mainly include voltage, current, power consumption, temperature and mechanical stress. The derating under transient stress and fault state is also considered. The starting points for voltage and current derating are the rated values or the absolute maximums. The operating voltage after derating is generally not lower than the voltage recommended by the manufacturer or the test supply voltage given in the product manual (when these data are used). The starting point of power derating is the rated power consumption under the specified operating conditions. Power consumption, heat dissipation conditions and temperature are closely related to each other. Temperature and temperature-related stress are the focus of derating design. For the electronic products with frequent starts and stops, the temperature change of the main heating components should be controlled to reduce the failure rate caused by thermal fatigue. The mechanical stress derating mainly considers the influence of vibration. Transient stresses should be derated in accordance with the specified derating factor. When no allowable transient stress value is provided, the derating shall be carried out according to the provisions of special technical document. The components (such as current-limiting resistors) used for fault isolation shall be derated under fault conditions. The derating factor shall be specified under the specific technical conditions.

4. Calculation of derating factor

   The derating factor is the ratio (difference) between the operating stress and rated stress of a part. Stress ratio is generally calculated for electrical stress parameters and temperature difference is generally calculated for thermal stress parameters. Before the concrete calculation, the temperature value and the electrical stress value should be obtained by electrical/thermal stress analysis and calculation or testing.

5. Derating design verification

   The derating parameters and derating factors should meet the requirements of the derating criteria after the derating design is complete.

8.1.6 Fault Tolerant Design

Fault tolerance refers to the technology of a system tolerant to faults. In other words, it is a technology with which the system in operation can automatically detect and diagnose the faults or errors in one or more of its key parts and take appropriate measures to maintain its specified function or keep its function within an acceptable limit. The fault tolerance methods can be divided into passive fault tolerance and active fault tolerance. The purpose of passive fault tolerance method is to reduce the dependence of the system on the operation of a single component, so that the system can still work even in the case of failure without correction (i.e., fault shielding). The method of active fault tolerance should first detect and diagnose the system faults automatically and timely, and then take measures to control or handle the faults. Therefore, the active fault tolerance usually includes fault detection and isolation, and system reconfiguration design.

Fault tolerance design is an important part of spacecraft concept design and detailed prototype design. Figure 8.2 shows the workflow of fault tolerance design for a spacecraft system. First, the relevant requirements of the spacecraft should be defined in the conceptual design stage, and then the basic function design should be carried out by using the

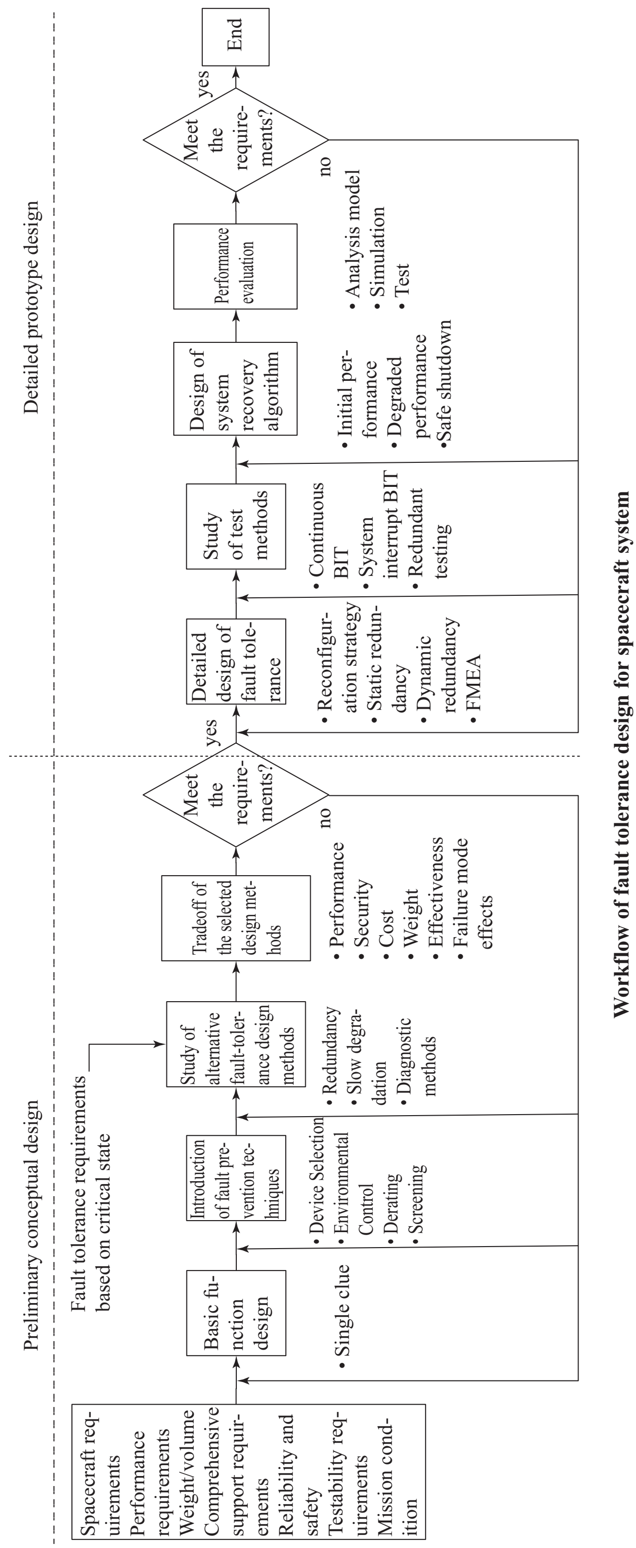

FIGURE 8.2 Workflow of fault tolerance design for spacecraft system.

methods such as environmental control, derating, screening and component selection to improve reliability and prevent the faults. In order to ensure the system safety, the fault prevention design should be followed by fault tolerance design, in which different fault tolerance schemes (such as redundancy, diagnosis management and slow degradation) should be proposed and weighed to find the optimal scheme. In the detailed prototyping stage, the detailed fault tolerance design should be carried out to define the strategy of system reconfiguration and the static and dynamic redundancy methods. For each fault tolerance design, the detection method and the algorithm of system recovery should be determined, and the design effectiveness should be evaluated.

At the subsystem or system level, fault tolerance design is mainly realized by functional redundancy, i.e., reconfiguration and operating mode redundancy. At the equipment level, the common fault tolerance techniques include information fault tolerance, time fault tolerance, hardware (structure) fault tolerance and software fault tolerance. Information fault tolerance is to detect or correct the information errors in the operation or transmission by adding some information. Time fault tolerance is a means to sacrifice time for the high reliability of a computing system. It is mainly to reduce the machine speed by a limited extent to increase the system reliability or to eliminate the impact of instantaneous errors by the repeated execution of instructions or programs. Hardware (structure) redundancy is direct redundancy, mainly including static redundancy, dynamic redundancy and mixed redundancy. Software fault tolerance is to improve the software reliability through the program addition. The programs to be added include the add-on program for error detection and diagnosis, the program for automatic computer system reorganization and degraded running and the programs of different versions independently written in different languages or ways.

8.1.7 FMEA

The purpose of FMEA is to identify all possible failure modes of product design or process through systematic analysis and induction, analyze the effects of and causes for failure modes, detect potential weak links and propose possible preventive/corrective measures and in-orbit compensation measures to reduce the severity and/or occurrence probability of failures.

FMEA is an important item of reliability analysis. The output results of FMEA can provide a basis for the qualitative verification of product reliability, the listing of key items and the process control, and provide information for determining the reliability test items, failure plan and safety design, and testability design.

The FMEA workflow for spacecraft products is shown in Figure 8.3. The “analysis condition review” is mainly to review the input information preparation, product definition, functional block diagram, reliability block diagram and relevant conventions and assumptions required for analysis, and examine whether the conditions for analysis are met. The “identify weak links through statistical induction and analysis” mainly emphasizes the statistics, retrieval and induction of the products at all levels in the process of analysis according to the requirements. The subsystem FMEA can not be a simple addition of equipment FMEAs, and the system FMEA can not be a simple addition of subsystem FMEAs.

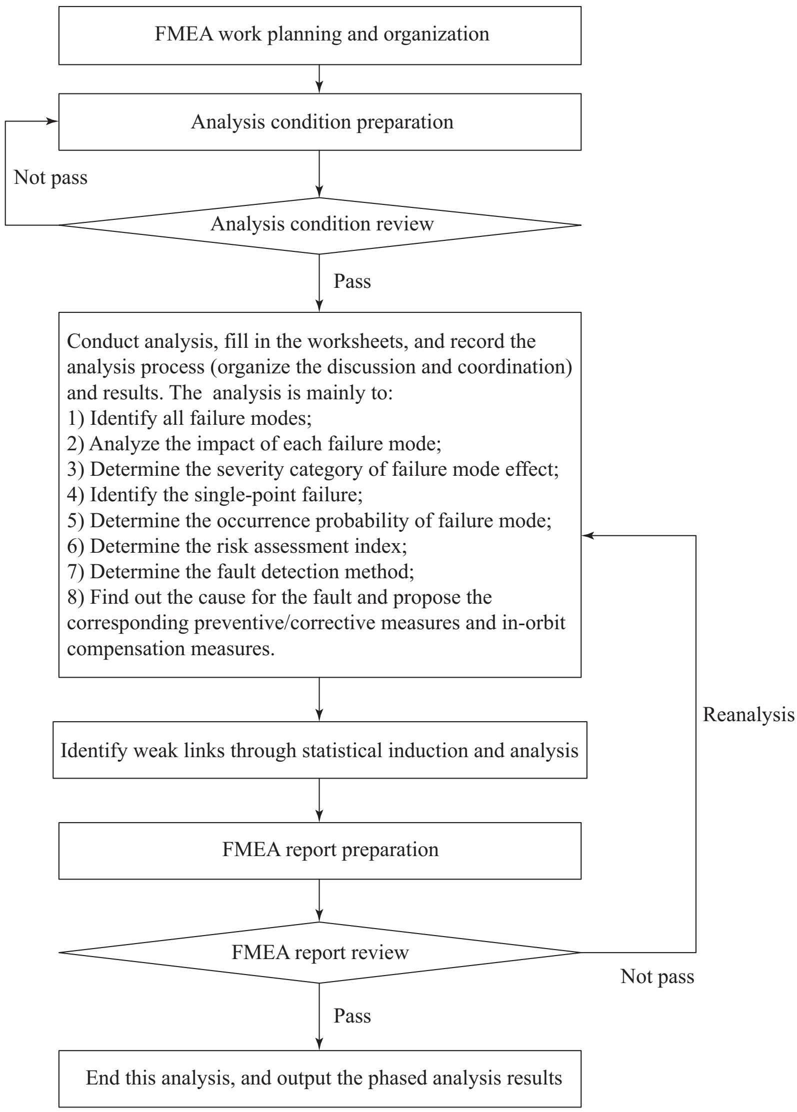

FIGURE 8.3 FMEA workflow of spacecraft products.

The FMEA of a product should use the FMEA of the next-level product as input, which, however, must be combined with supplementary analysis and induction to finally get the failure mode of each product level. The "end this analysis, and output the phased analysis results" emphasizes that the FMEA report is dynamic and should be re-analyzed through design improvements. The FMEA can achieve the goal of continuous improvement only by realizing dynamic management and closed loop management. After the worksheet is filled in and analyzed, the measures to be implemented shall be summed up and included in the control tracking form, and shall be managed and assessed by special personnel.[3]

The FMEA focuses on the analysis of spacecraft products at all levels, and can go up or down from any product level. The FMEAs at different levels are neither completely

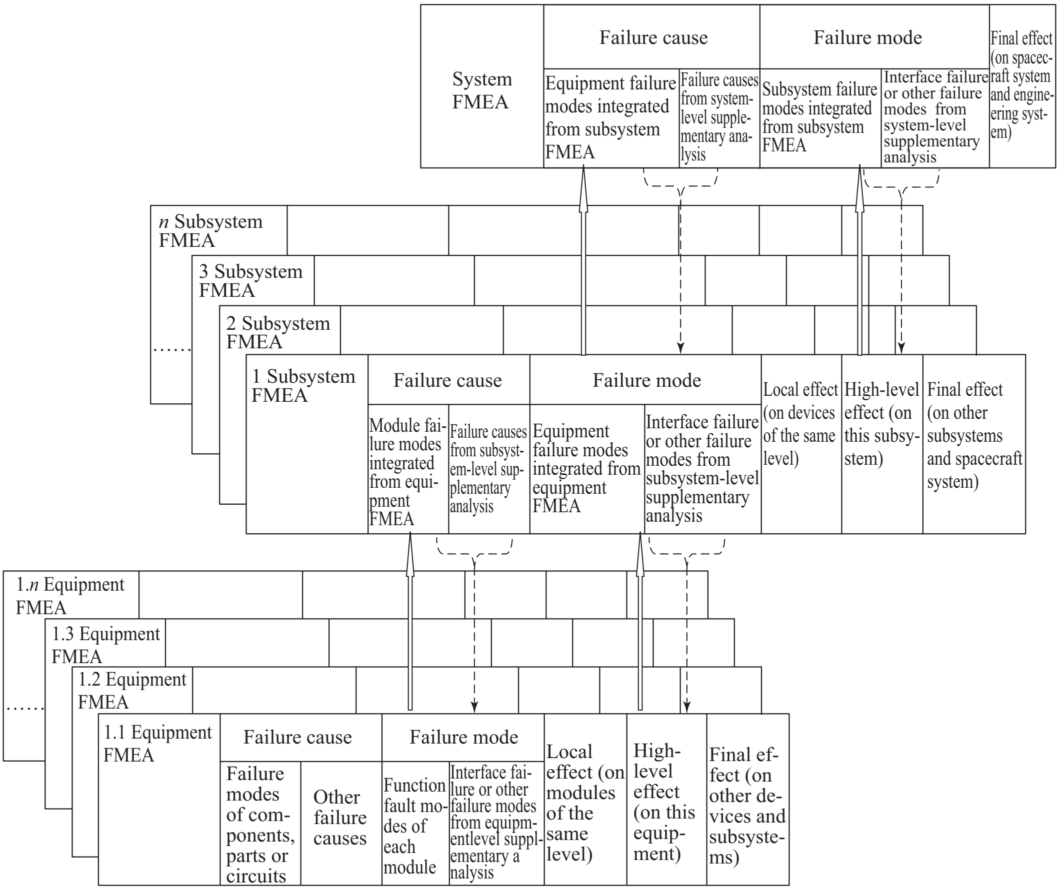

FIGURE 8.4 Relationship between spacecraft product FMEAs.

independent nor dependent on each other. Information is transferred among the failure cause, failure mode and high-level influence. The information transfer relationship among spacecraft system FMEA, subsystem FMEA and equipment FMEA is shown in Figure 8.4. In bottom-up analysis, the fault mode of the lower level product is passed up and becomes the fault cause of the higher level product, and the high-level influence of the lower level product is passed up and becomes the fault mode of the higher level product. In top-down analysis, the fault cause of the higher level product is passed down and becomes the fault mode of the lower level product, and the fault mode of the higher level product is passed down and becomes the high-level influence of the lower level product.

The worksheet for spacecraft product FMEA at each level is shown in Table 8.1.

8.1.8 FTA

The FTA is to analyze an undesirable system fault event (i.e., top event), identify the necessary and sufficient direct causes for the occurrence of the fault event through strict top-down layer-by-layer causal logic analysis, finally find out all the causes and cause combinations leading to the occurrence of the top event, and obtain through calculation

TABLE 8.1 Spacecraft Product FMEA Worksheet

						Failure Effect											
No.	Item Name	Functional Description	Failure Mode	Code	Mission Phase or Operation Mode	Local Effect	High-level effect	Final effect	Severity Category	Occurrence Probability	Risk Assessment Index	Single-point Failure	Fault Detection Method	Failure Cause	Preventive/Corrective Measures	In-orbit compensation measures	Remarks

the quantitative indexes (such as the occurrence probability of top event and the importance of bottom events) when the probability data of bottom events are available.

The FTA process is divided into five steps: (1) selecting the top event; (2) identifying the boundary conditions; (3) building the fault tree; (4) qualitative analysis and analysis result application; and (5) quantitative analysis and analysis result application.

8.1.8.1 Selecting the Top Event

In the product development stage, the failure modes of critical events in the support chain of spacecraft flight events and the failure modes of key subsystems that affect the success of the model should be taken as top events for FTA.

For example, the goal of the first project in the second phase of China's lunar exploration is to "break through the lunar soft landing technology". The first task of chang'e-3 probe is just safe soft landing, so the top event is defined as "probe soft-landing failure".

8.1.8.2 Identifying the Boundary Conditions

The boundary conditions include initial equipment state and assumed system inputs. The fault tree describes the performance of a system at the specified time, in the specified state and under the specified boundary conditions.

The analysis of the bottom event in system-level fault tree should consider the coupling and interface relationship between subsystems, and analyze down to the redundant units of the system. The analysis of the bottom event in subsystem fault tree should consider the coupling and interface relationship between devices, and analyze down to the independent function devices of the subsystem. The analysis of the bottom event in device (equipment) fault tree should consider the internal interface relationship of the device and analyze down to the independent function modules and components (if necessary) of the device.

The conditions assumed in the FTA of probe soft-landing failure are as follows: the soft-landing FTA only considers the operating process at the powered descent stage, and the probe is assumed to work normally without any fault in each mission phase before the powered descent.

The analysis of failure causes only considers the failures of the equipment in the subsystems related to soft landing, such as control subsystem, propulsion subsystem, landing buffer subsystem and data management subsystem. Other support equipment, such as the equipment in power supply subsystem and thermal control subsystem, is assumed to work properly. When analyzing the faults in the components of landing buffer subsystem, only the product design faults should be considered, and the effects of other conditions (such as landing speed) on the subsystem should be considered during the analysis of the whole soft landing mission.

8.1.8.3 Building the Fault Tree

The main work of building a fault tree is to proceed from the top event to find the intermediate events and the bottom event, and then choose an appropriate logic gate to connect these events.

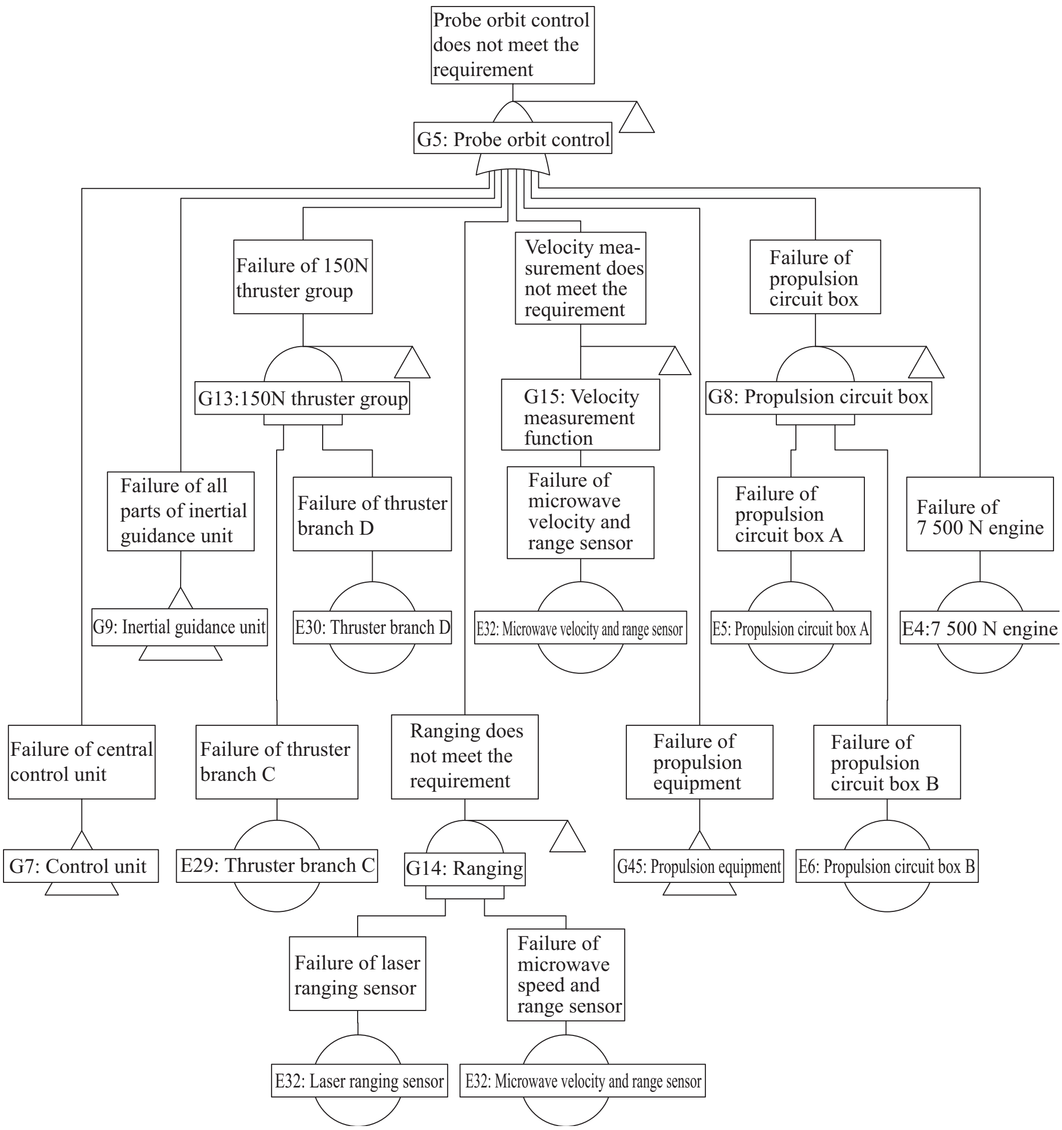

FIGURE 8.5 "Probe orbit control does not meet the requirement" fault tree.

The fault tree can be built with the aid of reliability software. Here, the subtree that the probe orbit control does not meet the requirement is selected as an example and is expanded downward (as shown in Figure 8.5).

8.1.8.4 Qualitative Analysis and Analysis Result Application

The qualitative analysis of a fault tree is to obtain all the minimum cutsets. Its basic purpose is to identify all possible system failure modes that lead to the top event. The entire "probe soft-landing failure" fault tree consists of 94 bottom events. See Table 8.2 for the list of first-order minimum cutsets, and Table 8.3 for the list of second-order minimum cutsets.

TABLE 8.2 List of First-order Minimum Cutsets of “Probe Soft-landing Failure” Fault Tree

No.	Bottom Event Number	Description of Bottom Event	Subsystem
1	E4	Engine failure	Propulsion subsystem
2	E7	Failure of high-flow self-locking valve in liquid path	Propulsion subsystem
3	E23	Failure of metal diaphragm tank	Propulsion subsystem
4	E24	Failure of high-pressure self-locking valve	Propulsion subsystem
5	E25	Failure of pressure-reducing valve	Propulsion subsystem
6	E27	Air cylinder failure	Propulsion subsystem
……			
27	E71	Failure of low-flow self-locking valve GV1	Propulsion subsystem
28	E72	Failure of low-flow self-locking valve GV2	Propulsion subsystem

TABLE 8.3 List of Second-order Minimum Cutsets of “Probe Soft-landing Failure” Fault Tree

No.	Bottom Event Number	Description of Bottom Event	Subsystem
1	E55	Abnormal data management instruction after receiving the CCU shutdown request	Data management subsystem
	E56	Abnormal communication between CCU and propulsion circuit box	Interface between GNC subsystem and propulsion Subsystem
2	E36	Failure of unit A of image processing board	GNC subsystem
	E37	Failure of unit B of image processing board	GNC subsystem
3	E5	Failure of propulsion circuit box A	Propulsion subsystem
	E6	Failure of propulsion circuit box B	Propulsion subsystem
……			
10	E78	Failure of low-flow self-locking valve LV9	Propulsion subsystem
	E79	Failure of low-flow self-locking valve LV11	Propulsion subsystem

8.1.8.5 Quantitative Analysis and Analysis Result Application

The quantitative analysis of fault tree mainly includes calculating the occurrence probability of top event and the importance of bottom events.

The occurrence probability of top event can be calculated with two methods: accurate calculation and approximate calculation. In engineering, accurate calculation is not necessary, because both the calculation workload and the error of occurrence probability of a bottom event are too large. The common approximate calculation methods of the minimum cutsets include the subitem approximate calculation based on inclusion-exclusion principle and the first-term approximate calculation based on inclusion-exclusion principle. The list and importance of bottom events in the “probe soft-landing failure” fault tree are shown in Table 8.4.

8.1.9 ETA

ETA is a method that, according to the time sequence of accident development, starts from the initial event to deduce and analyze the success rate of each event in the development process so as to identify the risks. The idea of this method is to discretize the real event

TABLE 8.4 List and Importance of Bottom Events in the "Probe Soft-landing Failure" Fault Tree

No.	Bottom Event Number	Description of Bottom Event	Subsystem	Number of Occurrences	Structural Importance
1	E4	Engine failure	Propulsion subsystem	6	4.9362425e-5
2	E32	Failure of microwave velocity and range sensor	GNC subsystem	10	4.9362425e-5
3	E5	Failure of propulsion circuit box A	Propulsion subsystem	5	1.64541417e-5
4	E6	Failure of propulsion circuit box B	Propulsion subsystem	5	1.64541417e-5
……					
55	E29	Failure of thruster branch C	Propulsion subsystem	5	1.64541417e-5
56	E30	Failure of thruster branch D	Propulsion subsystem	5	1.64541417e-5
……					

evolution into some macro events, and then quantify the occurrence probability of the obtained accident sequence.

The events constituting an accident sequence generally fall into the following three categories: (1) the action of protection system (the measures for eliminating the faults) or not (system event tree); (2) the implementation of safety function or not (function event tree); and (3) the occurrence of physical phenomena or not (phenomenological event tree).

The system event tree is used to determine the sequence of accidents in the devices and associated protection systems. The phenomenological event tree describes the evolution process of equipment accidents (such as fire and pollutant diffusion).

8.1.9.1 Event Tree Construction

An event sequence diagram (ESD) is a "success-oriented" graphic representation because it considers how to prevent the accident occurrence or mitigate the accident severity through human behavior and system responses (including software). It is essentially a flow chart, in which different paths lead to different consequence states. Each path in the flowchart is a chain of events. The occurrence of intermediate events along any path should be identified.

A typical ESD is shown in Figure 8.6.

The ESD can be mapped into an event tree. The event tree sequence is a part of Boolean logic. The purpose is to establish a tractable model of the important path from the initiating event to the consequence state, in order to systematically quantify the risks. Similar to an ESD, an event tree also starts with an initiating event and progresses through a chain of intermediate events, either successful or failed (also called critical events or top events), until it achieves the consequence state (see Figure 8.7).

8.1.9.2 Fault Modeling

The fault modeling is mainly to establish a fault tree model based on the initiating event and intermediate events of an event tree. These events often involve software reliability, artificial reliability, common cause failures and environmental impacts.

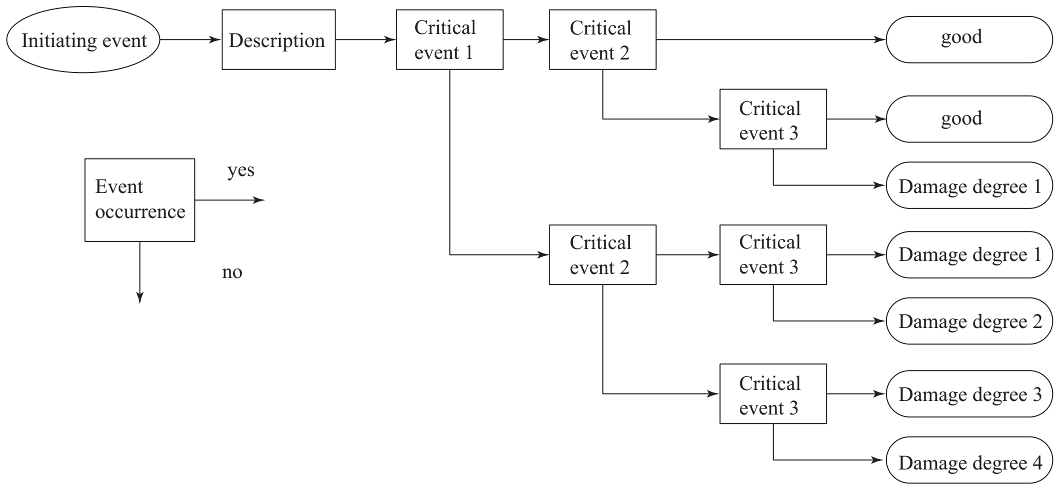

FIGURE 8.6 Typical ESD.

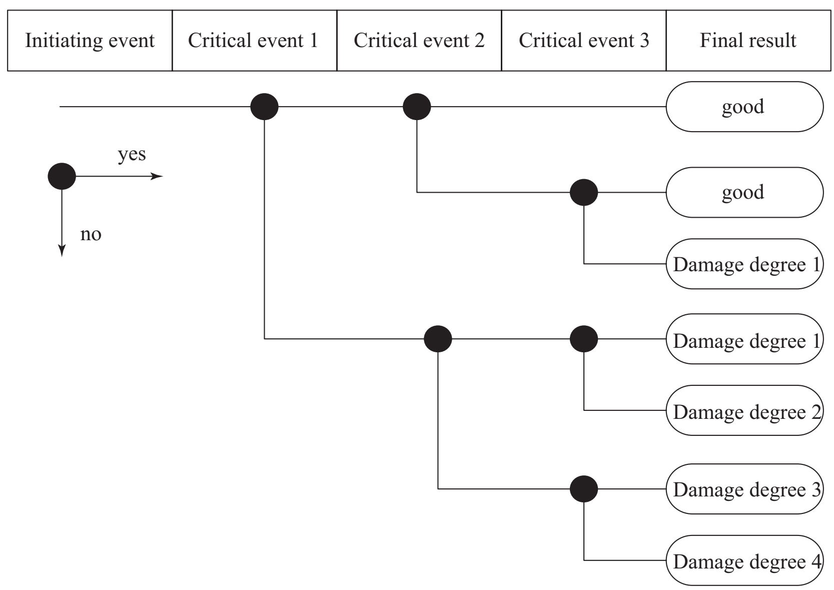

FIGURE 8.7 Typical event tree diagram.

In an event chain, no further fault modeling and analysis is needed for simple intermediate events (or initiating events) whose occurrence probability can be obtained directly. However, for complex intermediate events (or initiating events) whose occurrence probability can't be obtained directly, further modeling analysis is required to determine the probability of "success" or "failure" of these events. The models and methods used include fault tree, dynamic fault tree, Bayesian network, Markov chain and reliability block diagram. Among these modeling methods, fault tree is one of the most commonly used methods to link the fault information at "component level" with the fault information at "system level".

8.1.9.3 Quantitative Analysis

The occurrence probability of each consequence state is quantified by logically associating the fault trees that occur in each chain of events. The occurrence frequency of each consequence state in the event tree is the product of the conditional probability of the initiating event and the conditional probabilities of the intermediate events in the event chain linking the initiating event to the consequence state. The event chains are grouped according to their final results. The frequency of the consequence state represented by a group can be obtained by logical accumulation of the frequencies in the group.

8.1.9.4 Importance Calculation

The importance of each event is calculated according to the severity of the final result and the risk probability of the process. It should be noted that an event (such as failure of the component X) may occur in many low-frequency links, but may not occur in the main risk event chains. However, if the low-frequency links contribute to the overall risk to a considerable extent, the event itself will also have a high degree of importance. The calculation result and process of importance can be used to make multi-risk decisions (such as resource allocation), and provide a basis for the risk-reducing improvement measures, such as redesigning the hardware and adding the redundancy.

8.1.10 PRA

PRA is a comprehensive and structured logical analysis method to identify and assess the risks of complex technical systems. At the same time, FMEA, main logic diagram, ESD, FTA, ETA and other analysis tools are used. PRA has become the primary technical approach for identifying and analyzing the technical and safety risks associated with complex systems, projects and programs.

The PRA process identifies the system weaknesses that often have an adverse impact on system safety and performance, and mission success. The PRA result information provides suggestions on feasible RM strategies that can reduce risks and provides the decision makers with the cost-benefit areas where the improvements of design and application are the most cost-effective.

8.1.10.1 PRA Classification

1. Full-scope PRA: The full-scope analysis includes all PRA processes. The decisions on complex system items need to be supported by a full-scope PRA, while taking into account the uncertainties.

2. Limited-scope PRA: The application steps of a limited-scope PRA, except for the concerns, are the same as those of a full-scope PRA. The limited-scope PRA is only concerned about the consequence states of important decisions related to missions, rather than all the consequence states.

3. Simplified PRA: The steps of a simplified PRA are consistent with that of a full-scope PRA, except for the identification and quantification of significant mission

risk contributions. The simplified PRA is usually used in a system with relatively small technical complexity, or in a system where the available design data is less than that required to carry out a full-scope PRA. The simplified PRA consists of reduced or simplified event chains that take into account only the significant mission risk contributions.

8.1.10.2 PRA Process

The implementation process and basic method of PRA are shown in Figure 8.8.

8.1.10.2.1 Definition of Objective(s) As a first and most important step, the PRA objective should be clearly defined. Without this step, the rest of the assessment would be both incomplete and inadequate. The definition of risk assessment objective should relate to the unintended consequences of the benefit (called "consequence states"). These consequences include injury to personnel (for example, injury, illness or death), mission capability degradation, mission failure, loss of property or other consequences.

It is necessary to appropriately define the composition and lifetime profile of the analytic target as well as the rule considering the initiating events (i.e., whether external events such as micrometeors are included) depending on the PRA implementation scope.

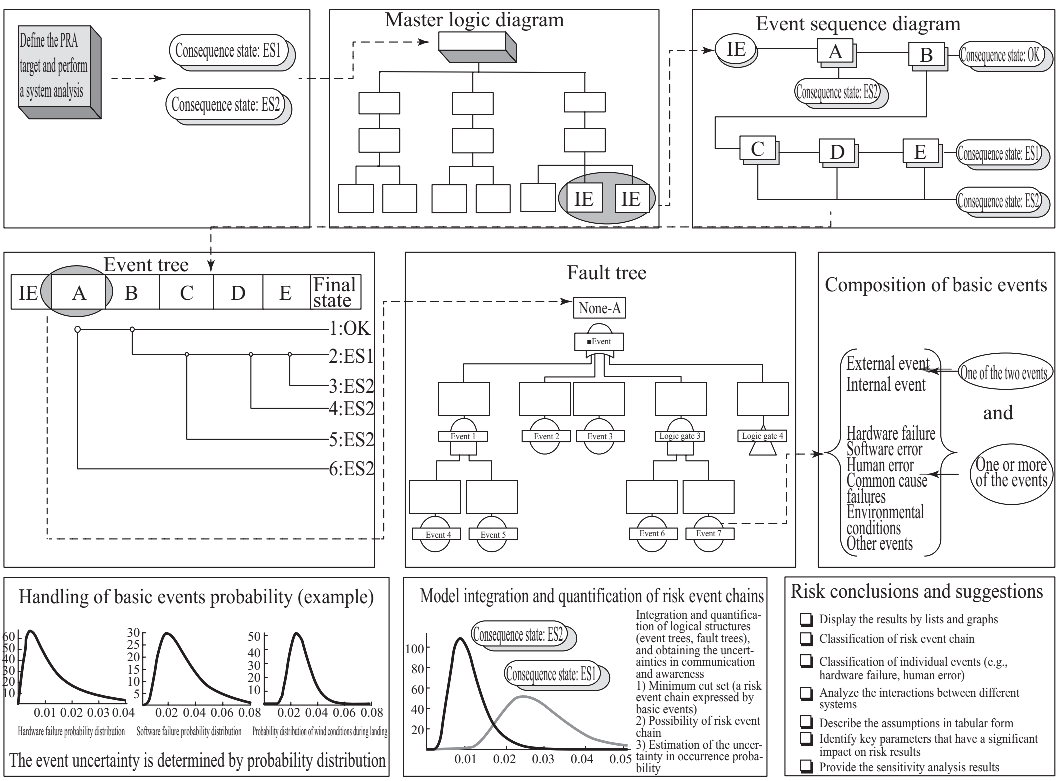

FIGURE 8.8 Schematic diagram of PRA implementation procedure and basic method.

8.1.10.2.2 System Familiarization Knowing well the system is the second step. This step includes a review of all information on design and use, including design and/or machining drawings, as well as operational, emergency and maintenance procedures. If PRA is implemented for an existing system that has been in operation for some time, this engineering information should be based on design state and use state; If PRA is performed for a new or demonstrated system, then the system in the design state will be the basis of the application. The goal of this step is to make the analyst thoroughly familiar with the system and its design and/or use, and to identify the success criteria and unintended consequence states of the target mission.

Initiating event is the beginning of a chain of events, and can trigger a subsequent series of events. Through identification, analysis and filtering, the designer can determine whether the initiating event has the potential to trigger a chain of events that can evolve into a defined consequence state. The initiating event leads to a chain of events, among which the events that have the same consequence state but are less likely to occur can be eliminated. The identification of an initiating event can be accomplished by a special type of superior logical tree. It's called master logic diagram. Other related technologies, such as FMEA, can also be used to identify an initiating event.

8.1.10.2.3 Scenario Modeling The PRA should identify and estimate the unintended consequences of a potential event chain. Modeling each sequence of events is an inductive process, called event tree, which usually involves drawing and logic tools/techniques. An event tree starts with an initiating event and progresses through a chain of intermediate events, either successful or failed (also called critical events or top events), until it achieves the consequence state.

8.1.10.2.4 Failure Modeling The PRA should estimate the failure (type and occurrence likelihood) of each event in the event chain identified above, and model the events by means of the fault tree.

8.1.10.2.5 Quantification The PRA should quantify the event chain. Quantification is the process of evaluating the occurrence probability and consequence of an unintended consequence state. The occurrence probability of each consequence state is quantified by associating the fault trees that occur in each chain of events. The occurrence frequency of each consequence state in the event tree is the product of the conditional probability of the initiating event and the conditional probabilities of the intermediate events in the event chain linking the initiating event to the consequence state. The event chains are grouped according to their final results. The frequency of the consequence state represented by a group can be obtained by logical accumulation of the frequencies in the group.

8.1.10.2.6 Uncertainty Analysis Since the PRA attempts to model uncertain events (the variability reflected by events cannot be estimated), the risk model is basically a model for uncertainty analysis. The PRA suggestions for decision makers should include a correct evaluation of all degrees of freedom regarding the result uncertainty and analyze which uncertainty resource is critical.

8.1.10.2.7 Sensitivity Analysis Sensitivity analysis is a type of uncertainty analysis that focuses on modeling the uncertainties in assumptions, models and bottom events. These analyses are often performed in PRA to evaluate which numerical changes in analysis inputs or components can lead to the greatest changes in local or final risk results. The purpose of sensitivity analysis is to assess the result changes caused by the changes in assumed input parameters. This type of analysis is typically performed to determine which parameters in the PRA are the most important. It deserves the greatest attention or requires improvement.

8.1.10.2.8 Ranking One of the important results of PRA is the importance of the calculated risk. Therefore, special importance measures, such as Fussell-Vesely, risk reduction value, Birnbaum, risk completion value and variance, are used to determine the guiding significant contribution factors of the risks in a sequence or chain of events. These contribution factors are arranged from largest to smallest. This sequence is called importance sequence.

8.1.10.2.9 Data Analysis The PRA should carry out the data analysis to support quantification. Data analysis is the process of collecting and analyzing information in order to evaluate different parameters of the PRA model. These parameters are used to obtain the occurrence probabilities of different events, including component failure rate, initiating event frequency and personnel and software failure probabilities.

8.1.11 Sneak Circuit Analysis

The sneak circuit analysis technology can identify the sneak problems in the spacecraft design during the product design stage, and take targeted measures to improve the product reliability and safety.

8.1.11.1 Basic Concept and Characteristics of Sneak Circuit

Sneak circuit is a kind of latent state inadvertently introduced to the design of an electrical or electronic system. It can cause the production of an undesired system function or the suppression of a desired function under specific conditions.

Sneak circuit is an inherent design state, which is hidden in the normal design function and is not recognized by the designer. In general, a sneak circuit has nothing to do with the failure of components themselves. It is not caused by the failure of components or equipment in the system. Instead, it is a kind of design defect introduced by designers due to the lack of overall grasp and understanding of a complex circuit system. That is to say, even in the case of no component failure, some excitation conditions may still cause abnormal system function.[4]

A sneak circuit is often latent, sudden, difficult to find and easy to correct. It generally has sneak paths, sneak timing, sneak indicators and sneak labels:

1. Sneak paths: an undesired path of electric current, that is, a current path leading to an undesired system function or suppressing a desired system function.
2. Sneak timing: an abnormal system state resulting in the occurrence of electric current and energy signals in an unexpected or contradictory time sequence, or at an unexpected moment, or in an unexpected period of time.
3. Sneak indicators: a vague or incorrect indication of the system operation state. A sneak indicator may mislead the system or the operator into an unexpected response.
4. Sneak labels: an incorrect or inexact label of system function. A sneak label may mislead the operator.

8.1.11.2 Sneak Circuit Analysis Methods

The sneak circuit analysis methods mainly include the "classic method" based on network tree generation and topology pattern recognition and the "simplified method" based on function node recognition and path tracking.

8.1.11.2.1 Analysis Method Based on Network Tree Generation and Topology Pattern Recognition

This method is first to divide the system appropriately and simplify the structure to generate a network tree; then identify all the topology patterns in the network tree; finally, analyze the network tree in accordance with the clue list to identify all the sneak states in the system. In this method, the system can be comprehensively and thoroughly analyzed, so the analysis has good integrity but a large analysis workload.

Network tree is a tree network structure obtained after the division and simplification of a circuit system, indicating the connection relationship between the interconnected components. The network tree is generated according to the division principle. The "division" here refers to the division of circuit topology structure, rather than the division of circuit function. Its purpose is to divide a network of interconnected cables into blocks. The general division approach is to use all levels of power supply busbars and return busbars in the electrical network diagram as the dividing points to divide the system into many separate and disconnected circuit blocks, which are called "network trees".

All circuits can be summarized into five basic circuit topology patterns, namely, the single line topology, the power dome, the ground dome, the combination dome and the reverse-current dome (as shown in Figure 8.9).

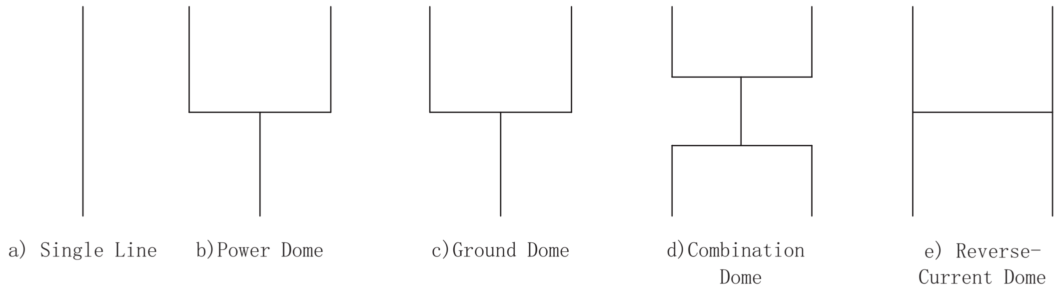

FIGURE 8.9 Basic circuit topology patterns.

Because the behaviors in basic circuit topology patterns are easy to know, the identification of all the basic topology patterns from the network tree enables the analyst to easily identify all the behaviors of the circuit system and thus identify the sneak circuits in the system.

The cue list is a series of suggestive question libraries that can be used to fully identify sneak circuit states in the network tree. The clue list is very important for sneak circuit analysis. It raises a variety of suggestive questions to expand the analyst's understanding of the circuit, and finds sneak problems through the answers to suggestive questions.

8.1.11.2.2 Analysis Method Based on Functional Node Identification and Path Tracking

This method is first to divide and simplify a complex system, and then identify the function nodes in the system, track the function paths and finally analyze the paths in accordance with the clue list. It can be regarded as a simplified method based on topological pattern recognition. In other words, it analyzes only the paths between the specified source and the target and the combination of these paths to identify sneak paths.

The function nodes can be divided into sources and targets. The sources are the signal and data sources (such as the AC and DC power points of the circuit system to be analyzed) of the system performing its expected functions. The targets are the execution units or key components (such as a functional signal executor) of the circuit system performing its intended function. The function node identification should identify the system operation mode and switching device state, and then identify the targets and sources according to the system function analysis.

Path tracking is to identify all possible paths between the source and the target, assuming that all switching devices in the system are closed. The tracking usually starts from the power supply point or signal source, and ends at the final function execution device or the low-level power supply point or the current/signal return point.

Path analysis is to analyze all the above tracked current paths in accordance with the list table to find sneak paths and sneak timing sequences. The analysis should delete the paths that are logically self-contradictory or impossible to exist in terms of time sequence or state, and verify whether each possible path will lead to undesired functions or suppress the desired functions, so as to obtain sneak analysis results.

8.1.12 Worst Case Analysis

The worst cases refer to the extreme cases (and their combinations) of environmental changes (temperature, humidity, radiation, vibration etc.), component parameter drifts (component quality level, component aging etc.) and input drifts (power supply, excitation) experienced by the circuit within the design limits in the mission profile.

WCA is a method of analyzing the circuit performance parameter deviation under the worst combination of circuit component parameters. The methods of WCA include extreme value analysis, root square sum analysis and Monte Carlo analysis.

8.1.12.1 Extreme Value Analysis

Extreme value analysis is the analysis of circuit output performance when all the variables in the circuit are set as the worst values. It uses the known variation limits of component parameters to predict whether the circuit parameter variation exceeds the allowable range, and then provides the direction of improvement. If the predicted value is within the specified range, the circuit will be stable and reliable. If it goes beyond the specified allowable range, the drift fault may occur. This analysis method is simple and intuitive, but the obtained results are conservative.

To apply the extreme value analysis method, a mathematical model is established and the circuit performance parameter y to be analyzed is expressed as a function of the design parameters $x_1, x_2, \ldots x_n$, i.e.,

$$y = f(x_1, x_2, \ldots x_n) \tag{8.9}$$

Then the sensitivity s_i of the circuit performance parameter y to the design parameter x_i is the partial derivative of the function f with respect to x_i, i.e.,

$$s_i = \frac{\partial f}{\partial x_i} \tag{8.10}$$

where each parameter $x_i\,(i = 1,\ldots, n)$ is a nominal value.

After determining the sensitivity, the maximum deviations of the performance parameters can be calculated by direct substitution and linear expansion.

8.1.12.1.1 Direct Substitution

The upper and lower limits of the circuit performance parameter are obtained by directly substituting the limits of the design parameters into the function expression (Equation 8.9) of the circuit according to the worst case combination.

When calculating the maximum value $y_{\max}$ of circuit performance parameter, the parameter x_i will reach the maximum value with $s_i > 0$ and the minimum value with $s_i < 0$.

When calculating the minimum value $y_{\min}$ of circuit performance parameter, the parameter x_i will reach the minimum value with $s_i > 0$ and the maximum value with $s_i < 0$.

If the maximum and minimum values of circuit performance parameter calculated in the worst case are within the range of the specified circuit performance deviation index, the circuit should have passed the extreme value analysis.

8.1.12.1.2 Linear Expansion Through the Taylor expansion of the analyzed circuit performance equation $y = f(x_1, x_2, \ldots x_n)$ near the operating point and the omission of the first-order and higher order terms, the linear relationship between the variation Δy of the performance parameter y and the variation Δx_i of the design parameter x can be obtained:

$$\Delta y = \sum_{i=1}^{n} \frac{\partial f}{\partial x_i} \Delta x_i = \sum_{i=1}^{n} s_i \Delta x_i \tag{8.11}$$

When calculating the positive limit Δy_+ of circuit performance parameter deviation, Δx_i can be determined by $\Delta x_i = x_{i\max} - x_{i\text{nom}}$ given $s_i > 0$, and by $\Delta x_i = x_{i\min} - x_{i\text{nom}}$ given $s_i < 0$. When calculating the negative limit Δy_- of circuit performance parameter deviation, Δx_i can be determined by $\Delta x_i = x_{i\min} - x_{i\text{nom}}$ given $s_i > 0$, and by $\Delta x_i = x_{i\max} - x_{i\text{nom}}$ given $s_i < 0$.

Therefore, the maximum value of y in the worst case is $y_{\max} = y + \Delta y_+$, and its minimum value in the worst case is $y_{\min} = y + \Delta y_-$. If both the maximum and minimum values of circuit performance parameter in the worst case meet the requirements of circuit performance index, it indicates that the circuit has passed the extremum analysis.

8.1.12.2 Root Square Sum Analysis

Root square sum analysis is a method to obtain the extreme value of circuit performance parameter under a given probability (e.g., $3\sigma = 99.7\%$) by considering the sensitivity of the performance to each parameter provided that all input parameters are independent of each other, the mean and variance of each parameter distribution are known, and the circuit performance obeys normal distribution.

It can be known from the linear relation (Equation 8.11) between the variation Δy of the performance parameter y and the variation Δx_i of the design parameter x_i that, if the standard deviation of Δx_i or x_i is σ_i, then the standard deviation σ_y and mean value μ_y of y can be calculated by the Equations (8.12) and (8.13), respectively, under the condition of mutual independence of x_i.

$$\sigma_y = \left(\sum_{i=1}^{n} \left(\frac{\partial f}{\partial x_i} \sigma_i \right)^2 \right)^{\frac{1}{2}} = \left(\sum_{i=1}^{n} (s_i \sigma_i)^2 \right)^{\frac{1}{2}} \tag{8.12}$$

$$\mu_y = f(x_1, x_2, \ldots x_n) \tag{8.13}$$

where the values of the parameters $x_i\,(i = 1, \ldots, n)$ are nominal values.

Under the given probability γ, the maximum value of the worst-case circuit performance will be $y = \mu_0 + \mu_{(1+\gamma)/2}\sigma_y$, and the minimum value will be $y = \mu_0 - \mu_{(1+\gamma)/2}\sigma_y$, where $(1+\gamma)/2$ is the upper quantile of the normal distribution.

8.1.12.3 Monte Carlo Analysis

Monte Carlo analysis is a statistical analysis method to analyze the circuit performance parameter deviation by sampling the values of circuit component parameters which obey a random distribution. The method assumes that all the input parameters are independent of each other and obey a certain probability distribution, and that the circuit performance parameters obey a normal distribution. The parameter values of each component are generated through random sampling and then are substituted into the circuit equation to calculate the circuit performance parameter values. After several repetitions, the distribution of the circuit performance parameter values is obtained, so the extreme value of the performance parameter under the given probability is determined.

The specific Monte Carlo analysis is carried out in the following way. According to the distribution of the actual parameter X of circuit components and other related quantities, the first random sampling of X is carried out to obtain the sampled values $(x_{11}, x_{12}, \ldots x_{1m})$, where m is the number of input parameters. The first random value $y_1 = f(x_{11}, x_{12}, \ldots x_{1m})$ is calculated by substituting the sampled values into the circuit performance parameter equation. After n repetitions, n random values of the performance parameter can be obtained. Thus, the probability that the circuit performance parameter y appears in different deviation ranges can be calculated through the statistical analysis of y.

8.1.13 Outage Analysis

Outage is the state in which a product cannot perform its required functions. Unplanned spacecraft outages are usually caused by soft faults such as single-particle soft errors and software anomalies.

Outage analysis is the process of identifying the cause for an outage, analyzing its impact on the task or function of a product (system or subsystem) and determining its recovery measures. Outage analysis is a kind of fault impact analysis technique, and is the application of FMEA technology in a specific outage fault.

8.1.13.1 General Requirements for Outage Analysis

The outage analysis starts from the lowest product level to the highest level. The system outage analysis should be supported by the outage data from subsystems/equipment. The typical inputs for outage analysis include:

1. Mission profile
2. Reliability block diagram
3. FMEA worksheet
4. Part failure rate and equipment/component failure rate (reliability prediction report)
5. Product functions, performance and features during startup, running and operation

With the deepening of product development, the above information has been gradually refined. Therefore, the outage analysis should also be updated as the design changes.

The main outputs of outage analysis include:

1. The cause for outage
2. The influence of outage
3. The failure rate of the item that causes the outage
4. The duration of outage
5. The method of outage detection
6. The method of outage recovery
7. The suggestions on design improvement
8. List of all sneak outages

The lowest product level of an outage analysis is determined based on the data information required by the outage analysis conducted by the spacecraft developer. In the outage analysis of a spacecraft system or subsystem, at least the equipment and component-level products should be determined as the lowest product level, and the system or subsystems as the highest product level. This allows for a clear and complete analysis of all the equipment, components and interfaces that lead to the outage of a spacecraft system or subsystem.

8.1.13.2 Implementation Procedure of Outage Analysis

The outage analysis proceeds from the basic event that causes the interrupt of main functions of a task or system or subsystem, and then gradually unfolds the analysis work. The analysis process generally includes the following steps:

1. Find out the equipment/component failure mode that leads to the interruption of product (system or subsystem) function or task, namely, the cause for outage, according to FMEA results and engineering experience.
2. Analyze the impact on product (system or subsystem) function or task.
3. Determine the fault detection method.
4. Estimate the failure rate of the item that causes the outage of a product (system or subsystem).
5. Determine the recovery strategy of a product (system or subsystem) under the function/task interruption.
6. Estimate the duration of a product (system or subsystem) outage.

TABLE 8.5 Typical Outage Analysis Table

No.	Item Name and Function	A Failure Mode That May Cause an Outage	Impact on Task/Function		Failure Detection Measures	Recovery Strategy	Outage Frequency	Outage Duration	Remarks
			Subsystem	System					

7. Put forward the improvement measures and suggestions. Feed the analysis results back timely, and determine, according to the analysis results, the necessity of design changes as the decision basis of design result confirmation. For design changes, the above analysis steps should be repeated to ensure the correctness of the design changes.
8. Fill in the outage analysis form to form an analysis report.

A typical outage analysis table is shown in Table 8.5. According to different analysis purposes, the relevant items in the table can be removed in the implementation process.

8.1.14 Reliability Mathematical Simulation Method

Mathematical simulation is to use a mathematical model, instead of a physical system, for experiments and research. Reliability mathematical simulation is an effective method to analyze the reliability of a complex system by using the reliability simulation model of the system for simulation experiment in order to solve the reliability characteristic quantities or analyze the system reliability problem.

Monte Carlo method is the basis of reliability mathematical simulation. Its basic idea is that when the problem to be solved is of random nature (such as the occurrence probability of an event and the expected value of a random variable), the solution of the problem can be obtained by sampling test.[5]

The typical process of reliability mathematical simulation is as follows:

1. Make clear the problems, simulation goals and plans. Clarify the connotation and constraints of the system, make clear the simulation indicators and the problems to be solved and draw up a simulation plan.
2. Establish the reliability simulation model. If the system S is composed of n basic components among which the i-th component is represented by $Z_i(i=1,2,\ldots n)$, the failure function of each basic component will be $F_i(t)(i=1,2,\ldots n)$. When the failure of the system S is top event, the system has n bottom events. The structure function of the system is represented by $\varphi\left[\vec{X}(t)\right]$, where $\vec{X}(t)=\left[x_1(t),\, x_2(t), \ldots,\, x_i(t), \ldots,\, x_n(t)\right]$:

$$x_i(t)=\begin{cases} 1\text{: the } i\text{-th bottom event occurs at } t; \\ 0\text{: the } i\text{-th bottom event does not occur at } t. \end{cases}$$

The state variable of the system failure at t is represented by $\varphi(t)$:

$$\varphi(t)=\begin{cases} 1: \text{the system fails at } t; \\ 0: \text{the system does not fail at } t. \end{cases}, \text{ and } \varphi(t)=\varphi\left[\vec{X}(t)\right].$$

3. Design the reliability simulation program.

   1. Randomly sample the failure time of n basic components by using the Monte Carlo method.

      Suppose that the sampled value of the failure time of the i-th basic component Z_i in the j-th simulation run is t_{ij}. Then $t_{ij}=F_i^{-1}\left(\eta_{ij}\right)$. The failure time of n basic components can be determined as $t_{1j},t_{2j},\ldots,t_{ij},\ldots,t_{nj}$.

   2. Find out the failure time of the system according to the reliability logic relationship of the system.

      Rank the failure times of n basic components as $t_{t_1},t_{t_2},\ldots,t_{t_i},\ldots,t_{t_j},\ldots,t_{t_n}$ according to their values. The corresponding sequence of basic components is $Z_1',Z_2',\ldots,Z_i',\ldots,Z_j',\ldots,Z_n'$. According to the above time sequence, the corresponding basic components Z_i' are set as the failure state successively from the time $t=t_{t_1}$. Then the system failure is judged by the structural function $\varphi\left[\vec{X}(t)\right]$. If the system does not fail, the simulation will continue until the failure of Z_j' causes the failure of the system. At this time, the failure time of the system is $t_{Kj}=t_{t_k}$, marking the end of the simulation.

   3. Use the interval statistics method for the distribution statistics of the number of system failures.

      After each simulation, the drop point of the system failure time should be determined so as to analyze the distribution of the system failure time. Suppose that the length of the time interval is ΔT. Then the number of system failures within the interval $(t_{r-1},t_r)(r=1,2,\ldots N)$ in N simulations will be $\Delta m_r=\sum_{j=1}^{N}\varphi_j(t_K)$ ($t_{r-1}<t_K\le t_r$), and the number of system failures at $t\le t_r$ will be $m_r=\sum_{j=1}^{N}\varphi_j(t_K)$ ($t_K\le t_r$).

4. For the reliability model of a large complex system, its correctness and accuracy need to be verified and confirmed through debugging and testing.

5. Determine the simulation conditions. Establish the conditions of reliability simulation (including the relationships between simulation output result and control variables), determine different combinations of control variables and the number of simulations and set the initial conditions of the system. For example, set the maximum operating time of the system as T_{max}, and carry out N simulations.

6. Operation and analysis. Carry out simulation operation, system reliability index calculation, simulation error analysis, component importance calculation and other work. For example, the system reliability $R_S(t_r) \approx 1 - \frac{m_r}{N}$ and the system failure density function $f_S(t_r) \approx \frac{\Delta m(t_r)}{\Delta t_r \cdot N}$ can be obtained. If $m_i(T_{max}$ represents the number of system failures caused by the failures of the basic component Z_i within $[0, T_{max}[]]$ in N simulations and M_0 represents the total number of failures of the basic component Z_i, then the importance of Z_i will be $W(Z_i) = \frac{m_i(T_{max}}{M_0}$ and its mode importance can be expressed as $W_N(Z_i) = \frac{m_i(T_{max}}{N}$ (the number of system failures is required to be equal to the number of simulations).
7. Add more operations and analyses. Determine whether to add more operations and analyses according to the operations and analyses that have been completed.
8. Document the simulation scheme, program as well as input and output results.

8.1.15 Reliability Assessment

The reliability assessment of a spacecraft product is the quantitative assessment of reliability index by using the product reliability data.

8.1.15.1 Reliability Assessment Method Based on Life Data

When the sample size of product data on site is large, the assessment method based on life data should be adopted:

$$Z = (z_1, \delta_1, z_2, \delta_2, \ldots, z_n, \delta_n) \tag{8.14}$$

where $z_i = \min(x_i, t_i)$, $\delta_i = I(x_i \le t_i) = \begin{cases} 1, x_i \le t_i \\ 0, x_i > t_i \end{cases}$ $i = 1, 2, \ldots, n$, x_i, $i = 1, \ldots, n$ represents the service life of the equipment and t_i, $i = 1, \ldots, n$ represents the observation time.

8.1.15.2 Reliability Verification Test and Reliability Assessment Method of Success-or-Failure Products

If the number of the launched success-or-failure products is $N \ge 10n$ (n is the number of test pieces), n can be deduced from the reliability calculation formula related to binomial distribution given by the national military standard GJB376 when the number of failures is $F = 0$:

$$n = \frac{\ln(1-\gamma)}{\ln R} \tag{8.15}$$

If $N < 10n$, n can be obtained according to the reliability index R, the number of launched products N and the confidence coefficient γ by looking up the hypergeometric distribution table in GJB376.

If the number of failures is $F=0$ in the success-or-failure firing test of n products under the specified environmental conditions (covering the most severe conditions within the mission profile), the product reliability can meet the reliability index requirements. n should be determined before the success-or-failure reliability verification test (firing test).

8.1.15.3 Reliability Verification Test and Reliability Assessment Method of Life-Type Mechanism

For a mechanism requiring long-time continuous or intermittent motion in orbit, its reliability characteristic quantity is generally its life. If the time (or number) of tasks is X_0, the reliability index of the life-type mechanism is R, the number of test pieces is n (generally $n \geq 2$) and the shape parameter of Weibull distribution is m ($1 < m \leq 3$), then the time (or number) of life tests should be X_R. When the time (or number) of tests reaches X_R and the product does not fail, the reliability of the product can be verified.

$$X_R = X_0 \left[\frac{\ln(1-\gamma)}{n \ln R(X_0)} \right]^{1/m} \tag{8.16}$$

where γ is confidence coefficient.

8.1.15.4 Reliability Verification Test and Reliability Assessment Method of Performance Parameter Measuring Mechanism

The reliability characteristics of some spacecraft products are performance parameters (such as velocity, acceleration, displacement and leakage rate). When these reliability characteristics are limited by one-sided parameters, the product reliability can be evaluated by using the performance parameter test data of n products obtained under the specified environmental conditions (covering the most severe conditions within the mission profile).

Suppose that the valid sample observations obtained from n performance parameter tests are x_i ($i=1, 2\ldots n$). According to the test under GB/T4882, x_i do not reject the null hypothesis of normality, and the sample mean $\overline{x}$ and sample standard deviation s are respectively:

$$\overline{x} = \frac{1}{n}\sum_{i=1}^{n} x_i \tag{8.17}$$

$$s = \sqrt{\frac{\sum_{i=1}^{n}(x_i - \overline{x})^2}{n-1}} \tag{8.18}$$

If $x \leq U$, the margin will be $K=(U-\overline{x})/s$; if $x \geq L$, the margin will be $K=(\overline{x}-L)/s$. Then the lower limit of reliability confidence, namely, R_L, can be obtained by referring to the national standard GB/T4885 according to the values of K, n and γ (γ is confidence level).

8.1.15.5 Comprehensive Reliability Assessment Method

The satellite platform is a complex system that can be represented as a pyramid structure. Based on this pyramid structure, the Bayes evaluation of the platform reliability is carried out from bottom to top. The basic steps of this comprehensive assessment method are as follows:

1. Platform reliability modeling

   The platform reliability modeling mainly includes two aspects: the establishment of platform reliability structure diagram and the determination of life distribution of the system, subsystems and equipment.

2. Determining the posterior distribution and posterior moment of equipment reliability

   Based on multi-source information fusion, the posterior distribution of equipment reliability can be obtained.

   After the posterior distribution of equipment reliability is obtained, the first m-order posterior moments of equipment reliability can be determined to provide input for calculating the prior moment of the subsequent platform.

3. Determining the prior moment of platform reliability

   From the steps (1) and (2), the structure function of the system and the first m-order moments of equipment reliability are obtained, respectively. Then, the first m-order prior moments of the system can be calculated from the system's structure function according to the reliability posterior moment of each equipment in the system.

4. Platform reliability fusion

   The system's first m-order prior moments are obtained at the step (3). If the prior distribution of system reliability is known, the moment equivalent method or the relative least square method can be used to calculate the prior distribution parameter according to the system's prior moment.

   If other prior information of the system is available, it should be combined with the integrated equivalent prior information of equipment to obtain a fused prior distribution.

5. Bayes evaluation of platform reliability[6]

   According to step (4), the prior distribution of satellite system reliability is obtained. If the test data on satellite system is available, the posterior distribution of system reliability can be obtained by Bayes theorem. Based on the posteriori distribution of reliability parameters, the system reliability can be evaluated, usually including the point estimation, confidence interval estimation and hypothesis testing of reliability parameters.

8.2 METHODS OF SAFETY DESIGN AND ANALYSIS

Safety is the state of exemption from death and injury, occupational disease, spacecraft damage, major property damage or environmental damage.

1. Safety is the opposite of danger. The existence of a danger may cause an accident, so danger is the source of potential threats to safety. Therefore, danger is defined as the state that may lead to an accident, also known as dangerous state. A factor that causes a danger is the source of danger. Accident is an event or a series of events resulting in death or injury, occupational disease, damage or loss of equipment (or property) or environmental damage.

2. Safety and reliability are often mistaken for each other. In fact, they are not only associated to each other, but also different from each other. The reliability requires no failure in an engineering system, while the safety requires no accidents in the system. The reliability design considers all possible functional failures, while the safety design considers those hazards that threaten the safety of astronauts, including the failures that can cause accidents (not all the failures).

However, reliability may not be equal to safety. For example, the reentry module is provided with a variety of initiating explosive devices (initiator and detonating fuse). These devices are reliable enough to ensure the reliable initiation of the through-bulkhead initiator (pyrotechnic executor), but the toxic gas produced after their operation may penetrate or leak into the cabin to harm the astronauts.

In some cases, reliability and safety are at odds. For example, redundancy is designed for the spacecraft propulsion system to improve reliability. However, the redundancy will increase the sealing joints in the system and the possibility of propellant leakage and gas leakage, thus causing a higher possibility of fire accident and lower safety performance.

8.2.1 General Safety Design Method for Spacecraft Products

8.2.1.1 Safety Objective

The implementation of safety assurance work can effectively identify the safety related risks, evaluate, control and minimize the risks, and reduce the risks to an acceptable level in the design, development, production and use of spacecraft products.

The implementation of safety assurance work should ensure that the following dangerous events will not happen on a spacecraft product:

1. Casualties

2. Environmental harm

3. Damage to public and private properties (including launching equipment)

4. Damage to the spacecraft and launcher

5. Damage to the ground equipment and facilities

8.2.1.2 Safety Design Criteria

Safety design criteria are the design methods or principles that should be considered before the product safety design, mainly including:

1. Safety-first principle: put safety design in the first place during the screening of various schemes and the weighing of various factors.
2. Risk minimization principle: ensure no risk in design, which, if inevitable, should be minimized.
3. Fail-safe design: set the safe mode for the spacecraft system. In case of an in-orbit failure, the spacecraft should enter the safe mode to ensure safety—energy safety first and information channel safety second.
4. Margin design: comprehensively consider the influence of various adverse factors (such as environmental factors and service factors), and leave a certain design margin.
5. Fault isolation design: prevent the fault of a product from putting the products interfacing with it into the Class-I or Class-II hazard.
6. Fault-tolerant design: a single failure in hardware or software or a single misoperation should not lead to serious or catastrophic consequences for the spacecraft.

8.2.1.3 Safety Design Priority

The safety design and the priority of safety design measures are:

1. Eliminate the dangers. The dangers should be eliminated from the design and operation schemes through the selection of design techniques and operating characteristics and the consideration of engineering constraints and mission objectives.
2. Minimize the dangers. For the dangers that cannot be eliminated by design means, their risks should be reduced to an acceptable level by using the minimum-hazard design principle, selecting appropriate design techniques and operating characteristics and considering the engineering constraints and mission objectives.
3. Use safety devices to control the hazards. If the identified hazards cannot be eliminated or reduced to an acceptable level through the selection of design techniques and operating characteristics, automatic safety devices should be used as a part of the system (subsystem or equipment) to reduce and control the hazards.
4. Use alarm devices to control the hazards. If the design measures and safety devices can not achieve or meet the requirements, the alarm devices that can detect the dangers in time and give alarm signals should be used, so that the user can take timely safety measures or shut off the dangerous unit.
5. Use special procedures to control the hazards. If neither a hazard or its risk can be eliminated or reduced to an acceptable level by design means, nor the safety requirements of the product can be met by using the safety devices or alarm devices, then special procedures shall be formulated to control the hazard.

8.2.1.4 Safety Requirements for Propulsion System

1. The leakage rates (external leakage rate and internal leakage rate) of fluid control devices and the related components shall meet the design indexes under the worst combination of environmental conditions, and shall be verified by tests.
2. In general, at least two mechanical isolation measures should be taken between oxidant and combustion agent/catalyst/propellant.
3. In order to prevent the thruster and engine from misfiring, at least two mechanical and electrical isolators (such as electric explosion valve, self-locking valve and solenoid valve) should be set between each propellant tank and the thruster/engine. They should be mechanically independent and connected in series, and be electrically controlled by at least one independent prohibition circuit.
4. To prevent the false initiation of electric detonator, at least three independent safety switches in series should be used to control the ignition circuit of initiating explosive devices.
5. Electric inhibit control device is a fail-safe device. It is in the OFF state before receiving an "ON" signal.
6. The activation of any fluid control device shall not trigger an explosion.

8.2.1.5 Pressure Vessel Safety Requirements

1. The design safety factors of all pressure vessels and tanks are generally not less than 2.
2. A mechanical isolation device shall be set between the pressure vessel and the storage tank as the first downstream component of the vessel, and shall be installed as close to the pressure vessel as possible.
3. All pressure vessels shall be equipped with the devices for monitoring the vessel pressure.
4. All pressure vessels and tanks must undergo ultrasonic flaw detection, X-ray flaw detection, pressure test, acoustic emission inspection and leakage rate detection.

8.2.1.6 Piping Safety Requirements

1. The minimum design burst pressure of all pipelines shall be four times the maximum pressure required for their operation in the system.
2. All pipelines should be separately tightened and firmly supported to prevent mechanical stress and vibration damage.
3. The inside of all pipelines must be strictly cleaned in accordance with the requirements of pipeline cleaning process.
4. The pipelines must be inspected for defects and leaks.

8.2.1.7 Safety Requirements for Valves, Pressure Regulators and Control Devices

1. The installation positions of the inlet/outlet (or discharge) valves should be easy to operate to ensure that the propellant and gas between any components in the system can be discharged or released.
2. The shut-off valve shall not be installed in series with a safety valve unless another reliable safety device operating independently is installed in parallel with the valve.
3. A check valve should be installed in the two-component pressure system to avoid the mixing of fuel and oxidant steam, which will cause a danger.
4. When the parallel pressure supply system and the pressure vessel serve a common downstream system, they shall be separated by valves.

8.2.1.8 Safety Requirements for Initiating Explosive Devices

1. The initiating explosive devices should be able to achieve their final function at 120% or more of the maximum input energy (or charge) or at 80% or less of the minimum input energy (or charge) and maintain the structural integrity, without increasing the initial free volume. The devices with output performance parameters (such as thrust, velocity and synchronicity) should meet the requirements of output performance indexes.
2. The protective measures such as shielding should be taken for the initiating explosive devices sensitive to electrostatic and electromagnetic environments. Electric detonator should be protected against short circuit before connecting with the ignition power system.
3. The initiating explosive devices must be protected against the misfiring caused by static electricity accumulation. In the required electromagnetic interference environment, they shall not fire and their performance shall not decline. The electric detonator shall be a 1A/1W anti-RF anti-static insensitive electric detonator that will not fire after being powered for 5 minutes.
4. The following aspects should be considered in the design after the initiation of an initiating explosive device:
    1. No fragment is generated, and no pollution caused by solid particles and gas is allowed to harm personnel and other equipment.
    2. No short circuit to the ground, such as the short circuit caused by the contact between igniter wire and ground, is allowed under any working conditions.
5. The internal quality of an initiating explosive device shall be examined by 100% X-ray (industrial CT) or γ ray.

6. According to GJB 1307A-2004 General Specification for Pyrotechnic Devices Used on Space Vehicles, the storage, transportation and operation environment of pyrotechnic devices shall meet the following requirements:
    1. Temperature: 0°C~40°C.
    2. Relative humidity: 20%–80%.
    3. It is prohibited to load, unload and transport the pyrotechnic devices in thunderstorms.
7. The power supply control for dangerous components such as pyrotechnic devices must follow the following principles:
    1. In the ground test, experiment and transportation of a spacecraft, the ignition safety circuit of pyrotechnic devices should be designed with at least four-stage safety.
    2. Before the launch, the safety switch of safety ignition circuit should have at least four-stage safety.
    3. During launch and operation, the safety switch of safety ignition circuit of a pyrotechnic device should have at least three-stage safety.
    4. The ignition state of all pyrotechnic devices can be detected by telemetry means.

8.2.1.9 Battery Safety Requirements

1. The battery cells must be sealed to avoid electrolyte leakage.
2. Each battery pack should have a good insulation design, so that the output of each cell and battery pack can be electrically insulated from the spacecraft structure before the busbar is grounded. When being installing on the spacecraft, the batteries should be uncharged or weakly charged.
3. The nickel-hydrogen batteries with high-pressure vessel should have good safety design. The safety factor of pressure vessel should not be less than 2.5. The pressure vessel should meet the requirements of fatigue life test and pressure test.
4. The charge control circuit of a cadmium nickel battery pack should be designed to prevent overcharge and overdischarge and to ensure insulation.
5. The zinc-silver battery cells should be sealed to avoid deflation, pressure leakage and alkali adhesion.
6. For lithium battery or lithium battery pack, measures should be taken to prevent explosion, photoelectricity and short circuit, and the norms on safe use should be formulated. The short circuit, overcharge, overdischarge and overheating of a battery pack are strictly prohibited.

8.2.2 Methods of Hazard Source Identification and Hazard Analysis

The basic work of safety analysis is the identification of hazard sources. The hazard sources of a spacecraft system and its subsystems and equipment should be identified, so as to control the risk of hazard occurrence in the whole life cycle of all levels of the products.

Possible hazard sources, including the general hazard sources that are directly exposed and the fault hazard sources that are not easily identified, should be systematically identified. The identification of hazard sources should consider at least the following aspects:

1. Hazardous goods (such as propellant, gunpowder, initiating explosive system, toxic substances, power supply, high-pressure gas source and nuclear source)
2. The working environments of the products, including natural environment and induced environment (such as vibration, impact, ultimate temperature, vacuum, lightning, electromagnetism, ionic radiation and plume)
3. Product function failure or abnormal working conditions
4. Product design defects, including the insufficient safety margin of mechanical structural parts, the incompatibility items (such as material incompatibility and electromagnetic interference), the undesirable working condition caused by a sneak circuit and the interface disharmony (for the interfaces between hardware, between software, between hardware and software and between information transmitter and receiver)
5. Defects in key instructions and control software
6. The dangers that may be caused or introduced in the process of use, test, experiment, maintenance and support
7. Misoperation or illegal operation
8. The storage, handling and transportation of dangerous goods, and the discharge of propellant and other flammable and explosive gas or liquid
9. Safety-related equipment, safety protection devices and other safety measures

The list of hazard sources can be drawn up by reviewing the lessons learned from similar products and the safety data, and can also be identified with the help of FMEA tools. For the list of general hazard sources (example), see Table 8.6; for the list of fault hazard sources, see Table 8.7.

The risks of the identified general hazard sources and fault hazard sources are analyzed in Table 8.8.

8.2.3 Safety Verification and Evaluation

The safety design and analysis of a spacecraft product is generally followed by safety verification and safety evaluation. Safety verification is to verify whether the safety design of a

TABLE 8.6 List of General Hazard Sources of a Spacecraft (Example)

Number of Hazard Source	Hazard Source	Position	Hazard Type: Fire	Explosion	Vibration Impact Acoustic Vibration	Explosive Impact	Heat	Pollution	Radiation	Electric Discharge	Pathology Physiology Psychology	Toxicity	Others
1	Bipropellant	Oxidant tank Combustion agent tank	√	√				√				√	
2	Tank	On middle panel		√									
3	Helium cylinder	On middle panel		√									
4	Nickel-hydrogen battery	Battery panel in service module		√				√					
5	Heat pipe	Outside the spacecraft				√	√	√				√	
6	Antenna	Outside the spacecraft							√				
……	……	……											

TABLE 8.7 List of Fault Hazard Sources of a Spacecraft

No.	Fault Hazard Source	Function/Mission Description	Failure Mode/ Human Error	Mission Phase	Severity	Possibility	Risk Index	Fault Detection Method	Failure Cause	Safety Improvement Measures	Remarks
									……		
			……								

TABLE 8.8 Spacecraft Hazard Analysis Table

System (subsystem)
Development stage **No.:**

											Safety Improvement Measures		After Improvement			
No.	Code	Product Name or Function	Hazard Source	Mission Phase	Hazardous Event	Reason	Consequence	Severity	Possibility	Risk Index	Elimination	Control	Verification Method and Result	Possibility	Risk Index	Remarks

spacecraft product meets the requirements and whether the safety-related control measures are effective. The safety verification includes qualitative safety verification and quantitative safety verification. Safety evaluation is generally to make a comprehensive evaluation of the system safety before the transition stage of spacecraft product development or before launch, so as to provide a basis for management decision.

8.2.3.1 Safety Verification

The purpose of safety verification is to ensure that safety has been designed and manufactured into the product, and to prove that:

1. Dangerous events are unlikely to occur.
2. Even if a dangerous event occurs, its risk can be accepted.

One or several of the methods, including test, demonstration, analysis, design review and inspection, may be used for safety verification.

8.2.3.2 Qualitative Safety Verification

The qualitative safety verification includes two aspects: the verification of safety measures effectiveness and the verification of safety-critical functions:

1. Verification of safety measures effectiveness
    1. Verify and track the safety improvement measures taken to eliminate or control the hazards, and ensure the effectiveness of each measure.
    2. The hazard that can not be eliminated by design and may cause catastrophic consequences should be controlled through safety device, alarm device and special procedure. The effectiveness of these safety measures can be verified by test (actual practice).
    3. For the safety items that need to be verified by mandatory inspection, a safety inspection shall be carried out before flight according to the compulsory inspection points, and a mandatory inspection or test shall be carried out for the deviation and out-of-tolerance degree of safety-critical items.
2. Verification of safety-critical functions

    The hardware, software, firmware and operation procedures that perform the safety-critical functions on a spacecraft are tested in three working modes, namely, normal mode, emergency mode and first-aid mode, to verify their safety-critical characteristics, performance, safety margin and fault tolerance, and to check whether they meet the safety requirements during the human participation in the operation procedures. The fault impact and fault tolerance are verified through the fault "injection" test.

8.2.3.3 Quantitative Safety Verification

For a spacecraft product with quantitative safety requirement, its safety should be quantitatively evaluated based on the quantitative reliability evaluation by making full use of in-orbit safety data. When an unsafe incident happens on an onboard device, remedial measures should be taken to reduce the risk to an acceptable level. The probabilistic risk evaluation method can also be used to evaluate the safety quantitatively.

8.2.3.4 Safety Evaluation

The flight safety evaluation before the spacecraft launch is to make a comprehensive evaluation of the previous safety work, evaluate the safety in the flight process and list the hazards and risks in each flight phase and flight event of the flight mission profile.

The safety evaluation is to evaluate the safety level of spacecraft system design as well as the possible hazards in the adopted hardware, software and operation procedures and the corresponding measures, and to check whether the risks are controlled to the prescribed acceptable level. It is mainly finished by risk evaluation to check whether the identified hazard has been eliminated or controlled to the specified acceptable level and whether the system can be transferred to the next development stage or flight test. The safety evaluation should also give suggestions on the safety of the relevant interfaces in the system.

8.3 MAINTAINABILITY DESIGN AND ANALYSIS

Maintainability is an inherent property of the spacecraft product. Good maintainability cannot be designed by calculation and analysis alone. Guidelines need to be drawn up based on the experience in design and use, in order to guide the design.

8.3.1 General

The spacecraft maintainability design is to analyze and identify the maintainability design requirements, and carry out the maintainability design to ensure that the spacecraft product meets the maintainability requirements. The maintainability analysis is to provide input for maintainability solution selection, fault contingency planning and support planning. The maintainability verification is to check whether the product meets the specified maintainability requirements.[7]

According to the mission and product characteristics, professional requirements, phase, complexity and criticality, operational (storage) environment, new technology content, cost, schedule, product maturity and other factors, the comprehensiveness and implementation time of spacecraft maintainability program can be analyzed to identify the maintainability program and its development stage.

The inputs to the maintainability design are:

1. GJB 1909A Demonstration of Reliability, Maintainability and Supportability Requirements for Materiel.
2. Other reference materials, including the maintainability information of similar products obtained during in-orbit operation, and the normative documents and materials including maintainability design criteria.

3. The inputs to system-level maintainability design also include: the general requirements for model development.
4. The inputs to subsystem maintainability design also include: the product assurance requirements for spacecraft system, the design and construction specifications of spacecraft products and the system engineering's requirements for subsystem technology.
5. The inputs to equipment maintainability design also include: the product assurance requirements for spacecraft subsystems, the design and construction specifications of spacecraft products and the equipment development plan.

The general principles of maintainability design at each stage of spacecraft development are as follows:

1. At the feasibility demonstration stage, the background model demonstration group shall fully communicate with the user, analyze the spacecraft operation requirements and tactical index requirements proposed by the user, identify the user's maintainability requirements and demonstrate their feasibility and determine the initial requirements for spacecraft maintainability. If the user does not make clear the maintainability requirements, the maintainability requirements should be determined by considering both the characteristics of model mission and the relevant standards and specifications.
2. At the conceptual design stage, the maintainability design criteria applicable to this spacecraft model shall be determined by the system engineering and incorporated into the specifications for spacecraft design and construction. At the prototype development stage, the maintainability design criteria shall be improved to contain the subsystem-level technical requirements. Meanwhile, they shall be supplemented and refined by subsystem/equipment designers according to the subsystem/equipment characteristics.
3. The maintainability design of spacecraft system/subsystems shall be carried out at the conceptual design stage and prototype development stage according to the maintainability design criteria, and the equipment maintainability design shall be carried out at the prototype development stage. If the flight model development stage contains the technical condition changes involving maintainability, the supplementary maintainability design shall be carried out at this stage.
4. The key of spacecraft maintainability design is in-orbit maintainability design.

8.3.2 Design Criteria for Hardware Maintainability

The general contents and examples of hardware maintainability design criteria are given according to the spacecraft development process, mainly including simplified maintenance design, accessibility design, standardization/modularity/ interchangeability design, error prevention design, maintenance safety design and human element engineering design.

8.3.2.1 Simplified Maintenance Design

Through the formulation and implementation of simplified maintenance design criteria, the spacecraft product maintenance can be reduced and facilitated. The simplified maintenance design mainly includes two aspects: functional structure simplification and maintenance procedure simplification. It can be carried out according to the following design criteria:

1. Carry out reasonable assembly design and reduce the connections, so that the maintenance operations (such as testing and replacement) become simple and convenient and the maintenance of one part requires no or less disassembly/ movement of other parts.
2. Simplify the maintenance operation contents, and reduce the requirement for maintenance technology level and the maintenance operation difficulty.
3. Consider the more use of common tools and the less use of special tools in the maintenance design.
4. Reduce the fasteners, or use the push-on fasteners to facilitate the disassembly/reassembly.
5. Enable the quick removal of electrical and hydraulic interfaces.
6. Reduce the varieties and quantity of accessories and spare parts as much as possible.

Example:

Figure 8.10 shows two layouts of an electronic device at the conceptual design stage and prototype development stage. In the equipment layout of prototyping stage, the PCB 1, PCB 2 and mounting plate of conceptual design stage are combined into the PCB 3. By combining the structures with the same or similar functions, the internal layout of the equipment is simplified and the varieties and number of parts are reduced. At the same

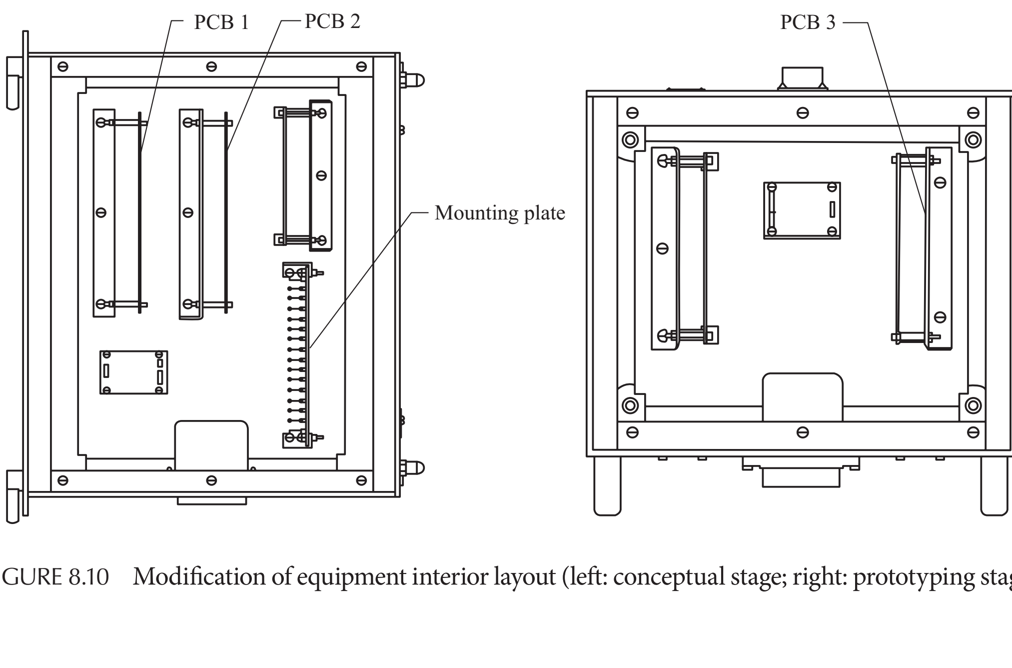

FIGURE 8.10 Modification of equipment interior layout (left: conceptual stage; right: prototyping stage)

time, the tool requirement at the prototyping stage is simple so that only common tools are required to complete the disassembly and assembly of the equipment.

8.3.2.2 Accessibility Design

Through the formulation and implementation of accessibility design criteria, the assembly or disassembly of a spacecraft product can be seen and touched, and can be conveniently implemented in enough working space without affecting the surrounding parts. The accessibility design can be carried out according to the following design criteria:

1. The system/equipment layout shall be arranged according to the failure rate, maintenance difficulty, size, weight and installation characteristics of its components.
2. No inaccessible dead corners can be found in the system area to be repaired.
3. Enough space should be left for maintenance and disassembly/reassembly in the overall assembly design and for operation after the maintenance and disassembly/reassembly of equipment and components. The check points, test points and maintenance points for all kinds of maintenance should be accessible.
4. The products and areas that need inspection, maintenance, mounting/dismounting and replacement should be well accessible, without disassembling other products during maintenance.
5. The laying of cables and waveguides should consider the space required during equipment maintenance.
6. High-pressure and high-temperature pipelines should be arranged in conspicuous positions for maintenance convenience.

Example: rubidium clock is a new key device of a satellite, without flight experience. Considering its relatively high failure rate, the rubidium clock is generally installed near the opening of the back floor, as shown in Figure 8.11, to facilitate the disassembly and reassembly.

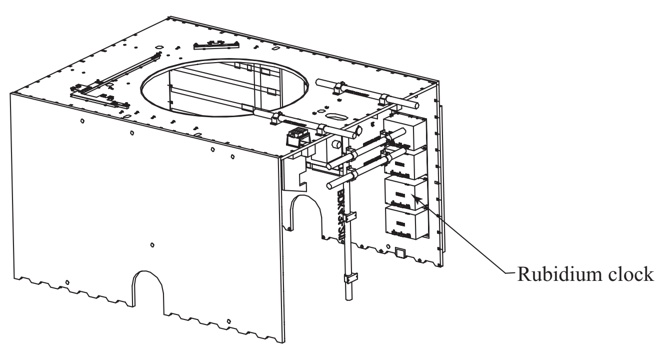

FIGURE 8.11 Installation position of rubidium clock in a satellite.

8.3.2.3 Standardization/Modularity/Interchangeability Design

The modular design concept is adopted. Through the formulation and implementation of standardization, modularity and interchangeability design criteria, the spacecraft products can be generalized, serialized and combined, the product varieties can be reduced, the faulty products can be easily replaced and repaired and similar spacecraft products can be interchanged physically (in terms of shape and size) and functionally. The standardization, modularity and interchangeability design can be carried out according to the following design criteria:

1. The hardware and tools should be generalized, standardized and interchangeable to reduce their varieties and quantity.
2. All documents (design drawings, technical documents etc.) should comply with the standards or specifications.
3. In the design, general parts and standard parts should be given priority to minimize the product varieties.
4. Components and materials should be selected from the relevant recommendation catalog.
5. The unified power supply and standard interface should be used for electronic equipment.
6. The products should be divided into mechanical, electrical, electronic and electromechanical modules as much as possible.
7. The modules should be removed and replaced as quickly as possible.
8. The overall assembly should be relatively concentrated and zoned according to the different functions of the equipment.
9. The components or units that are critical or damageable or have a high failure rate should have good interchangeability and universality.

Example of standardized design: during the design stage of a satellite, the Satellite Construction Code was compiled and the standardization program was formulated. In the design process, the system design was carried out in strict accordance with the satellite design standards and specifications. The satellite structure parts and the parts directly belonging to the satellite were generalized, serialized and combined into as few varieties as possible. The system engineering has developed the Design Atlas of the Generalized Parts Directly Belonging to Satellite, through which the parts directly belonging to multiple satellites have been generalized. In the design, general parts and standard parts were given priority, and the varieties of screws, fasteners and connectors were reduced in order to reduce the cost and facilitate the replacement and maintenance.

Example of interchangeability design: the devices with the same function inside each subsystem of the satellite are interchangeable, such as the reaction wheels A/B/C/D and

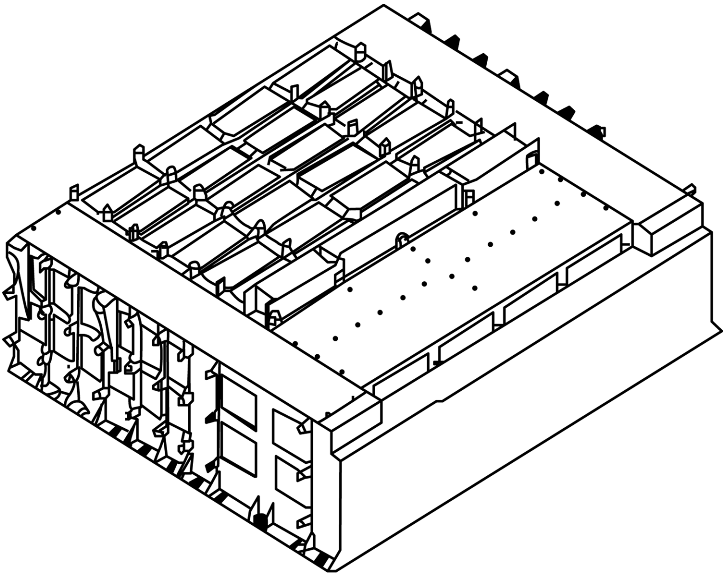

FIGURE 8.12 Modular structure of power controller on a satellite.

the digital sun sensors A/B/C/D. The same equipment is compatible with the interfaces of the products made by different manufacturers, which can be exchanged. For example, the traveling wave tube amplifier is compatible with both domestic and imported interfaces. The same devices on a batch of satellites flying in different orbits have the same interface and can be exchanged. The parts directly belonging to satellites and the standard parts can be exchanged as long as their specifications are the same. The standard parts comply with aerospace standards.

Example of modularity design: the power controller on a satellite, as shown in Figure 8.12, is composed of multiple modules in the same shape. Its internal space is reasonably allocated according to the needs of each functional module. According to the control characteristics of primary power system, the power controller is divided into six modules, including connection module, shunt module, telecontrol and telemetry module, busbar filtration module, charging module and boost discharge module, so as to minimize the maintenance impact in a failure.

8.3.2.4 Error-Proofing Design

Through the formulation and implementation of error-proofing design criteria and the adoption of design measures and identification marks, the mistake such as wrong, reversed or missed assembly can be prevented in the assembling of the products at all levels and of the whole spacecraft. The error-proofing design can be carried out according to the following design criteria:

1. In the design process, error-proofing design should be carried out and fault tolerant technology should be adopted, so that some errors will be prompted or alarmed and will not cause major accidents.

2. The parts, components and assemblies that may be misoperated, such as the adjacent parts with similar appearance but different functions, the important connectors and the parts that are prone to errors during installation, should be confined structurally (for example, adding a locating device) or carry obvious error-proofing identifiers.

3. The identifiers shall be kept clear and durable for a long time in the use, storage and transportation of the products.

4. The connection points between the test points and other relevant equipment should be marked with the product name and necessary data.

5. The valve bodies should be marked with direction, and the pipelines should carry identifiers. Different pipes or the pipes with flow direction requirement should be connected by the fittings with different diameters or threads to prevent wrong connection.

6. Terminals, cables and core wires should carry identifiers, and the identification information should be as complete as possible.

7. The operation manuals of the system and equipment should give anti-misoperation instructions.

System-level example: pipeline error-prevention design. The satellites generally use monopropellant propulsion system or bipropellant propulsion system. The bipropellant propulsion system has more complex design and more valve bodies and pipeline connections. In particular, the pressure reducer, high-pressure self-locking valve, low pressure self-locking valve, large liquid filter and small liquid filter have the direction requirements. In order to ensure the correctness of the welding direction, the following error-proofing design is adopted. All the valve bodies are engraved with arrows to show the directions so as to avoid installation status ambiguity, as shown in Figure 8.13.

Equipment-level example: various connectors are installed on the onboard instruments and equipment, and some equipment has the installation direction requirements. Therefore, in the satellite layout design, the equipment installation must be unique, while the equipment installation error must be prevented. For example, the installation reference point (commonly known as “Point R”) should be clearly marked on the equipment, while its position should be clearly marked on the equipment installation drawing to correspond to the mark on the equipment, as shown in Figure 8.14.

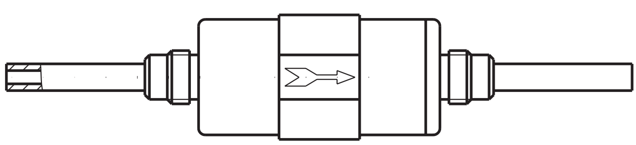

FIGURE 8.13 Outside view of the gas filter on a satellite.

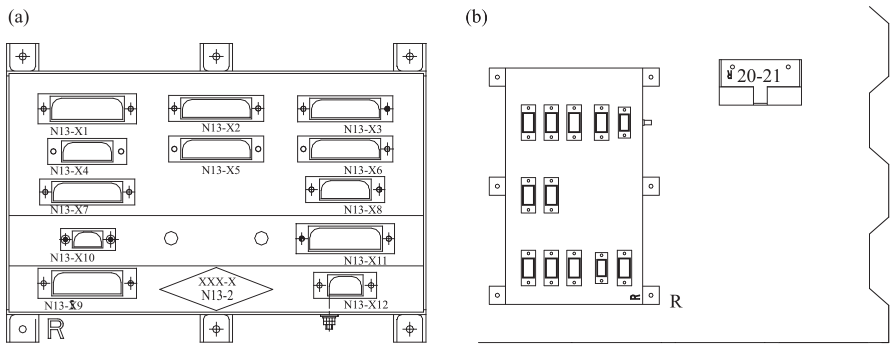

FIGURE 8.14 Layout of a satellite with electric equipment (a) and part of equipment installation drawing (b).

8.3.2.5 Maintenance Safety Design

The maintenance safety design criteria are formulated and implemented to ensure no casualties, equipment damage and other accidents in the spacecraft maintenance operations.

The maintenance safety design can be carried out according to the following design criteria:

1. The hardware and software with the function of autonomous fault diagnosis and recovery should be designed with the corresponding control switches, and should be enabled or disabled by telecommand to facilitate the safety control in orbit.
2. The operation and maintenance personnel should avoid high voltage, high voltage discharge, high temperature, low temperature, sharp edges and points, radiation, chemical pollution and other hazards.
3. Each lifting point and hoisting point of the spacecraft should be clearly marked.
4. The removal, replacement, testing or inspection of a product should not cause a danger.

System-level example: the openings of a propelling module structure, as shown in Figure 8.15, are circular, and the channels on the structure have no sharp edges and corners.

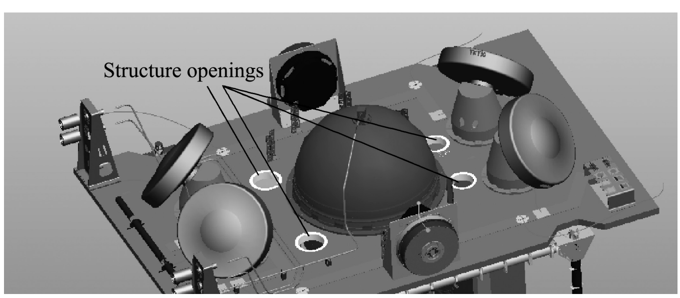

FIGURE 8.15 Openings of a propelling module structure

Equipment-level example: the disconnected electrical connectors are protected with plastic caps or manufacturer-supplied end covers. The disconnection of an electrical connector without proper protection can result in an electric shock to the operator connecting this socket/plug, an increased likelihood of surface corrosion due to environmental factors or an electric shock when the connector is reused.

8.3.2.6 Human-Element Engineering Design

The human-element engineering design criteria are formulated and implemented to ensure that the spacecraft product design can meet the needs of human element engineering, and to improve the quality and efficiency of maintenance work.

The human element engineering design is carried out according to the following design criteria:

1. Provide appropriate operation space in accordance with the operation/maintenance worker's position and posture and the tool condition, and ensure that the worker can operate the equipment in an appropriate posture (rather than kneeling, lying, squatting or other postures prone to fatigue or injury).
2. Consider the worker's physical limit in lifting, pushing and pulling, lifting and rotating an object.
3. Take measures to control the noise, vibration, temperature, humidity and electromagnetic radiation in the operation environment within the prescribed acceptable level, or to protect the operation/maintenance worker.
4. Reduce heavy and complex maintenance items as much as possible, especially the testing and maintenance items requiring strong skills during the equipment rotation.

Example: the lifting lugs and operating handrails are installed on heavy equipment (e.g., batteries) of a satellite to help the maintenance personnel conveniently lift out and handle the equipment.

8.3.3 In-orbit Maintainability Design

Through the formulation and implementation of in-orbit maintainability design criteria, the in-orbit maintenance requirements of spacecraft products can be met. The in-orbit maintainability design of satellites is mainly realized by software maintainability design.

According to the characteristics of in-orbit spacecraft operation, the in-orbit maintainability design can be carried out from the following aspects:

1. The measures such as system reconfiguration and autonomous resetting should be taken to maintain the continuity of system functions and reduce the impact of function interruption on spacecraft mission.
2. When the spacecraft fails in orbit and cannot recover autonomously, it shall be able to recover through the ground instructions on power on/off and generator tripping.

3. The software shall support in-orbit maintenance and modification to correct the in-orbit anomalies caused by software-related problems, and the ground information uplink channel shall meet the uplink requirements of in-orbit software maintenance and code fixing.
4. The design shall consider and provide emergency measures for the spacecraft to recover all or part of its functions in a short time under special conditions such as invisible segment and shadow period outside the country.

Example 1:

The data processing and routing unit of a satellite implements the management of a variety of onboard data through FPGA and software. Specific measures are designed to reconstruct the entire configuration item or module in orbit through program injection. In addition, a way is provided to repair the software defects after the orbit insertion. When a fault occurs in orbit, it can be repaired by autonomous reconstruction or by using the updated program blocks transmitted from the ground.

Example 2:

When the electronic equipment autonomously enters the resetting state due to the impact of in-orbit environment, the first thing is to check whether the payload function software is running normally. If it is normal, the software will jump to clear watchdog timer so that the state A of "watchdog" equipment will recover to 0 (0 is "normal operation").

8.4 TESTABILITY DESIGN AND ANALYSIS

The testability of a spacecraft product refers to a design characteristic that can timely and accurately determine the condition (working, not working or performance degradation) of the product in orbit and isolate its internal faults. The design of testability is completed in synchronization with the design of the spacecraft product. The purpose of testability design is to improve the spacecraft product capabilities of condition monitoring and fault diagnosis, including performance monitoring, fault detection and fault isolation.

8.4.1 Design of Inherent Testability

Inherent testability is the testability that depends only on the product design and does not depend on test incentives and test responses. The design of inherent testability is an important part of product testability design. It requires that the product design can ensure convenient product testing. Inherent testability is the basis of built-in testing, external automatic testing and manual testing. In order to realize the detection and isolation of faults, the design of inherent testability needs to consider three aspects: the reasonable division of function and structure of the system or equipment, the test observability and the test controllability.

1. Reasonable division of function and structure. A complex system is reasonably divided into different product levels, each of which is further divided by function and structure into simple constitutional units that can be tested separately. One replaceable unit had better implement only one function. If multiple functions are implemented by one replaceable unit, each function shall be able to be tested separately.

2. Test observability is a characteristic that determines or describes the degree to which the signals related to the system and equipment can be observed. During the design of inherent testability, the spacecraft in-orbit test system (autonomous fault detection and diagnosis system, BIT etc.) shall be able to observe the internal fault characteristic data of a product, which will be used for fault detection and isolation.

3. Test controllability is a characteristic that determines or describes the degree to which the signals related to the system and equipment can be controlled. During the design of inherent testability, the spacecraft in-orbit test system (autonomous fault detection and diagnosis system, BIT etc.) shall be able to control the internal operational module of a product to detect and isolate internal faults by means of test data uploading and self-inspection.

8.4.2 Design of Fault Diagnosis Strategy

Spacecraft fault diagnosis is an activity of checking, separating and preventing the faults in order to prevent the faults in advance, find the faults in time and eliminate the faults thoroughly.

Diagnostic scheme is the overall idea on the fault diagnosis of the identified diagnosis object, mainly including diagnosis object, scope, functions, applicable scheme, diagnosis requirements and capabilities. Both diagnosis and testability belong to the category of testing technology, but their concepts and connotations have different focuses. The difference is that the diagnosis emphasizes the process of fault detection and isolation, while the testability emphasizes the design characteristics of the system. The determination of diagnosis scheme is inseparable from the testability design. The system and equipment with good testability design can reduce the occurrence of undetected faults. When various testability design methods are used for fault diagnosis, the best diagnostic scheme that meets the testability requirements should be selected through tradeoff analysis.

Both qualitative tradeoff analysis and diagnostic capability analysis need to be considered in the process of determining a diagnostic scheme. The initial scheme is determined by qualitative tradeoff analysis, mainly according to the requirements of testability and maintainability of the spacecraft system, the requirements for fault diagnosis and recovery time and the spacecraft computing capability used for autonomous fault diagnosis. Through the diagnosis capability analysis, the designer can preliminarily estimate whether the initial diagnosis scheme meets the requirement for fault detection/isolation coverage, and analyze and compare the data and experience of similar products to determine the alternative diagnostic scheme.

Due to the limitation of autonomous processing capability on a spacecraft, the comprehensive fault detection and isolation can not be achieved. Therefore, in the practical application of aerospace engineering, the typical method is the ground-assisted in-orbit fault diagnosis of ground operation management system based on telemetry information to obtain a universe-Earth–integrated diagnosis scheme. Part of the scheme is to use the spacecraft's own resources for in-orbit autonomous fault diagnosis, and the other part is to use the ground operation management system for diagnosis after acquiring the telemetry information on the ground.

1. In-orbit autonomous diagnosis system

   The in-orbit autonomous diagnosis system should have the following four basic functions: (1) fault detection: identifying whether the system is faulty according to the measurement information; (2) fault location: finding out the failure cause under the given fault conditions, and determining the specific location of the faulty component; (3) fault isolation: preventing the fault propagation to avoid the damage to or the effect on the function of other components; and (4) fault recovery: taking measures to restore all or part of the functions according to the existing in-orbit system status and countermeasures knowledge.

2. Ground operation management system

   Ground operation management system is composed of a series of ground-based auxiliary systems that monitor, diagnose and treat the faulty condition during the satellite orbiting, with the functions such as telemetry data management, condition monitoring, ground simulation, intelligent fault diagnosis, health assessment, reproduction, validation, decision support and recovery. The necessary functions of ground operation management system are as follows: (1) data acquisition and management: the storage, processing and management of telemetry data; (2) state inspection: automatically interpreting and classifying the telemetry data, and giving alarm information at an appropriate level; (3) health assessment: comprehensively and intelligently describing the in-orbit satellite health status by means of information fusion in accordance with telemetry data and state information as well as a wide range of ground data sources; (4) pre-diagnosis: the fault prediction based on performance trend analysis and diagnosis, modeling and pattern knowledge; and (5) decision support.

8.4.3 Design of Embedded Diagnosis

During the in-orbit spacecraft operation, the embedded diagnosis (such as autonomous fault detection and diagnosis system, BIT and performance monitoring) is usually adopted to monitor the performance and operating condition of each component of the spacecraft periodically or continuously, to detect and isolate the faults, to analyze, process and store the test information and to transmit the fault information to the ground.

The design of embedded diagnosis mainly includes detailed BIT design, performance monitoring design, autonomous fault detection and detailed diagnosis system design.

1. Detailed BIT design

   There are generally two BIT modes for spacecraft products, including periodic BIT and power-on BIT.

The periodic BIT is to continuously detect and isolate sneak faults in an operational product, and store and report the related fault information. The watchdog circuit is a common BIT method for spacecraft products.

Power-on BIT is generally applicable to a spacecraft product under non-continuous operating condition. When the product is powered up, the BIT starts to work and also tests the important parameters that cannot be verified in a system running properly. The power-on BIT generally lasts for a few minutes. The successful completion of BIT is followed by an indication of whether the product is normal. If a fault is found, it should be reported.

2. Performance monitoring design

   For the products without BIT, the capabilities of processing the telemetry parameters and the related information should be designed to facilitate real-time performance monitoring (or condition monitoring). The key performance, functional or characteristic parameters are transmitted down to the ground control system by telemetry.

3. Autonomous fault detection and diagnosis system

   On the basis of BIT and performance monitoring design, the spacecraft system can use computer system and related software to acquire the test parameters of each tested unit for autonomous fault detection and isolation, which, together with fault processing and recovery, constitutes an autonomous health management system. Advanced diagnosis design and mature fault feature detection technology can be combined with advanced software modeling and artificial intelligence inference machine to enhance the diagnosis capability and obtain accurate fault detection and isolation results.

8.5 SUPPORTABILITY DESIGN AND PLANNING

Supportability is an attribute of the equipment system composed of equipment and its support resources. As far as only the spacecraft itself is concerned, the spacecraft supportability can be defined as the ability of the spacecraft design characteristics and planned support resources to meet the requirements of continuous and stable in-orbit spacecraft operation during the specified life. Among them, the "design characteristics" can be divided into fault-related maintenance support characteristics and operation-related operational support characteristics. The maintenance support characteristics are generally represented by reliability, maintainability and testability. Obviously, the higher the reliability, maintainability and testability, the better the spacecraft supportability. The operational support characteristics are generally reflected in the in-orbit maintenance operation frequency and the demand degree of support resources such as manpower and material resources. The "planned support resources" of a spacecraft mainly refer to the technical data related to in-orbit operation and maintenance, as well as the human and equipment resources under necessary conditions. In addition, the spacecraft supportability is oriented toward the in-orbit operation and maintenance process, emphasizing the continuity and stability of in-orbit operation.

8.5.1 Supportability Design and Supply

The basic method of supportability design is to increase the available time and continuous operating time and reduce the demand for operational and maintenance support resources by means of reliability design, maintainability design and testability design.

The spacecraft supportability can be divided into in-orbit operational supportability and in-orbit maintenance supportability. In-orbit operational supportability is to reduce the number and time of in-orbit maintenance operations and the demand for maintenance resources through mission analysis, orbit design and payload characteristic design, and to formulate reasonable and efficient in-orbit operation strategies. In-orbit maintenance supportability is to reduce the in-orbit satellite failure rate, improve the satellite mission duration, reduce the demand for maintenance resources and formulate a complete in-orbit failure plan through the design of satellite reliability, maintainability and testability.

The general principles of spacecraft supportability design are as follows:

1. Minimize the maintenance work during the use of a spacecraft, especially the maintenance work that affects the service provided by the spacecraft.

2. Optimize the system design and the in-orbit product operation strategy, reduce the demand for ground support resources (TT&C stations, surveying vessels, people etc.) and improve the usability.

3. Achieve the compatibility with the existing ground operation control, TT&C and operation management support facilities as much as possible, and reduce the demand for special support equipment.

4. The spacecraft in orbit should have certain self-support capabilities, such as fault self-recovery, system reconstruction and autonomous operation, to maintain the continuity of system functions and reduce the dependence on ground support resources.

5. The in-orbit faults affecting the spacecraft mission should be designed and handled according to the principle of the quickest recovery. For example, after the satellite is transitioned to safe mode, high-precision attitude measurement equipment should be selected to shorten the time to return to normal.

6. The ground uplink channel should meet the uplink requirements of in-orbit software maintenance.

At the stages of conceptual design and prototype development, the spacecraft system/subsystem designers should carry out the supportability design and determine the in-orbit operation and maintenance strategies according to the supportability design criteria. In case of technical condition change involving supportability during the stage of flight model development, additional supportability design should be implemented according to the supportability design criteria.

8.5.2 Support Planning

The purpose of support planning is to ensure continuous and stable spacecraft operation in orbit by planning the support activities and support resources for the in-orbit spacecraft lifetime.

The support activities implemented during the in-orbit spacecraft lifetime can be divided into operational support activities and maintenance support activities. For most of the spacecrafts, the support resources are mainly the technical data provided to users. Some spacecrafts require the availability of technical personnel and computer software. In addition, the model developer may establish its own model-dependent support system as part of support planning to ensure stable in-orbit spacecraft operation.

At the stages of conceptual design and prototype development, the system engineering should determine the in-orbit operational support activities (such as phase maintenance and north-south station keeping) in accordance with the spacecraft orbit design and mission analysis. The subsystem designers should determine the in-orbit long-term maintenance strategy according to the technical requirements and the analysis of mission profile. At the same time, the system engineering should analyze whether the existing ground support equipment and facilities meet the requirements of in-orbit support activities. If the requirements cannot be met, they should develop necessary special equipment and software tools, such as ground simulation verification system and model-specific in-orbit data analysis software.

In case of technical condition change at the stage of flight model development, the in-orbit support activities of the spacecraft should be reanalyzed and reconfirmed. For the faults or anomalies that may occur during the in-orbit operation, a detailed in-orbit failure plan based on FMEA results should be formulated before the delivery of the spacecraft, and the maintenance support activities should be planned. Before the spacecraft delivery, the technical documents such as Spacecraft Engineering Manual, In-orbit Long-time Operation Management Requirements and TT&C Monitoring Manual should be prepared for the user in accordance with the requirements for in-orbit spacecraft operation support.

As for the specific support work specified in the in-orbit spacecraft failure plan, the spacecraft system/subsystem designers should develop a list of technical materials related to in-orbit fault handling (such as engineering drawings, technical specifications, technical manuals, technical reports and software documents) for reference during in-orbit fault handling.

8.6 RISK IDENTIFICATION AND CONTROL

The technical risk of a spacecraft refers to the uncertainty that causes the failure to meet the requirements of technical or tactical indicators, endangers the attainment of mission objectives or leads to the mission failure. It is generally measured by the occurrence probability and the consequence severity. The general quality characteristics such as reliability, safety, maintainability, testability and supportability are important technical indexes of a spacecraft, so their technical risk is an important part of the technical risk of the spacecraft.

The work of risk identification includes identifying risk sources and conditions, describing the risk characteristics and determining the risk events that may affect the spacecraft development task. The risk identification should be carried out iteratively and repeatedly throughout the development stage. The approaches of technical risk identification and analysis should be selected according to the spacecraft characteristics and development stage. The proven effective approaches of reliability/safety risk identification include: FMEA, FTA, ETA, PRA, reliability prediction/assessment, margin design analysis, derating design analysis, SCA, WCA, outage analysis and hazard analysis.

The common risks include: single-point-of-failure risks (classes I and II), sneak path risk, worst case risk, structural reliability risk, mechanism reliability risk, mechanical environment risk, thermal risk, electromagnetic compatibility risk, electrostatic risk, micro-vibration risk, plume risk, stay light risk, pollution risk, information flow reliability risk, power supply/distribution reliability risk, derating insufficiency risk, redundancy insufficiency risk, margin insufficiency risk as well as general hazard source risk and fault hazard source risk.

The persons in charge of the products at all levels shall qualitatively and quantitatively assess the identified technical analysis items in accordance with the characteristics of each development stage of the products (including spacecraft), determine the consequence severity, occurrence probability and comprehensive risk level of technical risk items, develop a list of technical risk items at all levels and formulate the control measures. For high-risk items, they shall develop the risk elimination/reduction measures, establish an independent control chart of technical risk items and fully describe the information related to technical risk items. In addition, they shall define the risk mitigation/control measures, risk control plans, risk control results and inspection methods and risk control accountability units, and introduce them into each link of spacecraft development and production. For medium-risk items, they shall treat those items as the key contents of milestone quality control and milestone review, pay close attention to them in each link of follow-up development and production and take effective measures in consideration of the development process to ensure that the spacecraft risk is reduced to an acceptable level before leaving the factory. For low-risk items, necessary monitoring shall be conducted to track and record their subsequent status changes so as to prevent the risk escalation.

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CHAPTER 9

Spacecraft System Testing and Verification

Liu Jie, Liu Hongyu, Yu Songbai, and Zhou Liping

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Spacecraft system verification is the complete process of demonstrating that a spacecraft system meets all the application performance requirements. This verification has two main purposes. The first purpose is qualification, that is, to prove that the spacecraft design is well able to meet all applicable requirements, that is, it fits and meets the planned mission with a margin. This means that the qualification test conditions should exceed the flight environment conditions. The second purpose is acceptance, that is, to prove that the flight parts of a final product have neither defects in the production process and material nor errors in the assembly and integration process. A very important prerequisite for acceptance is to confirm that the flight parts have been built according to the qualified design.

Through the analysis of the connection between test verification method and analysis method, this chapter describes in detail the environmental verification test, electrical-satellite comprehensive measurement and test, electromagnetic compatibility (EMC) comprehensive test and other special test verification methods used in the process of spacecraft system development.

9.1 CONNECTION BETWEEN SPACECRAFT SYSTEM VERIFICATION METHOD AND ANALYSIS

9.1.1 Test Verification Methods

Different verification methods, often in the form of combination, may be used in different construction phases according to different principles and requirements. There are mainly four verification methods:

1. Test. The optimal verification methods include: (1) inputting representative external influences, as excitation, into the hardware, or simulating the external environment (such as vibration, temperature, light or radiation source); (2) measuring the responses with electrical signals (from spacecraft or temporary non-flight probe), physical motion and telemetry data. The results of each test are expected to be within the specified numerical range. The test phase also includes measurement work, for example, measuring the spacecraft's mass characteristics (mass and center of mass) and measuring the calibration parameters to obtain the pointing direction.

2. Analysis. If it is physically impossible or very expensive to verify the design by test verification or other methods, the analytical techniques based on mathematical models and computational simulations can be used. One sub-type of analysis is similarity analysis. On this basis, the spacecraft design can be verified by direct but detailed parameter comparisons with other missions, including the comparisons between each hardware and test conditions.

3. Inspection. Inspection is also a verification method to determine the conformity with the specified structural characteristics, engineering drawings, physical properties and process standards. The inspection may be performed using standard metering and testing equipment in a laboratory. The inspection results include the conclusions drawn by quality assurance personnel in accordance with the prescribed procedures.

4. Design review. Design review is also a verification method. It is intended to prove conclusively that the requirements are being met, by checking the approved design reports, technical specifications and engineering drawings, as well as the evidences supporting the effectiveness of the measures in those documents. One application of design review is to show that the previous equipment designs are qualified, without the need for qualification analysis or testing.

The verification level refers to the level of verification implementation related to the hardware architecture. Depending on the level of detail, the verification levels can be divided into spacecraft level, cabin (or subsystem) level, unit level, equipment level and component level.

9.1.2 Relationship between Analysis and Test Verification

It needs to be recognized that no test can truly and fully represent the environment to which the mission is exposed. For example, the spacecraft thermal control needs to consider many different thermal environments during different mission phases – launch, transfer orbit, apogee, perigee, shadow region and sunlight region. Therefore, it is not practical to test the thermal control performance for each flight state, and it is impossible to conduct long-term tests for some states.

Therefore, analysis is essential in the verification process of spacecraft system. The relationship between analysis and test is shown in Figure 9.1. Based on the analysis and modeling, the test verification is carried out. The result of test verification is fed back to the analysis model for modification and improvement. If the result is consistent with the model, the model is considered to be valid; if they are inconsistent, the anomaly and its reason need to be studied. If the test result is within the specified range but differs from the expectation, the model needs to be studied until the anomaly can be reasonably explained and a new flight prediction can be obtained from the revised model. If the test result is not within the specified range, the specified range can be modified to tolerate a

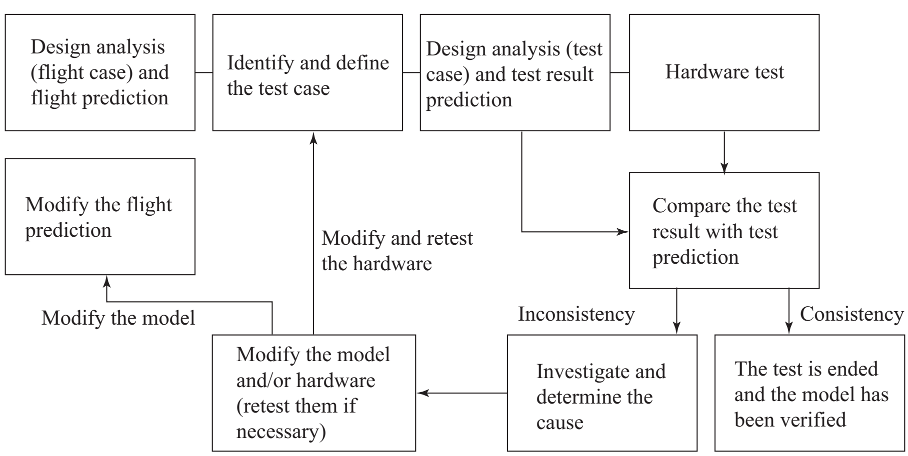

FIGURE 9.1 Relationship between analysis and test.

small inconsistency, or the hardware can be redesigned to a certain extent. At that point, the model would need to be modified to reflect the new design and then rerun to modify the flight prediction. Depending on the severity of the difference and design modification, some tests may need to be redone.

9.2 REQUIREMENTS FOR SPACECRAFT ENVIRONMENT TEST VERIFICATION

Spacecraft environment tests are various adaptability tests of spacecraft products that are done under various space environment conditions, mechanical environment conditions and thermal vacuum environment conditions.

9.2.1 Purpose of Environmental Test

Different environmental tests should be carried out for the spacecraft at different development stages and should achieve different purposes. The environmental tests are generally concentrated at the prototype development stage and flight model development stage, in order to:

1. Find out the errors in the design and process of spacecraft products that may cause the product failure in various environmental tests, so as to verify the correctness and rationality of spacecraft product design and process.
2. Expose the quality defects in the components, raw materials and manufacturing process (processing, assembly, welding and debugging etc.) of spacecraft products through tests, eliminate early failure, improve the operational reliability during the service life and verify whether the quality and performance of products meet the requirements of service environment.

9.2.2 Categories of Environmental Tests

The spacecraft environment tests can be divided into the following three categories:

9.2.2.1 Development Test

Development test is the test of key technologies before the product enters the prototyping stage. Its purpose is to evaluate the design scheme and process scheme at the early development stage, obtain the design and process data, modify the mathematical model and verify whether the design scheme of the product meets the design requirements. Through this test, necessary modification measures can be taken before the start of the qualification test, in order to continuously improve the inherent reliability of the product. The development test is necessary for a new model.

9.2.2.2 Qualification Test

The qualification test is an environmental test carried out at the prototype development stage to check whether the design scheme and process scheme meet the predetermined requirements for strength and performance. It is mainly to verify that the tested product can not only withstand the predicted highest environment, but also has a certain margin. The qualification test shall be carried out with the product which can represent the status of flight model products. If this test is conducted during the prototype development stage, the technical configuration and test documents of the tested product shall meet the qualification requirements of flight model products.

9.2.2.3 Acceptance Test

The purpose of acceptance test is to expose the faults caused by potential defects in the components, raw materials and manufacturing process of a flight model product, so as to eliminate early faults, ensure the operational reliability of the product and confirm that the product meets technical requirements and can be accepted and delivered. The acceptance test is the test performed on a flight model product during the flight model phase to prove that each product delivered for flight is acceptable.

9.2.3 Environmental Test Tailoring

The requirements of environmental test tailoring depend on the test standards. The various environmental test standards and specifications for spacecrafts are usually general baseline test requirements. However, considering the product complexity, the existing technical level, the importance of flight mission, the cost, the acceptable degree of risk and other factors, the baseline requirements for a specific model need to be tailored through the adjustment of test contents, test level and test time. The baseline requirements may become looser for some models and tighter for the other models.

The purpose of tailoring is to improve the cost effectiveness of environmental tests. On the one hand, redundant tests should be avoided to reduce the waste of manpower and financial resources. On the other hand, the test inadequacy should be avoided for fear that the spacecraft defects can not be detected and may lead to the failure of the entire mission. Therefore, the successful tailoring of a test program can not only save time and cost, but more importantly, can ensure the quality and reliability of products.

9.2.4 Retest

A test that needs to be repeated due to the occurrence of abnormal phenomena or other reasons is called a retest. In principle, a retest is necessary in the following four situations.

1. A failure or an anomaly is found in the qualification or acceptance test. If an abnormal phenomenon occurs in the test, the test should be interrupted and the fault should be preliminarily analyzed when the configuration of product software and hardware and the setting state of test equipment are frozen. If the failure is caused by ground test equipment, the test can continue after troubleshooting as long as no over-stress test conditions have been created. If it is caused by a product fault, the test can continue as long as the test does not affect the abnormal area or the fault can be located only with the help of the test. If the preliminary analysis determines that there is no point in continuing the test, the test should be "discontinued" and the product should be repaired. If only minor redesigns or fixes have been made, then just the test

that failed should be redone; if the failure involves a major repair, all acceptance tests should be redone except for the aging test; if the failure involves a number of major repairs, the aging test should also be redone; if significant design changes are made to the product, all qualification tests need to be repeated. This retest phenomenon mostly occurs to the component-level tests and also to the spacecraft system-level tests, but various factors need to be weighed before deciding whether to retest.

2. The product design needs to be modified after qualification test. When a new type of product needs to inherit a previously qualified product, the product often shall go through necessary design modification in order to withstand more severe operational environment. The modifications are divided into "major" modifications and "minor" modifications. If it is a major modification, the product should be requalified like a new product according to the regulations. If it is a minor modification, the flight-class prototype test may be required.

3. The flight product needs to be stored for a long time after acceptance. If the flight product requires long-term storage between acceptance test and launch, the whole or part of the functional and environmental tests shall be redone as prescribed before the product reuse.

4. The qualified product is used for flight. For the reason of funding or scheduling, the qualified product needs to be repaired before flight. The actual or potential overstress part should be replaced after the qualification test, and then the whole or part of the acceptance test should be performed depending on the replacement condition.

9.3 DESIGN OF TEST MATRIX

9.3.1 Design of Qualification Test Matrix for Spacecraft System

9.3.1.1 Qualification Test Matrix

For the basic requirements of spacecraft qualification tests, refer to all the "required" and "evaluation-required" tests specified in Table 9.1. These test requirements can be tailored according to specific circumstances. If required, an "evaluation-required" test may become a "required" test.

9.3.1.2 Qualification Function Test

The qualification function test is mainly to verify whether the mechanical and electrical properties of a spacecraft meet the design requirements and are compatible and harmonious with ground support equipment, and to prove the correctness and effectiveness of test program, telecommands, data processing software and all redundant components or mechanisms.

The state of mechanical devices, valves, deployable mechanisms and separation subsystems in the mechanical function test shall be consistent with that of the spacecraft during launch, orbiting or recovery. It is necessary to verify that the spacecraft can operate under the most severe conditions (including environmental conditions and working time) with positive margins in strengths, torques and related motions and clearances.

TABLE 9.1 Design of Spacecraft Qualification Test Matrix

Test	Suggested Test Sequence	Rocket	Upper Stage	Spacecraft
Inspection[a]	1	R	R	R
Function[a]	2	R	R	R
Pressure/leak detection	3,9,12	R	R	R
EMC	4	R	R	R
Impact	6	ER	R	R
Acoustic[b] or random vibration	7	ER	R	R
Sine vibration	8	ER	ER	R
Thermal equilibrium[c]	10	—	R	R
Thermal vacuum	11	ER	R	R
Modal observation	5	R	R	R
Magnetism	13	—	—	ER

Note 1: "R" stands for a "required" test. It is a mandatory test, because it is effective and highly likely to be done.

Note 2: "ER" stands for an "evaluation-required" test. It is an optional test depending on the specific condition of product development, because it is generally not very effective and less likely to be done. The "evaluation-required" tests shall be assessed on a case-by-case basis. An "evaluation-required" test will become a "required" test if it is evaluated to be effective.

Note 3: "—" means a "not required" test. It is not required because it is not effective and is therefore very unlikely to be done.

[a] If appropriate, the inspection should be done before and after each test (including special tests).

[b] For the compact mass-intensive spacecrafts generally not more than 450 kg, acoustic test can be replaced by random vibration test. The interchangeability between random vibration test and acoustic test should be determined after the evaluation of spacecraft structure characteristics.

[c] It can be combined with thermal vacuum test.

The configuration of a spacecraft shall be consistent with its actual flight state (except for initial explosive devices). The test shall verify the circuit integrity, matching and compatibility of each onboard subsystem with electrical performance requirement, measure the performance parameters, electrical interface, redundancy design, end-to-end channel and polarity and demonstrate the subsystem characteristics when the radio frequency (RF) and sensor signals are input.

The test design shall ensure that all the main and backup components will participate in the test and that all the instruction programs and software programs will be run to the extent practicable. If the components are controlled by sensors, electrical and electronic devices, coding algorithms or computers, they shall undergo the end-to-end performance test.

9.3.1.3 Qualification EMC Test

The qualification EMC test is used to verify the EMC of a spacecraft in the simulated electromagnetic environments during launch, orbiting, orbital transfer and return from orbit, and to ensure appropriate margins.

The tests shall be able to verify that the electrical and electronic components can function properly in their own and external electromagnetic radiation environments under all possible conditions (such as launch, orbiting and return from orbit). Special attention shall be paid to those areas which are in critical state according to analysis.

When a spacecraft is flying in orbit, its surface and internal components are charged with space plasma and charged particles, which may cause electric discharges. An electrostatic generator should be used to simulate the discharge effect on the spacecraft surface and verify whether the electrical and electronic components of the spacecraft can function properly under electrostatic discharge.

9.3.1.4 Qualification Impact Test

The qualification impact test is mainly to verify the spacecraft's ability to withstand the qualification-level impact and function normally, as well as the rationality of the component limit and maximum predicted impact environment specification.

During the test, the support and structure state of the spacecraft should be as similar to the real flight state as possible, so that the amplitude, frequency component and transmission path of impact load are similar to the dynamic response during flight. In the spacecraft-rocket separation impact test, the spacecraft should be connected with appropriate simulators and equipped with real initiating explosive devices and sufficient impact sensors, which can measure the impact response in key areas.

All initiating explosive devices and other devices likely to produce a strong impact, including those not installed on the tested spacecraft, should be triggered at least once. A strong impact is the impact spectrum that is generated at any component location and differs from the response spectrum envelope of all impact sources by ≤6 dB. These shock sources should be triggered two more times to allow for the changes in the test, providing the data required to predict the component limits and the maximum expected impact environment.

9.3.1.5 Qualification Sound Test

The qualification sound test is used to verify the spacecraft's ability to function normally under the qualification-level acoustic environment, as well as the rationality of the component limit and maximum predicted random vibration specification.

The spacecraft is mounted on a flight-type support structure or an appropriate analog part and then placed in a reverberation field or traveling wave field. In the reverberation field test, at least four microphones far from each other are needed to control the acoustic conditions. Generally, the microphones should be placed at half the distance between the specimen and the nearest reverberation chamber wall, but not less than 0.5 m away from the specimen surface and the chamber wall. When the spacecraft support structure is isolated from the ground by shock absorber, the first-order frequency of the whole specimen system shall be lower than the lower limit of the test frequency. The test load shall simulate the total sound pressure level and sound spectrum at the qualification level. If necessary, sufficient vibration sensors shall be installed on the spacecraft to measure the acoustic vibration response and stress at the installation points of the spacecraft structure and components.

9.3.1.6 Qualification Vibration Test

The qualification vibration test is used to verify the spacecraft's ability to function normally under the qualification-level vibration environment, as well as the rationality of the component limit and maximum predicted vibration specification. The vibration tests include sinusoidal vibration test and random vibration test. Generally, for the compact mass-intensive spacecrafts generally less than 450 kg, acoustic test can be replaced by random vibration test.

9.3.1.7 Qualification Pressure and Leak Test

The qualification pressure and leak test is mainly to verify the ability of pressurization subsystem to meet the specified flow rate, pressure and leakage rate. During the test, the spacecraft is placed in a device. Preliminary tests are performed as required to verify the compatibility between the spacecraft and the test equipment, and to ensure the normal operation of equipment control and test functions. When the valves, pumps and motors are operated, the flow rate, leakage and regulation required by pressurization subsystem should be measured, and the test state of pumps, motors and pipelines should be proved to be normal through flow inspection.

During the leak test, the threaded joints of the pressurization subsystem which is not assembled by brazing or fusion welding shall be checked to prove that they conform to the specified assembly torque. Helium gas is charged to the maximum expected working pressure and then the leakage rate is checked by helium mass-spectrometry leak detector. When the total leakage rate of the system is not met, routing inspection is required to find out the leakage position and leakage rate.

9.3.1.8 Qualification Heat Balance Test

The heat balance test is to mainly verify the correctness of thermal analysis model and the ability of thermal control subsystem to keep the entire spacecraft and its components and subsystems within the specified operating temperature range.

The heat balance test can be combined with the thermal vacuum test. This test should be carried out at extreme high temperature and extreme low temperature under the conditions such as all flight seasons, flight attitude, solar incidence angle and eclipse, subsystem operating mode and maximum and minimum caloric values of components. If necessary, the transient and special operating conditions related to thermal analysis shall be added. Special emphasis shall be attached to setting the test conditions for those components (such as cameras and batteries) that have strict requirements for temperature control. A sufficient number of temperature points shall be placed on the spacecraft's internal and external components to validate their thermal analysis and design. The test shall verify the power requirements of all automatic and remotely controlled heaters and coolers and demonstrate their ability to control temperature in accordance with the design requirements.

9.3.1.9 Qualification Thermal Vacuum Test

The thermal vacuum test is mainly to verify the spacecraft's ability to withstand thermal cyclic stress at vacuum and qualification-level temperature, and to check whether the spacecraft performance meets the design requirements in all operating modes.

During the test, the spacecraft is placed on the test bracket in the vacuum container, and all test cables are drawn out of the container through the vacuum sealing flange. The functional measurement/test equipment of the spacecraft shall be completely checked before vacuumization to demonstrate that the whole test system has been in the normal readiness state. The components operating in the launch phase shall be powered to monitor the low-pressure discharge phenomenon during the test pressure reduction. The components that are not operating during the launch phase shall be powered after reaching the specified test pressure. The test pressure of the spacecraft and upper stage is not greater than 6.65×10^{-3} Pa. The pressure of delivery test shall be equal to the atmospheric pressure at the operating altitude.

The thermal vacuum test starts from normal ambient temperature, which will rise to high test temperature and hold at it. After the high-temperature soaking, the temperature is reduced and stabilized to low test temperature. After the low-temperature soaking, the temperature is increased to normal ambient temperature. Therefore, a thermal cycle is formed. If the thermal vacuum test is preceded by a heat balance test, the highest and lowest temperature values of the components with temperature measuring points in the heat balance test under each condition should be expanded by at least 15°C as the test temperature range. The temperature under the last condition of heat balance test should be based on to decide whether the test temperature is shifted to high temperature or low temperature.

9.3.1.10 Qualification Modal Observation Test

In the qualification modal observation test, the dynamic analysis model is mainly verified and modified with the vibration modes and the measured data of frequency and damping. This model is used to analyze and predict the environment of structural loads, which are used to determine the structural margins and verify the rationality of load conditions in structural static test.

The test piece generally consists of a flight structure and its subsystems and components, among which the propulsion subsystem should be filled with simulated liquid. Due to the complexity and feasibility of such a test, the test for a large spacecraft can be carried out in stages according to the needs. In particular, for large launch vehicles, a test program combining ground and flight tests, including the collection and analysis of critical flight data, may be required in order to obtain the necessary data for model validation. If the test proves that the resonance of a fixed flight structure is outside the specified range of modal observation frequency, then the flight structure can be replaced by a mass simulator. If the resonant frequency of the unit is within the frequency range of concern, a dynamic simulator can be used to accurately reflect the dynamic characteristics of the unit. On the other hand, a mass simulator can also be used if the flight structure is subjected to modal observations separately to meet the qualification requirements.

9.3.1.11 Qualification Magnetic Test

For a spacecraft which has strict requirements for magnetic torque and disturbance torque during in-orbit flight, a qualification magnetic test should be conducted. Generally, the magnetic tests are carried out in a zero magnetic field and generally include initial magnetic tests and magnetization/demagnetization tests. In general, the residual magnetic moment of the spacecraft should be measured first. If the residual magnetic moment exceeds the specified index, magnetic compensation should be carried out. Then, stray magnetic moment of the spacecraft should be measured to evaluate the effect of magnetic compensation. If necessary, a demagnetization test is required to eliminate the magnetic moment caused by external field magnetization.

9.3.1.12 Vacuum Discharge Test

Vacuum discharge test is to verify the ability of active electronic components and microwave components to withstand corona, arc and dielectric breakdown in low-vacuum environment. This test is required for the active electronic components and microwave components which start to operate after the satellite launch. For the components that operate intermittently and repeatedly, their restart capability shall also be tested. The vacuum discharge test shall also be carried out on the components that do not work during satellite launch but are closed under atmospheric pressure and allow slow air leakage after entering the orbit. This test can be carried out in conjunction with the thermal vacuum test of the spacecraft.

The test piece should work under the rated current and voltage. In the depressurization process, the vacuum discharge is most likely to occur when the pressure is almost 759 Pa (5.7 torr), so the depressurization speed should be slowed down.

9.3.2 Design of Subsystem Qualification Test Matrix

9.3.2.1 Subsystem Qualification Test Matrix

If some subsystems of the spacecraft require subsystem-level acceptance tests, subsystem-level qualification tests must be done to verify whether the subsystem design meets the requirements.

The boundary condition simulation of subsystem-level test should be more realistic than that of sub-subsystem-level assembly test, such as structural static test. In addition, some component-level tests, such as the tests of interconnected pipelines and cables, can be carried out at the subsystem level if they are hard to implement. Table 9.2 lists the requirements of subsystem qualification test. The unspecified subsystem types can be tested according to the requirements of spacecraft-level test.

9.3.2.2 Subsystem-Level Qualification Structure Static-Load Test

The qualification structure static-load test is mainly to verify whether the strength and stiffness of a structural subsystem can meet the design requirements when the subsystem is exposed to the environment (such as temperature, humidity, pressure and load) expected to exist during its operating life and the qualification margin.

The test-oriented structural part should be consistent with the flight-oriented structural part in terms of structural form, material and manufacturing process. In order to meet the requirements of structural strength and stiffness, the areas that need to be reprocessed or strengthened should also be reflected in the flight-oriented structural parts.

TABLE 9.2 Design of Subsystem Qualification Test Matrix

Test	Spacecraft Subsystem	Space Test Equipment	Rocket Subsystem	Payload Fairing
Mechanical function	ER	ER	ER[a]	R
Static load	R	ER[a]	R	R
Sound or vibration	Er[b]	ER[b]	ER[b,c]	R[d]
Thermal vacuum	ER	R[e]	ER[c]	ER
Separation and deployment	R	—	—	R

Note 1: "R" stands for a "required" test. It is a mandatory test, because it is effective and highly likely to be done.
Note 2: "ER" stands for an "evaluation-required" test. It is an optional test depending on the specific condition of product development, because it is generally not very effective and less likely to be done. The "evaluation-required" tests shall be assessed on a case-by-case basis. An "evaluation-required" test will become a "required" test if it is evaluated to be effective.
Note 3: "—" means a "not required" test. It is not required because it is not effective and is therefore very unlikely to be done.

[a] Not required if it is not performed at other assembly levels.
[b] For mass-intensive subsystem, acoustic test is replaced by random vibration test.
[c] Required for a subsystem equipped with key equipment (such as guidance unit), and not required for spacecraft level.
[d] The acoustic test is required.
[e] Optional for spacecraft level.

The test-oriented support and loading fixture should simulate the actual boundary conditions of adjacent structural sections to the greatest extent. A yield design load and an ultimate design load, both in the form of static load, are applied to the structure, and the strain and deformation data of key areas are measured and recorded. Under the yield design load, the structure should not yield or deform. Under the extreme design load, the structure should not become unstable or fracture.

The qualification test conditions should include design margins and take into account possible failure modes. The test conditions should consider the worst case of the combination of acceleration, vibration, pressure and temperature. The effect of temperature on material strength reduction is often considered in terms of equivalent mechanical force load.

9.3.2.3 Subsystem-Level Qualification Separation Test

The qualification separation test is mainly to verify whether the separation velocity, acceleration and angular motion of separation device, the disengagement time and clearance of separation hardware, the deformation and load of flexible body, the number of fragments and the impact magnitude of explosive device meet the design requirements. The separation test data are also used to verify the correctness of separation analysis method and basic assumptions, and to predict whether the separation system meets the mission requirements in the worst case. For a payload fairing using a high-energy separation subsystem, the structural integrity of the fairing and its attachments should also be verified under the separation impact load.

9.3.3 Design of Component Qualification Test Matrix

9.3.3.1 Component Qualification Test Matrix

The basic requirements for typical component qualification tests, as shown in Table 9.3, include "required", "evaluation-required" and "not required" tests.

Usually, component qualification tests should be performed at the component level, but the tests of some components, such as interconnected pipes, RF circuits and cables, may be partially or completely tested at the subsystem level or the spacecraft level. If the movable mechanical components or other components have static or dynamic liquid interfaces or need to be pressurized during operation, these conditions shall be simulated in the component qualification tests. The component performance shall, to the greatest extent, meet the design requirements in the whole range of qualification environment tests. Upon completion of all required qualification tests, the qualified components shall be disassembled for inspection.

9.3.3.2 Component Qualification Function Test

The functional tests are primarily electrical and mechanical tests, including the measurement of electrical continuity, stability, response time, collimation, pressure, leakage or other special functional characteristics. In an electrical performance test, the expected voltage, impedance, frequency, pulse and waveform should be applied to the electrical interfaces of the components, including all redundant circuits. These parameters shall vary within the required range according to the desired flight operation sequence. The component output shall be measured to verify whether the component is working properly. In the mechanical function test, the technical state and ON/OFF state of the component shall correspond to the state of the component exposed to the environment. The torque-angle and time-angle – and rigidity, damping, friction and disconnection characteristics for some components – shall be measured. For the moving mechanical components containing redundancy, their performance shall be checked against requirements in each redundancy mode. For the components with other special functions, the thermal, optical and magnetic functional tests may also be carried out.

9.3.3.3 Qualification Thermal Cycle Test of Electrical and Electronic Components

This test is mainly to verify the ability of electrical and electronic components to withstand the qualification-level thermal cycle environment and the thermal cycle environment applied to flight components during the acceptance test.

The first-cycle test starts from room temperature (normal ambient temperature), at which the components are powered for performance testing. After the end of the test, the temperature begins to rise. When entering the allowable deviation range of the qualification temperature, the temperature should be controlled to a stable point. Then the component is powered off. After at least 30 minutes, it is hot-started. When the temperature rises, it should be controlled within the allowable deviation range and should be kept stable. The performance test shall be completed within the specified duration of hot dip at the high-temperature end. And then the temperature starts to go down. Before reaching the allowable deviation range of the qualification temperature, the module is powered off and

TABLE 9.3 Design of Component Qualification Test Matrix

Test Type	Suggested Test Sequence	Electrical and Electronic Components	Antennas	Mechanical Moving Components	Solar Arrays	Batteries	Valves or Propulsion Components	Pressure Vessel	Thruster	Thermal Components	Optical Components	Structural Components
Inspection[a]	1	R	R	R	R	R	R	R	R	R	R	R
Function[a]	2	R	R	R	R	R	R	R	R	R	R	ER
Leak detection[b]	3,6,12	ER	—	R	—	R	R	R	R	R	—	—
Impact	4	R	ER[c]	ER[c]	ER[c]	ER[c]	ER[c]	ER	ER[c]	ER[c]	ER[c]	ER
Vibration	5	R	R[d]	R	R[d]	R	R	R	R	R	R[d]	ER[e]
Sound	5	ER	R[d]	—	R[d]	—	—	—	—	—	R[d]	—
Acceleration	7	ER	ER	ER	ER	ER	—	ER	—	—	ER	R
Thermal cycle	8	R	ER	ER	ER	ER	ER	ER	ER	ER	ER	—
Thermal vacuum/vacuum discharge[f]/microdischarge[g]	9	R	R	R	R	R	R	R	R	R	R	—
Climate	10	ER	ER	ER	ER	ER	ER	ER	ER	ER	ER	ER
Inspection pressure[h]	11	ER	—	ER	—	R	R	R	R	ER	—	—
EMC	13	R	R	ER	ER	ER	ER	ER	ER	ER	ER	—
Life	15	ER	ER	R	ER	R	R	ER[i]	R	ER	ER	ER[j]
Burst pressure[h]	16	ER	—	—	—	R	ER	R	ER	ER	—	—
Magnetism	14	ER	—	—	—	—	—	—	—	—	—	—

Note 1: "R" stands for a "required" test. It is a mandatory test, because it is effective and highly likely to be done.

Note 2: "ER" stands for an "evaluation-required" test. It is an optional test depending on the specific condition of product development, because it is generally not very effective and less likely to be done. The "evaluation-required" tests shall be assessed on a case-by-case basis. An "evaluation-required" test will become a "required" test if it is evaluated to be effective.

Note 3: "—" means a "not required" test. It is not required because it is not effective and is therefore very unlikely to be done.

[a] If feasible, required to be done before and after each test, including necessary special tests.

[b] Required for sealed or pressurized components.

[c] Required when the *g* value is used to represent more than 0.8 times the frequency value (Hz) in the maximum expected impact response spectrum.

[d] Either the sound or vibration test is required and the other is optional.

[e] Required if the structural parts have low fatigue margins or the static-strength qualification test is not carried out.

[f] The components operating in the launch phase shall be powered, and the low-pressure discharge phenomenon shall be monitored during the process of test pressure reduction. The components that are not operating during the launch phase shall be powered after reaching the specified test pressure.

[g] A microdischarge test is required for the components and devices that have reached certain RF power.

[h] Required when the component is pressurized.

[i] For pressure components (except for corrugated pipes and other flexible fluid devices or pipelines), the life test is optional.

[j] For a pressurized structure, the pressure cycle test is required.

the temperature continues to drop. When reaching the allowable deviation range of the qualification temperature, the temperature should be controlled to achieve stability. Then the cold test is started. The rising temperature should be controlled within the allowable deviation range and shall be kept stable. The performance test shall be completed within the specified hot-dip duration at the low-temperature end. Then the temperature rises to room temperature to complete the cycle. The last cycle and the intermediate cycle (except for cold start and hot start) are operated in the same way as the first cycle.

9.3.3.4 Component-Level Qualification Thermal Vacuum Test

This test is mainly to verify the ability of a component to endure the qualification-level thermal vacuum and the thermal vacuum applied to the flight component during its acceptance test.

The thermal control coating on the component surface shall be the same as that on the flight component. The component shall be mounted in a vacuum container on the test support or heat sink in a manner similar to actual installation on the spacecraft. The position of test temperature control point should be consistent with that of flight telemetry temperature point. For the components cooled by the substrate mounting surfaces, the control points shall be selected on the component substrates or on the heat sinks. For the components that rely mainly on radiation heat transfer, the control points shall be selected in the representative areas of the component shells. The proportional relationship between the conductive heat transfer from the component to the temperature-controlling heat sink and the radiative heat transfer from the component to the environment shall be controlled to be the same as the calculated result in the flight environment.

For the components operating in the launch phase, the low-pressure discharge phenomenon shall be monitored during the process of test pressure reduction. When the atmospheric pressure gets close to the test pressure, observation shall be made to determine whether there is a microdischarge phenomenon. The components not operating during the launch phase shall be powered after reaching the test pressure.

9.3.3.5 Component-Level Qualification Vibration Test

In order to verify the ability of a component to withstand the qualification-level vibration, the component can be mounted on a fixture through the specified mounting points and tested on each of the three orthogonal axes. The test magnitude is controlled at the joint surface between the component and the fixture. The qualification and acceptance vibration tests shall use the same test fixture. If the component is connected with cables and pipes, the connection state before the first fixed point shall be as consistent with the flight state as possible. For the components that need to operate under pressure during the spacecraft launch, they shall be pressurized during the test to simulate their actual working condition, and their pressure drop shall be monitored.

9.3.3.6 Component-Level Qualification Acoustic Test

The component with the launch configuration is installed on a flight support structure or reasonable simulation structure and then placed in the sound field. The sound field should have sufficient volume, sound energy, sound spectrum forming ability and modal density to excite the acoustic vibration response of the component. During the test in the reverberation field, at least four microphones far from each other are needed to control the acoustic conditions. Generally, the microphones should be placed at half the distance between the specimen and the nearest reverberation chamber wall, but not less than 0.5 m away from the specimen surface and the chamber wall.

9.3.3.7 Component-Level Qualification Impact Test

In order to verify the ability of a component to withstand the qualification-level impact, the component can be mounted on a fixture through the specified mounting points. The test magnitude is controlled at the joint surface between the component and the fixture. The qualification and acceptance impact tests shall use the same test fixture. If the component is installed on a real structure or a structure with almost real dynamic characteristics, the test should be more realistic than that when the component is installed on a rigid structure such as a shaking table or a slide table. If the component is used with a bracket or shock absorber, the component under test should also carry a bracket or shock absorber. The test magnitude should be controlled at the connection surface between the bracket (or shock absorber) and the fixture. The low frequency limit of the impact response spectrum should be less than 0.7 times the natural frequency of the shock absorber.

9.3.3.8 Component-Level Qualification Leak Test

The leak test of a component is generally carried out after the function test and also after the vibration test and pressure test.

The leak detection method is selected according to the allowable leakage rate of the component, and the leakage rate is detected according to the required threshold, resolution and accuracy. The leak detection should consider the change of leakage rate with pressure difference and temperature. The leakage test should be carried out when the pressure difference of the component is greater than the maximum operating pressure difference or less than the minimum operating pressure difference to ensure a proper leakage qualification margin.

9.3.3.9 Component-Level Qualification Pressure Test

It is necessary to verify whether the pressurized component has adequate margins to ensure that structural failure does not occur until the designed burst pressure is reached or that excessive deformation does not occur at the maximum predicted operating pressure.

At least one pressure test shall be conducted for the specimens such as pressurized structures and pressure components. In the event of any leakage, a permanent deformation or distortion beyond specified dimensional tolerances or other types of failure, the component shall be deemed to have failed the pressure test.

The pressure cycle test shall be carried out on the pressurized structure and pressure vessel. When the external load and internal pressure are applied simultaneously in the test process, the loading requirement shall be determined according to their relative values and the instability effect of external load stress. No external load will be required if the compressive stress in the test envelops the maximum operational combined tensile stress.

9.3.3.10 Component-Level Qualification Acceleration Test

To verify the ability of a component to withstand the qualification-level acceleration, the component can be mounted on a fixture through the specified mounting points and tested on each of the three orthogonal axes. During the test, the acceleration direction of the component shall be the same as that during flight, and the value of the acceleration applied to the center of gravity of the component shall be equal to the value of the test acceleration. The acceleration gradient along the component shall not cause the acceleration of key parts of the component to fall below the qualification-level requirement.

9.3.3.11 Component-Level Qualification Life Test

The life test is to verify the long-term service time of a component during its operating life and its ability to keep its performance within the specified range and with appropriate margins.

The life test shall be performed on the components that may have wear, performance drift, fatigue failure or degraded performance. The test shall simulate the environmental conditions to which the component may be subjected during its operating life. The environmental conditions shall be selected taking into account the operating requirements of the component at EOL and its main life characteristics. The normal environment, thermal environment and thermal vacuum environment can be used to evaluate the typical failure modes including wear and performance drift, and the pressure, heat and vibration environments can be used to evaluate the fatigue failure modes.

9.3.3.12 Component-Level Qualification EMC Test

The EMC test is mainly to verify the ability of a component in normal operation to resist external electromagnetic interference with a certain safety margin. At the same time, it shall verify whether the electromagnetic energy of radiation emission and conducted emission of the component will cause interference to other components and affect their normal work.

9.3.3.13 Component-Level Qualification Magnetic Test

Generally, the magnetic test shall be carried out in the zero magnetic field. During the test, the component is placed on a two-axis turntable at the center of main coil of zero magnetic equipment, and the magnetic field intensity is measured with magnetometer in two-axial directions by changing the angular position step by step, and then the magnetic moment of the component is inverted. If necessary, the demagnetized test can be carried out in accordance with the specified demagnetization frequency and demagnetization time, but after the magnetization test.

9.3.3.14 Component-Level Qualification Climate Test

The climate test is mainly to verify the survivability and operational capability of a component exposed to various climatic conditions, including humidity, sand and dust, rain, salt spray and explosive atmosphere.

The climatic and environmental conditions that the components are subjected to during manufacturing, testing, transportation, storage, launch preparation, launch and return include humidity, sand and dust, rain, salt spray and explosive atmosphere. In order to avoid the environmental effects that may result from extreme ground climatic conditions, necessary program controls and special support equipment should be in place. Only those uncontrollable environments need to be considered in the design and testing.

9.3.3.15 Component-Level Vacuum Discharge Test

The vacuum discharge test is to verify the ability of active electronic components and microwave components to withstand the corona, arc and dielectric breakdown in the low-vacuum environment. This test shall be carried out on the active electronic components and microwave components which have been operating since the satellite launch. For the components that work intermittently and repeatedly, their restart capability also needs to be tested. This test also applies to the components that do not operate during satellite launch but are closed under atmospheric pressure and allow slow air leakage after entering the orbit.

9.3.3.16 Component-Level Microdischarge Test

The microdischarge test is to verify the possibility of microdischarge in the RF components and equipment, and to obtain the power capacity and microdischarge threshold.

The microdischarge test is generally implemented by adding a microwave pulse signal, whose duty ratio can be adjusted between 1% and 10%. If the testing conditions permit, the continuous-wave method can also be used. In order to simulate the actual hot working condition, the microwave pulse signal should maintain a certain pulse bottom level, so that the average pulse power can reach the actual rated power (e.g., by increasing the pulse signal width).

The electrical performance of the device under test (DUT) should be tested before and after the microdischarge test to help determine whether the microdischarge phenomenon will occur.

9.3.4 Spacecraft System Acceptance Tests

For the basic requirements of spacecraft acceptance tests, refer to all the “required” and “evaluation-required” tests specified in Table 9.4. These test requirements can be tailored for each model according to specific circumstances. If required, an “evaluation-required” test may become a “required” test. Some special tests (such as collimation measurement, instrument calibration, antenna pattern and quality characteristics) shall also be carried out as part of the acceptance tests. If the spacecraft is controlled by an onboard data processing device, the flight computer software shall be housed in the onboard computer during the test and the software operation shall be verified to the greatest extent.

The spacecraft acceptance test items are basically the same as qualification test items, except for the test magnitude slightly decreased. These tests are mainly to expose the faults caused by potential defects in the components, raw materials and manufacturing process of a flight model product, so as to eliminate early failures.

TABLE 9.4 Design of Spacecraft Acceptance Test Matrix

Test	Suggested Test Sequence	Rocket	Upper Stage	Spacecraft
Inspection[a]	1	R	R	R
Function[a]	2	R	R	R
Pressure/leak detection	3,7,11	R	R	R
EMC	4	ER	ER	ER
Impact	5	ER	R	R
Acoustic[b] or random vibration	6	ER	R	R
Sine vibration	8	ER	R	R
Thermal vacuum	9	ER	R	R
Thermal equilibrium[c]	9	—	R	R
Magnetism	11	—	–	ER
Storage	Random	ER	ER	ER

Note 1: "R" stands for a "required" test. It is a mandatory test, because it is effective and highly likely to be done.

Note 2: "ER" stands for an "evaluation-required" test. It is an optional test depending on the specific condition of product development, because it is generally not very effective and less likely to be done. The "evaluation-required" tests shall be assessed on a case-by-case basis. An "evaluation-required" test will become a "required" test if it is evaluated to be effective.

Note 3: "—" means a "not required" test. It is not required because it is not effective and is therefore very unlikely to be done.

[a] If appropriate, the inspection should be done before and after each test (including special tests).

[b] For the compact mass-intensive spacecrafts generally not more than 450 kg, acoustic test can be replaced by random vibration test. The interchangeability between random vibration test and acoustic test should be determined after the evaluation of spacecraft structure characteristics.

[c] To be done on a spacecraft in maiden flight, possibly in combination with thermal vacuum test.

9.3.5 Subsystem Acceptance Test

9.3.5.1 Subsystem Acceptance Test Matrix

In order to ensure the validity of the test, the subsystem-level acceptance test should be conducted if it is more effective. The subsystem-level test requirements are usually determined according to the spacecraft-level test requirements. Table 9.5 lists the subsystem acceptance test requirements. The unspecified subsystem types can be tested according to the spacecraft-level test requirements.

9.3.5.2 Subsystem-Level Acceptance Structure Inspection-Load Test

This test is to expose potential defects in the material, processing and manufacturing quality of the structure under the inspection load.

All bonded, composite or sandwich structures shall undergo the inspection load test. No inspection load test will be required if a proven non-destructive test method is used and clear pass-failure criteria are available.

This test is carried out on a flight structure. The test-oriented support and loading fixture should simulate the actual boundary conditions of adjacent structural sections to the greatest extent. If several test load conditions are to be applied, a method of sequential loading should be determined to ensure that the load will be progressively increased in

TABLE 9.5 Design of Subsystem Acceptance Test Matrix

Test	Spacecraft Subsystem	Space Test Equipment	Rocket Subsystem	Payload Fairing
Mechanical function	ER	ER	ER	ER
Static load	R[a]	ER	ER	ER
Sound or vibration	—	ER[b]	ER[b,c]	ER
Thermal vacuum	—	R[d]	ER[c]	—
Inspection pressure and leak detection	R	ER	R	—

Note 1: "R" stands for a "required" test. It is a mandatory test, because it is effective and highly likely to be done.

Note 2: "ER" stands for an "evaluation-required" test. It is an optional test depending on the specific condition of product development, because it is generally not very effective and less likely to be done. The "evaluation-required" tests shall be assessed on a case-by-case basis. An "evaluation-required" test will become a "required" test if it is evaluated to be effective.

Note 3: "—" means a "not required" test. It is not required because it is not effective and is therefore very unlikely to be done.

[a] Required for bonded structures and composite structures.

[b] For mass-intensive subsystem, acoustic test is replaced by random vibration test.

[c] Required for a subsystem equipped with key equipment (such as guidance unit), and not required for spacecraft level.

[d] Optional if it has been done at spacecraft level.

sequence under each condition. The strain and deformation data obtained in key areas before loading and after unloading and at several intermediate loading levels should be measured and recorded for post-test data analysis.

9.3.5.3 Subsystem-Level Acceptance Pressure and Leak Test

The pressure and leak test is mainly to verify that the (liquid) pressurization subsystem can meet the requirements for flow rate, pressure and leak rate.

The pressurization subsystem is placed in a device during the test. Before the test, all joints, mating surfaces, plugs and pipes of this subsystem shall be checked for external leakage rate, which shall meet the leakage rate required by the mission. Then the structure and pressure components shall go through at least one cycle of pressure test. They shall be free from the leakage higher than the allowable leakage rate limit, the permanent deformation or distortion out of the dimensional tolerances specified in the drawings or other forms of failure.

9.3.6 Component Acceptance Test

The component acceptance test matrix, as shown in Table 9.6, includes the "required", "evaluation-required" and "not required" tests.

Compared with the component-level qualification test matrix, the acceptance test matrix has one more test, namely running-in test, which is mainly to detect the defects in the material and manufacturing quality of a mechanical component at the beginning of life (BOL), and to check its running-in condition to ensure that it can operate in a stable, coordinated and controlled state.

TABLE 9.6 Design of Component Acceptance Test Matrix

Test Type	Suggested Test Sequence	Electrical and Electronic Components	Antennas	Mechanical Moving Components	Solar Arrays	Batteries	Valves or Propulsion Components	Pressure Vessel or Its Components	Thruster	Thermal Components	Optical Components	Structural Components
Inspection[a]	1	R	R	R	R	R	R	R	R	R	R	R
Function[a]	3	R	R	R	R	R	R	R	R	R	R	R
Leak detection[b]	4,7,12	ER	—	R	—	R	R	R	R	ER	—	—
Impact	5	ER[c]	ER	ER	–	ER	ER	–	ER	–	ER	–
Vibration	6	R	R[d]	R	R[d]	R[e]	R	R	R	R	ER[f]	–
Sound	6	ER	R[d]	—	R[d]	—	—	—	—	—	–	—
Thermal cycle	8	R	ER	ER	ER	ER	ER	–	ER	ER	ER	—
Thermal vacuum/vacuum discharge[g]/microdischarge[h]	9	R[i]	R	R[j]	R	R[e]	R	ER	R	R	R	—
Running-in	2	—	—	R	—	—	R	—	R	—	—	—
Inspection pressure	10	—	—	ER	—	R	R	R	ER	—	—	—
Inspection	11	—	ER	ER	—	—	ER	ER	ER	ER	ER	ER[f]
EMC	13	ER	ER	—	ER	ER	ER	—	—	—	—	—
Magnetism	14	ER	—	—	—	—	—	—	—	—	—	—

Note 1: "R" stands for a "required" test. It is a mandatory test, because it is effective and highly likely to be done.

Note 2: "ER" stands for an "evaluation-required" test. It is an optional test depending on the specific condition of product development, because it is generally not very effective and less likely to be done. The "evaluation-required" tests shall be assessed on a case-by-case basis. An "evaluation-required" test will become a "required" test if it is evaluated to be effective.

Note 3: "—" means a "not required" test. It is not required because it is not effective and is therefore very unlikely to be done.

[a] If feasible, required to be done before and after each test, including necessary special tests.

[b] Only required for sealed or pressurized components.

[c] Required for high impact value.

[d] Either the sound or vibration test, whichever is more suitable, is required and the other is optional.

[f] Required for a bonded or composite structure.

[e] Not required for the batteries that can no longer be charged after test.

[g] The components operating in the launch phase shall be powered, and the low-pressure discharge phenomenon shall be monitored during the process of test pressure reduction. The components that are not operating during the launch phase shall be powered after reaching the specified test pressure.

[h] A microdischarge test is required for the components and devices that have reached certain RF power.

[i] Optional for sealed or low-power components.

[j] Except for the hydraulic components of the rocket.

Under typical working loads, speeds and environments, the component whose sensitive parameters are monitored should operate for a specified period of time. To ensure the detection of early failure in valves, thrusters and other products, the best way is to choose an appropriate number of operating cycles rather than operating time. The functional cycles should be performed at room temperature. For thrusters, a cycle refers to the thermal ignition process including start, steady-state operation and shutdown. For the thrusters using hydrazine propellant for thermal ignition, all hydrazine stains should be removed from the flight-specific valves after the test firing.

9.4 SPACECRAFT DESIGN TEST VERIFICATION

9.4.1 Structural Design Test Verification

9.4.1.1 Method of Structural Design Verification

The methods of structural design verification include one or several of the following methods: analytical verification, inspection verification, analogical verification and experimental verification. The satisfaction of most of the design requirements needs to be verified by a combination of several methods, such as the combination of analytical verification and experimental verification or the combination of analytical verification and analogical verification.

1. Analytical verification is to analytically verify certain requirements that the structure needs to meet. At the stage of satellite structure design, analytical verification is more important than the other verification methods. Through analysis, a detailed design concept, including structural configuration, structural material selection and structural connection method, can be defined to estimate the mass characteristics of the structure and the costs of design, production, test and operation. This analysis is the first step of structural design verification, and can be used to explain whether the structure meets the requirements for strength, rigidity, natural frequency, dynamic envelope and mass characteristics.

2. Inspection verification is to verify the technical conditions of product drawings. It is usually carried out at the manufacturing, qualification, acceptance, final assembly and launch stages, in order to verify the requirements of structural dimensional accuracy, mass characteristics, surface state, physical characteristics (such as electric conduction, insulation and heat conduction) and mechanical and electrical interfaces. Inspection verification is the direct verification of a product, so it can only be carried out during or after the manufacturing of a structural product. This means that the inspection verification needs to take into account the cost of construction. In a sense, inspection verification is the verification of a product rather than the verification of a design.

3. Analogical validation is usually combined with analysis to demonstrate that one structure is similar to another structure that has been qualified under equivalent or more stringent criteria. It includes the assessment and review of the status and application of a structural product and of its previous test data, as well as the comparison

of previous test level with new specific requirements. Analogical verification is a very effective verification method for a structure with good inheritance. This verification method can not only reduce the technical risk, but also save the development cost and speed up the development progress.

4. Test verification is generally used to verify whether a structural product can meet the design requirements during and after experiencing the relevant environment.

9.4.1.2 Verification of Structural Static Strength

For different satellites or different development stages of the same satellite, the purpose, contents and role of static strength verification are different. In general, the static strength verification has the following characteristics at different development stages.

1. Conceptual design stage

   The static strength verification of a structure at this stage is mainly based on the static analysis of the structure and the research test of local structure. Its purpose is to provide a basis for the selection of design concept and the determination of material and sectional dimensions of the main load-bearing member. The local structures that need to be tested refer to the key parts whose static strength can't be easily determined by analysis, such as the embedded parts under load and the composite load-bearing structure. The static analysis model is relatively simple and rough. It mainly focuses on the analysis of the whole structure and adapts to the change of the structural concept.

2. Prototype development stage

   At this stage, the structural design concept has been basically defined, and the structural static strength verification is mainly based on the static structural analysis (strength check) and the qualification test of the main load-bearing structure.

   The main contents of structural strength analysis include: analyzing various load conditions, explaining the design rationality and pointing out the weak links and deficiencies in the design so as to improve the design. Analytical verification is also a preparation for the qualification test. According to the analysis results, the test conditions can be reasonably selected, so that the ability of a structure to withstand the static load can be more comprehensively assessed with fewer test conditions. The analysis model is more detailed and can accurately reflect the actual situation of the structure. At the prototype development stage, another content of static structural analysis is the pre-analysis of static test to ensure that the test can effectively simulate the actual forces on the structure.

3. Flight model development stage

   The static strength verification of the structure at this stage is mainly based on static structural analysis. The main work of static structural analysis is to correct the analysis model obtained at the prototype development stage in accordance with the results of mechanical tests (including static and dynamic tests and component development tests) and in full consideration of the problems in the tests and the prototype design modification in the flight model design, so as to verify whether the flight model design of the structure meets the design requirements.

9.4.1.3 Verification of Structural Dynamic Characteristics

Considering the problem of dynamic characteristics, the structural design of a satellite should meet the requirements of both natural frequency and dynamic strength. The requirements of natural frequency mainly focus on the power coupling between satellite and rocket: for a satellite/rocket system, a reasonable matching relationship should exist between the dynamic stiffness of the satellite and that of the carrier rocket. If the natural frequency of the satellite is unreasonably designed relative to the rocket, a resonance phenomenon will occur in the process of active flight and the satellite will be affected by a large dynamic load.

For different satellites or different development stages of the same satellite, the purpose, contents and role of dynamic characteristics verification are different. In general, the dynamic characteristics verification has the following characteristics at different development stages:

1. Conceptual design stage

   The verification of dynamic characteristics design is mainly based on the modal analysis of the whole spacecraft. Its purpose is to provide a basis for the selection of design concept and the determination of material and sectional dimensions of the main load-bearing member. The analysis model is relatively rough, mainly focusing on the analysis of the whole structure and adapting to the change of the structural concept.

2. Prototype development stage

   The verification of dynamic characteristics design is mainly based on modal analysis, response analysis and qualification test. The modal and response analysis of the whole spacecraft is mainly to point out the unreasonable design aspects for improvement or explain whether the design meets the requirements, and to prepare for modal test and qualification test. Structural dynamic qualification test is one of the keys to structural design verification and is directly related to the success of a structural design. Before the test, the satellite/rocket coupled dynamic analysis should be carried out to reasonably define the concave control conditions of test load.

On the basis of dynamic analysis, the dynamic characteristics of a prototype structure can be verified through the following tests:

1. Modal test

   Modal test is an effectiveness verification test. Its main purpose is to define the inherent characteristics of a structure (natural frequency, modal damping and vibration mode etc.), to verify and modify the mathematical model of the satellite structure, to check the effectiveness of dynamic response prediction results and to verify whether the modal characteristics of the structure meet the design requirements.

The modal test methods include hammering method, single-point sinusoidal method, single-point random method, multi-point sinusoidal method and multi-point random method. An appropriate method should be selected according to the satellite structure characteristics, test equipment (hardware, software) condition, funds and progress, as well as test purpose.

2. Qualification test

The qualification test is usually carried out on a structural qualification part (while most of the space-borne devices are used as balance weights), mainly to qualify the dynamic strength of the structure. The qualification test is mainly to: verify the rationality of satellite structure design and assess the ability of a satellite structure to withstand the qualification-level vibration, that is, to qualify the dynamic strength of the structure so as to provide a basis for prototype design modification and flight model design; examine the components (such as pipeline system) that cannot be reasonably assessed through the component-level tests; expose the defects in the material and manufacturing process of satellite structure; obtain the response parameters of each cabin section; and prepare for determining the vibration test conditions of flight model. Depending on the different purposes of sinusoidal vibration qualification test, the configurations of structural qualification parts are also different.

The sinusoidal vibration test conditions are generally adopted in the vibration tests. The test conditions are defined according to the environmental specifications and the results of satellite-rocket coupling analysis. The general environmental test conditions are shown in Table 9.7. The tests in each excitation direction are divided into pre-vibration-level test, characteristic-level test, acceptance-level test and qualification-level test.

In the acceptance test and qualification test, some frequency bands (mainly including the satellite's first-order resonance band) are usually under concave control. The concave control is to lower the test conditions in some frequency bands in order to avoid the damage to the structure or onboard equipment caused by excessive response. The reason why the test conditions can be concave is that the conditions specified in the test specifications are not the actual environmental conditions that the satellite must undergo, but the linear envelope of the maximum values of various environments. The actual environmental conditions experienced by the satellite are

TABLE 9.7 Sinusoidal Vibration Test Conditions

Frequency Range	Acceptance Level	Qualification Level	Pre-vibration Level	Characteristic Level
5–7 Hz	3.0 mm	6.0 mm	0.1 g	0.2 g
7–100 Hz	0.6 g	1.2 g		
Scanning speed	4 oct/min	2 oct/min	4 oct/min	4 oct/min

of high magnitude only in some bands (usually near the frequency band close to the natural frequency of the satellite/rocket combination, and near the frequency of the exciting force acting on the satellite/rocket combination). For the satellite structure, the concave control magnitude of a test is determined according to the following principles: the stress on the main structure under test is not greater than that on the main structure under static load; and the vibration magnitude should not be lower than the result of satellite/rocket coupling analysis. Generally, the two principles shall not be contradictory. If they are contradictory, it means that the dynamic stiffness matching between the satellite and the launch vehicle is unreasonable.

The vibration tests in the same direction can generally be carried out in the following sequence: pre-vibration-level test, fixed-frequency calibration test, first characteristic-level test, acceptance-level test, second characteristic-level test, qualification-level test and third characteristic-level test. Such a sequence is reasonable and is even necessary for the spacecraft vibration test.

The purpose of pre-vibration-level test is to: check the harmony between vibration control system and measurement system; check the installation condition of the test fixture and the conducting condition of the sensor; check whether the test fixture meets the test requirements; initially understand the dynamic characteristics of the satellite structure, judge whether the sensor installation position related to concave control is appropriate and prepare for the definition of concave control conditions.

The fixed-frequency calibration test is used to calculate the amplification coefficient.

The characteristic-level test is used to study the dynamic response characteristics of a satellite structure. This test is very important for structural vibration test. On the one hand, it can help determine the concave level of a high-level test (acceptance-level test or qualification-level test). On the other hand, the comparison of the response curves of characteristic-level tests before and after the high-level test (acceptance-level test or qualification-level test) can effectively judge whether the state of the satellite structure has changed, that is, whether the main satellite structure has something wrong after the high-level test. In general, if the fundamental frequency of the satellite has an obvious forward shift, the main structure of the satellite may have had a problem, such as connection looseness, fastener fall-off or fracture or structural part damage. Characteristic-level test is an effective and important means to check whether the satellite structure is damaged.

3. Flight model development stage. The verification of structural dynamic characteristics at this stage is mainly based on modal analysis and acceptance test. The main work of modal analysis is to correct the analysis model obtained at the prototype development stage in accordance with the results of mechanical tests (including static and dynamic tests and model tests) and in full consideration of the problems in the tests and the prototype design modification in the flight model design, so as to verify whether the flight model design of the structure meets the requirement of natural frequency. The acceptance test of the satellite's flight model is used to verify whether the structure meets the requirements of dynamic characteristics.

In the phase of flight model development, the verification of structural dynamic characteristics mainly involves analytical verification and acceptance test. The flight model design can be verified by analyzing the following three states:

1. The test state of prototype satellite. The prototype analysis model is corrected, that is, the test data acquired at the prototype development stage is mainly used to correct the structural parameters of the prototype and improve unreasonable structural simplification, so as to obtain a more accurate analysis model reflecting the state of prototype satellite structure in the dynamic test.
2. The state of flight model. According to the prototype design modification in the flight model design, the prototype correction model is modified accordingly to obtain a more accurate analysis model reflecting the state of a flight model satellite.
3. The test state of flight model. The fuel tank is empty, just like in the acceptance test of a flight model satellite. Strictly speaking, the acceptance test under the empty-tank condition cannot effectively verify the main structure of the satellite.

9.4.2 Thermal Design Test Verification

Thermal design verification is generally based on heat balance test. It is mainly to verify two contents: (1) the correctness of thermal design and the conformance of thermal control components; and (2) the accuracy of the spacecraft-level thermal analysis mathematical model. Through thermal design verification, the mathematical model of thermal analysis can be used to make reliable temperature predictions under specific mission conditions (orbit, attitude, payload operation mode etc.).

In order to verify the overall thermal design performance of a spacecraft in space environment and ensure the reliable operation of the spacecraft in orbit, a sufficient number of environmental simulation tests must be carried out on the ground. Among the environmental simulation tests, the most important one is the thermal test conducted in the simulated spatial thermal environment. According to different test purposes, the thermal tests can be divided into two categories: heat balance test and thermal vacuum test. The two tests may be mistaken for each other by beginners because of their similarities in many aspects, and thus deserve enough attention. The similarities and differences between them are shown in Table 9.8.

The thermal vacuum test of the spacecraft is a system-level test. At the prototype development stage, a special thermal control satellite model is developed, and the heat balance test and thermal vacuum test are completed for thermal control satellite. At the flight model stage, the thermal vacuum test is completed for flight satellite. During the test, the spacecraft is placed in a space environment chamber. The temperature of spacecraft components can reach the required test temperature by adjusting the external heat flow and internal thermal power consumption of the spacecraft. The performance tests of the spacecraft in orbit are performed under various operating conditions to verify the ability of the components to function properly under the specified thermal cycling environment stress. The interface matching between the spacecraft subsystems is also tested.

TABLE 9.8 Comparison between Two Thermal Tests

Comparison Item	Heat Balance Test	Thermal Vacuum Test
Test purpose	Verify the correctness of thermal design and assess the capability of thermal control subsystem; obtain the spacecraft temperature data, and correct the mathematical model of thermal analysis	Expose the defects in the design, material and manufacturing process of a satellite, eliminate early faults and evaluate the performance of the entire satellite
Test model	Thermal control satellite (prototype) and first launch satellite (flight model)	Every launch satellite (flight model)
Control parameter	External heat flow value. The heat flow absorbed by the satellite's external surface is controlled to be equivalent to the external heat flow absorbed by the satellite surface in space	Temperature. Control the temperature of on-board equipment to the qualification or acceptance level
Test process	Apply an external heat flow in one operating condition and set the operating mode of the satellite until the satellite reaches thermal stability. Then measure the temperature of each part and turn to the next condition	Adjust the power of infrared heating device or the working condition of the onboard equipment according to the cycle profile, until the equipment temperature reaches the maximum and minimum. Hold for a certain time, and then test its electrical performance (function)

The thermal test items and test process are shown in Figure 9.2.

The heat balance test is carried out under three operating conditions, namely low-temperature operating condition at BOL, high-temperature operating condition at EOL and low-temperature operating condition in safety mode. The thermal vacuum test is composed of four cycles, each lasting for 8 hours at high and low temperatures. The low-pressure discharge and microdischarge of the spacecraft are monitored in the process of vacuum vessel depressurization, but are not monitored in the process of temperature recovery.

The operating conditions of heat balance test and thermal vacuum test are arranged as follows: low-temperature operating condition at BOL – low-temperature operating condition in safety mode – low-temperature operating condition in the first cycle of thermal vacuum test – high-temperature operating condition at EOL – high-temperature operating condition in the first cycle of thermal vacuum test – the second cycle of thermal vacuum test – the third cycle of thermal vacuum test – the fourth cycle of thermal vacuum test. The test profile is shown in Figure 9.3.

After the balancing of low-temperature operating condition at BOL, the test results are summarized to judge the effectiveness of heat balance test results under the low-temperature operating condition, and to determine the low-temperature down-deflection datum of thermal vacuum test. After the balancing of high-temperature operating condition at EOL, the test results are summarized to judge the effectiveness of heat balance test results under the high-temperature operating condition, and to determine the high-temperature down-deflection datum of thermal vacuum test.

The low-temperature operating condition in safety mode is mainly used to judge how long the equipment temperature can be maintained in the emergency satellite attitude and

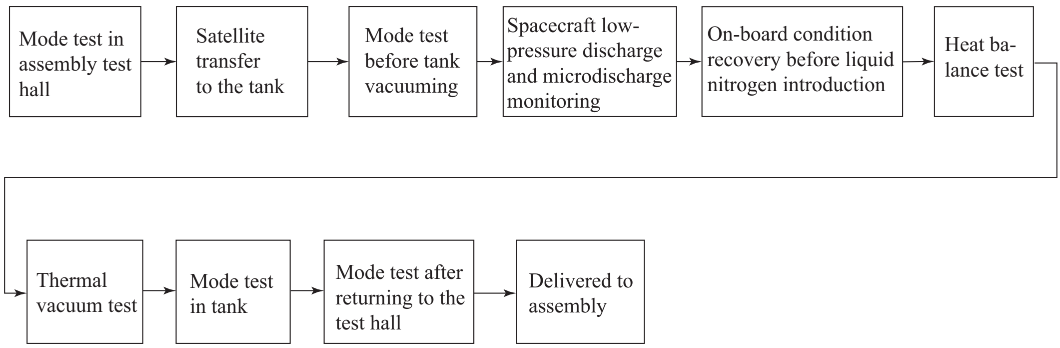

FIGURE 9.2 Thermal test process.

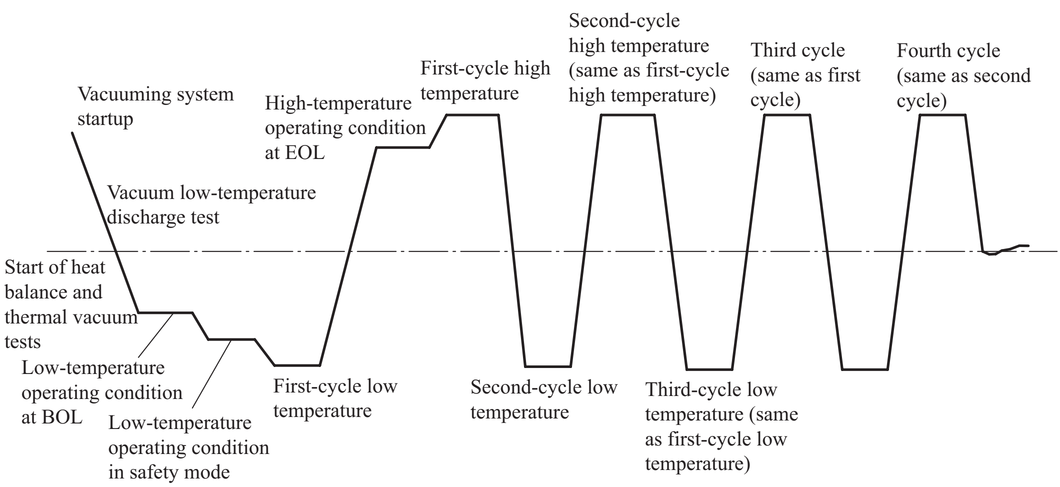

FIGURE 9.3 Profile of heat balance and thermal vacuum tests.

how the equipment temperature will change, so as to provide test data for the in-orbit operation management of the satellite.

The thermal vacuum test begins with the first cycle of low-temperature operating condition and ends up with the fourth cycle of high-temperature operating condition. By controlling the external heat flow and equipment operating mode in this test, the temperature level of low-temperature operating condition at BOL is pulled down by 5°C–10°C and then holds for 8 hours before an electrical performance test, while the temperature level of high-temperature operating condition at EOL is pulled up by 5°C–10°C and then holds for 8 hours before an electrical performance test.

9.4.3 Electrical Performance Test Verification

Electrical performance test verification, also known as electrical performance test, is intended to check whether the functions and electrical performance indexes of a spacecraft system meet the general design requirements, whether the interfaces of various subsystems are matching, whether the transmitted telecommands and telemetry parameters are reliable and accurate, whether the initiating explosive devices are operating safely and reliably, whether the system-level EMC meets the requirements, whether the launch and flight procedures are reasonable, feasible and coordinated and whether the test equipment, test software and test documents are correct.

1. Unit-level test

   The unit-level test is usually the process of testing an independent unit before it is assembled on the spacecraft or after a fault is detected in the unit during the system testing. In order to expose the problems as early as possible, the following sequence of tests is recommended:

   1. Space radiation tolerance test. As the spacecraft is exposed to a large dose of radiation during the operation in space, some components must be specially designed to withstand the cumulative dose over a predetermined lifetime. For some components (such as integrated circuit chips), a single-event upset test must be performed to ensure that the components are free of latch-up and tolerant of soft errors.

   2. Initial magnetic field test. Initial magnetic field test is required for a spacecraft with magnetism-sensitive measuring unit to ensure that the parasitic magnetic fields do not interfere with the correct measurement of orbital magnetic field.

   3. Leakage test. The sealed components and subsystems must be tested to ensure good sealing performance during the tests of vibration, noise and thermal vacuum, during blast-off and in the space vacuum.

   4. Electrical performance baseline. After the initial electrical performance testing for hardware, software and test programs, a complete set of electrical performance tests must be performed at normal ambient temperature, pressure and humidity to test the DUT at the expected highest, lowest and normal input voltages of spacecraft power subsystem. The data from these tests will serve as a reference baseline for the electrical performance of the subsystem before, during and after the thermal vacuum test, vibration test and noise environment test.

   5. EMC. The EMC testing of subsystems or components must also be completed as early as possible to identify the incompatibilities with other components and resolve the problems prior to system-level integration.

   6. Temperature characteristics. The components and subsystems must go through several heating-cooling cycles in a thermal environment (non-vacuum) to enable the electrical connections between dissimilar materials to work under different thermal expansions, and to measure electrical performance under the different combinations of temperature and supply voltage. For a component with high heat capacity, enough time must be available to ensure that its temperature can be

stabilized to an extreme value and that its performance can be measured correctly in the test. The electrical performance baseline tests shall be completed when the hot and cold temperatures are stable. The test results are then compared with those obtained at normal temperature. The subsystems or components that may experience rapid temperature changes shall go through thermal shock tests.

7. Vibration. Each subsystem or component shall be powered and then go through triaxial sinusoidal and random vibration tests at the levels defined by the project. In order to expose potential problems, the powered equipment shall be monitored during the actual vibration. If some equipment does not need to be powered during launch, powering it up for vibration testing will be undoubtedly a risk and thus must be dealt with very cautiously.

8. Mechanical impact and noise. Mechanical and noise testing shall be done according to the different conditions of subsystems and components and the engineering requirements. Noise tests are required for the devices with large area and small mass or with film windows. Impact tests are required for those components which are prone to impact events or whose installation positions will be affected by other impact events.

9. Deployment. In order to verify whether the design of some devices has a margin before and after the application of mechanical stress and whether the “hooking” will occur, the deployment test must be done for those devices. The devices requiring a deployment test are the sensor cantilever, the antenna, the solar array, the instrument cover and the separation mechanism between the spacecraft and the launch vehicle.

10. Thermal vacuum. A thermal vacuum test is used to verify the ability of a component or subsystem to operate in a space vacuum environment and to verify the correctness of thermal design of the components or subsystems with large heat consumption. If a device is supplied with high-voltage power during blast-off, it must be powered and monitored in the vacuuming process to ensure no corona discharges (i.e., no arcing). If a device requires a high-voltage power supply but does not work during launch, it must be vacuumed for enough time in the test before being powered.

2. Subsystem-level tests

The subsystem-level tests can be conducted at either the subsystem level or system level. The main test items are as follows:

1. Check the power supply interfaces. The power supply interfaces must be checked before connecting any component with power supply. The power interface tests are divided into power-off test and power-on test. The power-off test is used to check whether all power cables are isolated from the return wires and ground wires. The power-on test is used to check the correctness of the voltages on electrical

connector contacts in the cable network and the presence of voltages on other contacts in order to ensure no short circuit that could cause the spacecraft failure.

2. Functional test. Check whether all the power supply paths and data paths into and out of the DUT are correct. The test shall examine multiple telecommands, telemetry data and multiple functions provided by the subsystem, without the need of external excitation or ground test equipment. If possible, this test shall check some redundant interfaces.

3. Performance test. A performance test is a test that examines the subsystem under test in detail, including the examination of as many of the subsystem specification indexes as possible. If necessary, an external excitation source or a ground support equipment is required to fully test the subsystem. Redundant interfaces and internal circuits as well as all operating modes must be tested.

3. System-level test

System-level test is the electrical function testing completed at each stage of the system-level test process. It mainly includes the functional tests performed during the system operation, sometimes dealing with a limited number of subsystem performance tests. The system-level test items include:

1. Electrical performance baseline. This is a test activity that provides an electrical performance reference point (or baseline) for all the other spacecraft tests to judge whether the performance is correct. This test is performed under normal ambient temperature, pressure and humidity.

2. Interface joint test of ground station. This test is used to check the correctness of the interface between the ground control station and the spacecraft and to test the RF link compatibility between them.

3. Fly simulation test (or "fly simulation" for short). This is a test that provides validation and drills for the flight operator to control the spacecraft. It verifies through tests whether the configuration operates correctly before and after the launch according to the desired procedures and familiarizes the operators with the desired system performance.

4. EMC. This is a spacecraft-level test to verify whether the compatibility between subsystems is satisfactory. In some cases, it may not be necessary to fully follow the EMC test standards, but rely solely on self-compatibility testing, to check if the spacecraft will have its own electrical interference.

5. Remanence measurement. The spacecraft remanence must be measured so that it will not affect the magnetic sensor on the spacecraft. For a spacecraft with magnetic torque control, the magnetic field and magnetic moment characteristics of its subsystems must be measured in order to calibrate and eliminate the influence of remanence on the magnetic sensor in orbit.

6. Dynamic test, including the examination of vibration, impact and noise. An appropriate stress level and duration shall be applied to the spacecraft in order to validate the force coupling analysis and simulate the launch environment so that all subsystems are in the as-launched state. The spacecraft shall undergo the sinusoidal and random vibration tests. For large spacecrafts, the acoustic (noise) test shall be carried out. These tests are designed to verify the ability of a spacecraft and its subsystems to withstand the launch environment. The impact test is to verify the ability of a spacecraft to withstand the impact forces brought by the ignition and flameout of rocket engine, the fairing jettison and the separation between the rocket and the spacecraft.

7. Deployment test for pyrotechnic device and mechanism. This test is required to verify whether the equipment can function properly after undergoing the vibration, noise and impact in the simulated launch environment. In some deployment tests (such as the tests of solar arrays and antennas), gravity compensation measures are needed to create a zero-gravity field.

8. Thermal vacuum test. The thermal vacuum test consists of two parts: heat balance test and heat cycle test. In the heat balance test, the heat balance of spacecraft system is confirmed through the operation of the equipment itself. In the thermal vacuum test, the operation condition of the equipment within the temperature limit of thermal environment (wider than the orbital temperature range) is checked.

9. System-level aging test. This test is a continuous and uninterrupted simulation of in-orbit operation over 100 hours to eliminate early failure and improve the system reliability.

9.4.4 EMC Test Verification

9.4.4.1 Equipment EMC Test Verification

The spacecraft equipment should achieve self-compatibility in all working modes during the development and flight phases. The electromagnetic energy generated by a device should be controlled within a certain range, so that the device can work compatibly with the related devices. At the same time, its performance will not be affected by the electromagnetic environment during the final assembly, test, pre-launch preparation and orbiting.

The equipment EMC test items typically include:

1. Conducted emission of WCE101 power wire (25 Hz–10 kHz)
2. Conducted emission of WCE102 power wire (10 kHz–10 MHz)
3. Conducted emission of WCE106 antenna terminal (10 kHz–18/40 GHz)
4. Radiated emission of WRE101 magnetic field (30 Hz–50 kHz)
5. Radiated emission of WRE102 electric field (10 kHz–18 GHz)

6. Harmonic and spurious output radiation emission of WRE103 antenna (10 kHz–18/40 GHz)
7. Conducted susceptibility of WCS101 power cable (30 Hz–150 kHz)
8. Conducted susceptibility of injected WCS114 cable bundle (10 kHz–200 MHz)
9. Conducted susceptibility of injected WCS115 cable bundle
10. Damping sinusoidal transient conducted susceptibility of WCS116 cables and power wires (10 kHz–100 MHz)
11. Radiosensitivity of WRS101 magnetic field (30 Hz–50 kHz)
12. Radiosensitivity of WRS103 electric field (10 kHz–18/40 GHz)
13. ESD sensitivity
14. Start-up transient current of DC power supply
15. Voltage surge sensitivity of power wire
16. Electromagnetic leakage assessment of microwave passive components

9.4.4.2 System-level EMC Test Verification

The EMC test of spacecraft system is mainly to obtain the EMC interface characteristics of the whole spacecraft to ensure that the radiated emission and radiosensitivity during the spacecraft launch are compatible with the electromagnetic environments of launch vehicle and launch site, and to verify that the spacecraft can operate with self-compatibility in all kinds of normal operation modes from the separation from rocket to the entry into orbit.

The spacecraft shall undergo the system-level EMC verification tests at the prototyping stage and flight model stage. The optional system-level EMC test items are as follows:

1. Lap joint and grounding resistance measurement
2. Conducted transient interference test of system power bus
3. Conducted voltage ripple test of system power bus
4. Sweep-frequency conducted emission test of system power bus
5. RF leakage test
6. Antenna coupling test
7. In-band noise test of the receiving antenna
8. System self-compatibility test
9. Test of the radiation compatibility between spacecraft system and launch vehicle (and launch site)

9.4.5 Magnetic Test Verification

The spacecraft-level magnetic test is aimed to measure the magnitude and direction of remnant magnetic moment of a spacecraft under different operating conditions and of the induced magnetic moment in geomagnetic field, to determine the installation position of the compensation magnet and the magnitude and direction of the compensated magnetic moment and to control, through magnetic compensation, the remnant magnetic moment of the spacecraft (excluding solar arrays) under all operating conditions to be less than the design value.

The methods of magnetic moment measurement can be divided into two types: direct measurement methods and indirect measurement methods. The direct measurement methods can be further divided into the measurement in the Earth's magnetic field and the measurement in a constant magnetic field.

1. Measurement in the Earth's magnetic field

   At first, a satellite is suspended or placed on a platform floating in a liquid. The interaction between the horizontal component of the Earth's magnetic field and the satellite's magnetic moment component in the horizontal plane will produce a torque on the suspension system. If the magnetic field and moment (which are usually small) in the local area can be measured accurately, the horizontal component of the satellite's magnetic moment can be determined. Then the satellite is relocated until its axis originally perpendicular to the horizontal plane is located in the horizontal plane. At last, repeated measurement is made to determine the direction and magnitude of the satellite's magnetic moment.

   The accuracy of this measurement method is limited by the following factors: the measurement accuracy of the mean value of the horizontal component of the Earth's magnetic field and the measurement accuracy of small moments.

2. Measurement in a constant magnetic field

   In the measurement process, the whole device is placed in the action area of a coil system and the magnitude and direction of magnetic field in this area are precisely controlled to exclude the influence of geomagnetic field measurement accuracy and perturbation on the measurement results. By observing the deflection of the suspension part of the device in the magnetic field, the resulting moment can be measured directly. The component of the satellite's magnetic moment in one coordinate axis can be obtained by calculation, and then the components of the satellite's magnetic moment in the other two coordinate axes can be measured by changing the direction of magnetic field.

   The main factors influencing the measurement are the airflow near the measuring device, and the flow and surface tension of suspended liquid.

3. Indirect measurement method

   The satellite is placed in a zero-intensity magnetic field or a constant-intensity magnetic field. The magnetic field intensity of the satellite is measured at a certain distance, and then the magnetic moment of the satellite is calculated.

After obtaining the magnetic moment measurement of the satellite not powered, magnetic compensation shall be made for the system according to the spacecraft requirements. The size, direction and installation position of the compensation magnet are determined, and the magnet is attached to the outer wall of the cabin above the docking surface. Then the magnetic moment of the satellite not powered is directly measured. If the measured value is unsatisfactory, the magnet shall be adjusted. Then the magnetic moment is remeasured. This process will be repeated until a satisfactory magnetic moment measurement is obtained.

CHAPTER 10

Digital Design and Development of Spacecraft System

Fan Haitao, Liu Xia, Xie Zheng, Yuan Yi, Zhang Liang, Zhang Liqiang, Zhao Han, and Wang Jiachun

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CONTENTS

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The digital development of a spacecraft is to solidify the work experience, knowledge and process through the wide application of digital technology in the spacecraft engineering development, to standardize the development process and improve the working efficiency and quality by use of digital tools, and to improve the system engineering and overall design capability through the establishment of "digital spacecraft". The range of digital spacecraft development is very broad, covering the whole lifecycle of spacecraft product development, which includes digital product design/digital performance analysis and simulation, digital manufacturing, digital testing and digital operation/maintenance and in-orbit management. The horizontal disciplines involved in digital spacecraft development include computer-aided design (CAD), computer-aided engineering, computer-aided process planning (CAPP), computer-aided manufacturing (CAM), computer-aided testing and product data management (PDM). This chapter will highlight the digital design of spacecraft systems.

10.1 DIGITAL DESIGN TECHNIQUES

Digital design is to apply digital means to the product design process, that is, to make full use of computer, data storage, network transmission and other digital techniques to establish a set of processes and environments oriented to product design process and realize the deep integration of digitalization and product development process. After years of continuous digital development, many new design approaches and management modes have been derived from digital techniques.

10.1.1 Digital Mock-Up (DMU) Technique

Before the advent of computers, the evaluation and testing of almost all complex products relied on physical mock-ups (PMUs). The so-called PMU refers to a physical entity model, which is full-scale or scaled up/down compared with real product. It is generally made of paper, wood, metal or actual production material and is used to verify the product design. However, its production process is time-consuming and labor-intensive, and its subsequent maintenance is very expensive. Moreover, the PMU can't accurately describe the performance of a product with rich information.

To solve this problem, the DMU technique emerged gradually in the 1990s.[1,2] Compared with PMU, DMU is a digital model established according to product characteristic information by using the computer technology. It describes the simulation, testing and evaluation of product structure, functions, performance and other characteristic information. Its advantages are very obvious. By using the constructed DMU, the designers can find potential design problems and design changes in time before the PMU completion, reduce the cycles of design changes after physical verification, so as to shorten the development cycle, reduce the costs, improve the design quality and facilitate the DMU reuse. Because of the advantages mentioned above, the DMU technique has been developed rapidly in a short time. In recent years, with the further development of computer technology, the DMU function has been gradually changing from three-dimensional (3D) display to the simulation and verification of product functions and performance.

10.1.2 Model-based Definition (MBD) Technique

With the development of CAD technology, the product definition technique has developed from engineering drawing technique to 2D CAD, and then to 3D modeling technique. With the characteristic of "WYSIWYG", the 3D model can intuitively present the real 3D solid structure of a product and fundamentally change the engineering design method of the product. However, a long time of 3D development history proved it difficult to produce and test the products directly by relying only on 3D models. The 3D models contain the detailed geometric information not available in the 2D drawings but lack dimensional tolerances, surface roughness, surface treatment methods, heat treatment methods, specifications and standards. Therefore, the 3D models and 2D drawings need to be released at the same time, and the 2D drawings are still the main basis of the product manufacturing process. This phenomenon hinders the data transfer in the process of product design and manufacturing. In addition, there may be potential conflicts between the 3D models and the 2D drawings. In particular, when the geometric features of a model are changed, the control of product data version will become very difficult.

To further improve the product quality and shorten the production cycle, the MBD technique has emerged at the right moment.[3] In this technique, all the process description, properties, management and other product information are attached to a 3D model so that only one digital model is needed to obtain all required information on the product. In this way, the MBD has reduced the over-reliance on other information systems, broken the barrier between design and manufacturing, and effectively solved the problem of design-manufacturing integration. The MBD cannot be simply understood as mapping the tolerance and process information contained in the 2D engineering drawings to a 3D model. Instead, it makes use of the powerful presentation effect of 3D models to explore a more efficient way of design information presentation that is easy for users to understand and for machines to identify. The MBD has solved the problem of describing the product dimensions, tolerances and process in a 3D model and has used 3D models, instead of traditional 2D drawings, as the only basis for the manufacturing process. At the same time, the manufacturing department can directly use this digital model for the design and simulation of production process and tooling, thus greatly shortening the product development cycle.

10.1.3 Multidisciplinary Design Optimization (MDO) Technique

The development of large complex aerospace products is a complicated system engineering involving machine, electricity, heat, control and other disciplines. For a single discipline, the corresponding mathematical model can be established for calculation analysis and optimization design. However, a unified design and analysis model can't be easily established for a large complex engineering system where various disciplines will intersect and influence each other and the design optimization of a single discipline cannot replace the optimization of all the disciplines.

To solve this problem, the American Institute of Aeronautics and Astronautics proposed an MDO technique in the 1990s.[4] It is a methodology to design complex systems and subsystems by exploring and utilizing the interactive synergistic mechanism of the systems.

It mainly deals with the coupling and trade-off problems in the engineering design of large complex systems. By defining a global function, the constraints of all the disciplines and systems are simultaneously satisfied through the interaction and coupling of variables, constraints and performance indexes. The high-precision analysis models and optimization techniques of various disciplines are organically integrated to find the best system concept and obtain the overall optimal solution.

10.1.4 Product Lifecycle Management (PLM) Technique

In the 1960s and 1970s, with the increasing application of CAD and CAM techniques, a large number of information systems were constructed by manufacturing companies. However, due to the lack of overall information planning, each information system was separate from the others. Because of the lack of effective information communication and coordination, those systems became "information islands". With the continuous development of computer technology, different types of data in the information systems were rapidly expanding, so they had weak points such as difficult search, low information sharing, poor security, uncontrollable version and data inconsistency, which caused great pressure to the data management of companies.

To solve these problems, the concept of PLM was formally proposed.[5] However, due to its rapid development, a unified understanding and a clear definition of PLM technique have not been developed. CIMdata, the world's leading PLM strategy consulting and research organization, defined the PLM as: "PLM is a tool to help engineers and other personnel manage the product data and product development process. The PLM system ensures the tracking of a large amount of data and information required by design and manufacturing, and thus supports and maintains the products".

The PLM has its broad sense and narrow sense. The PLM in the broad sense can manage the data or information on the whole lifecycle of a product from market demand, research and development (R&D), product design and manufacturing to sales, service and maintenance, while keeping the product data consistent, shared and safe throughout the lifecycle. In contrast, the PLM in the narrow sense only manages the product data or information related to engineering design. In recent years, the PLM technology has made great progress and become an important means of data management for most of the enterprises.

10.2 DIGITAL SPACECRAFT DEVELOPMENT

Digital spacecraft development is to make full use of digital technology to build and use digital models for information transfer, collaborative design, spacecraft product development and integrated verification in the spacecraft's overall design and analysis, manufacturing, assembly and integration, comprehensive test, experimental verification and other processes.

In this section, the digital spacecraft development and its characteristics are discussed from the perspective of digital technology application, the key digital contents involved in the process of spacecraft system design are presented, and the considerations of manufacturability, assemblability and testability in digital system design are proposed in accordance with the characteristics of spacecraft system design, manufacture, assembly, measurement and testing.

10.2.1 Characteristics of Digital Spacecraft Development

Different from traditional spacecraft development, digital spacecraft development is to achieve the consistency of design information between different development stages and work items through a unified information expression model, optimize the input and output of each stage in the development process, minimize the information transmission steps, and realize the parallel cooperation in the development process; to gradually replace traditional input/output documents and drawings with digital models and establish a unified data source; to tightly link the design, analysis and simulation verification through simulation analysis, and further advance the verification work. At present, the quantitative design, simulation analysis and verification based on digital models have become an effective supplement to traditional development methods such as the physical verification based on physical objects.

Compared with traditional development mode, the digital development mode has the following characteristics in terms of environment, team, process and output:

10.2.1.1 Digital Development Based on Collaborative Environment

At the conceptual design stage, the design work is carried out in a collaborative environment. The design requirements and intentions can be effectively reported and released to ensure that the design elements are not missed. The multi-disciplinary designers will work in parallel, timely transfer, feedback and correct the design results, and analyze and verify the system's functional and performance indexes in real time to ensure more sufficient multi-concept comparison.

At the detailed design stage, the detailed design of each discipline is carried out in a collaborative design system. The interfaces between disciplines and the parameters indexes have been defined. To ensure the integration of design results, the type, version and design of engineering software shall follow the strictly required references and constraints. The design work is done on a unified collaborative design platform to facilitate the sharing of design results. The design iteration is accelerated by using the discipline-oriented collaborative design tools, and the design output is provided according to the discipline-oriented model system specifications.

10.2.1.2 Personnel Organization in the Form of Integrated Product Team (IPT)

At the conceptual design stage, a special conceptual demonstration team is established to rapidly carry out early conceptual design and optimization and effectively support the model project approval. The team members include the personnel engaged in model system, subsystems, scheduling, quality and technics.

At the detailed design stage, temporary interdisciplinary design teams based on IPT can be set up depending on the degree of collaboration closeness. For example, to strengthen the information communication and improve the development efficiency, an Interface Data Sheet (IDS) signing team is formed by the designers of the spacecraft system and its subsystems; an interdisciplinary 3D collaborative design team is formed by the designers of system, structure and thermal control; a collaborative design team is formed by the designers and technologists; and a review team is formed by senior professional designers.

10.2.1.3 Digital Development Process with Model as the Core

With the wide application of digital methods and means (such as software tools and simulation analysis) in the process of spacecraft development, the 3D models have gradually replaced documents and drawings as the design input and output and information carriers. Using digital simulation to replace the physical test has become one of the means to verify the correctness of the system design. The parallel collaboration of system engineering, disciplines, manufacturing, final assembly and testing based on the unified data source has been gradually promoted. The development processes such as spacecraft design, manufacturing, final assembly and verification have undergone profound changes.

10.2.1.4 Changes of Output

The main outputs of digital spacecraft development are all kinds of digital models, including system model, geometric model, engineering verification model, tabulated data and database. Drawings and documents are the results of model derivation. For example, the main outputs of conceptual design are system demand model, functional model, architecture model, simulation verification model, index assignment table and interface data. The main outputs of detailed design are the 3D models oriented to production and assembly and the engineering verification models oriented to testing. The user needs and the technical requirements at all levels are the outputs of the demand model. The manufacturing and final assembly is based on the 3D models and the manufacturing requirements attached to the 3D models.

10.2.2 Collaborative Design of Spacecraft System Engineering/Structural/Thermal Control

Mechanical system design, structural design and thermal control design are important parts of the overall spacecraft design. They have very strong systematicness and coupling performance. The design change of any one of them often leads to the design iteration of the other two.

10.2.2.1 Traditional Design Process

In the traditional spacecraft design process (as shown in Figure 10.1), mechanical system design, structural design and thermal control design are basically a serial design process. They rely mainly on design documents and drawings for the mutual transfer of design requirements and interface parameters. A variety of design information with low reuse rate, complex information transfer and understanding, as well as untimely design status change and transfer often lead to a long iteration cycle, complex process and design mismatch (such as mutual interference), which adversely affect the development progress and design quality of the spacecraft.

10.2.2.2 Digital Design Process

In the digital spacecraft development process (as shown in Figure 10.2), the "top-down" design pattern is followed. The mechanical system, structure and thermal control share a unified framework model. An overall spacecraft 3D model is built on the spacecraft

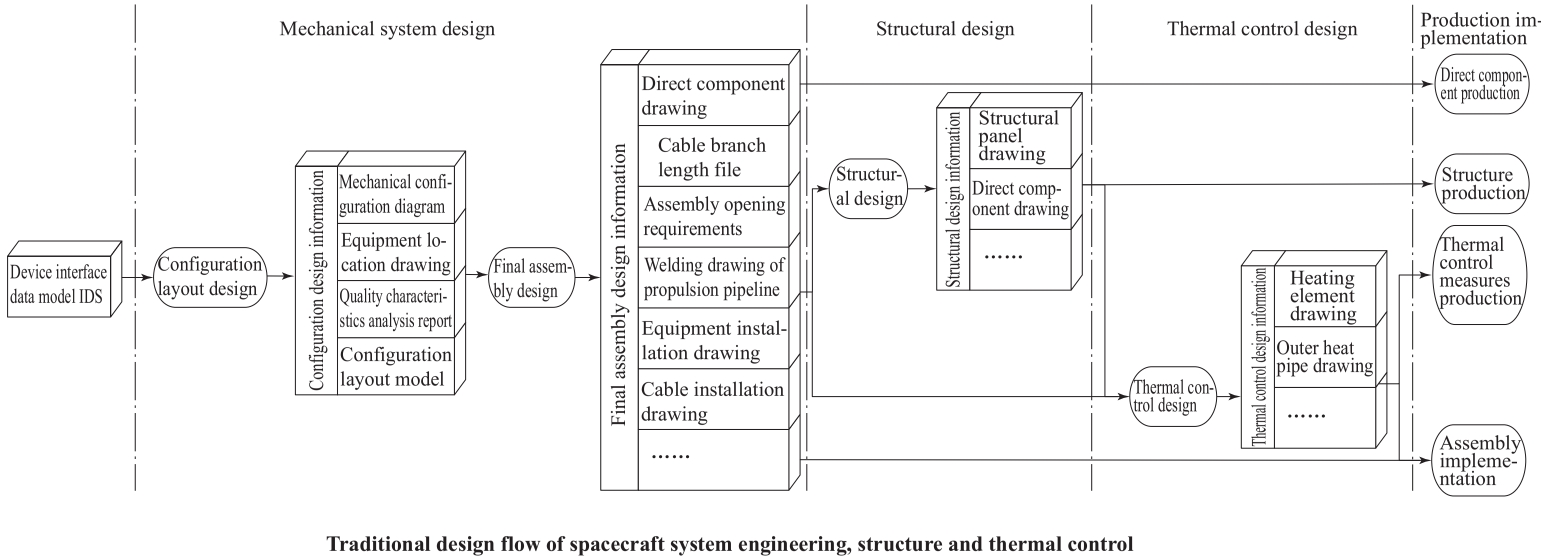

FIGURE 10.1 Traditional design flow of spacecraft system engineering, structure and thermal control.

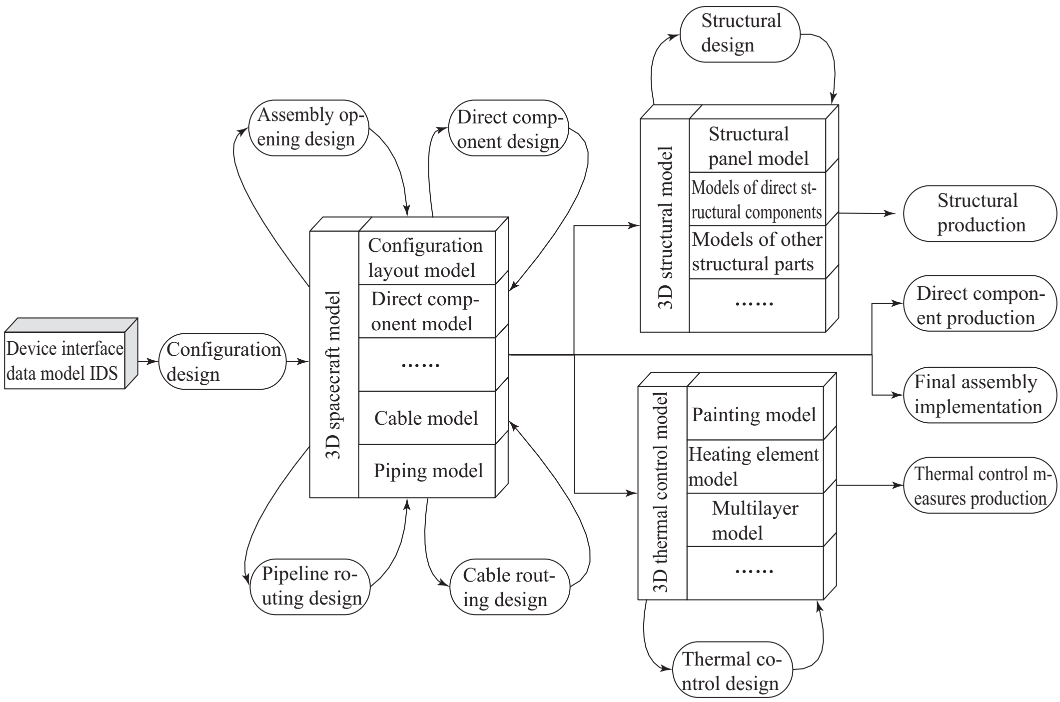

FIGURE 10.2 Digital technique process of spacecraft system engineering, structure and thermal control.

configuration layout design. Meanwhile, device interface data is used as the data source of collaborative mechanical, electrical and thermal design of the system and its equipment to carry out detailed parallel 3D design of final assembly, structure and thermal control:

1. The final assembly design (including equipment installation, cable routing, pipeline routing and direct-component assembly) is carried out based on the 3D spacecraft model, and the design results are directly reflected into the 3D model as the data source for downstream design and production.

2. The design information (such as structural layout, key structural size and final assembly opening) needed for detailed structural design is extracted from 3D models to form the 3D model of the spacecraft structure. The detailed structural design is carried out based on the 3D structural model, which is further improved at the same time. Finally, all the detailed design information is reflected in the 3D model of the structure to serve as the data source for downstream process design and production.

3. All the information needed for detailed thermal control design is extracted from 3D models to form the 3D thermal control model of the spacecraft. The detailed thermal control design is carried out based on the 3D thermal control model, which is further improved at the same time. Finally, all the thermal design information is reflected in the 3D thermal control model to serve as the data source for downstream process design and production.

10.2.3 Spacecraft Design-Process Collaboration Mode

In the process of spacecraft production, the design information shall be transferred to the production and processing department to guide the product processing and production. In the spacecraft assembly, integration & test (AIT) phase, the assembly implementation department needs to carry out final assembly, measurement and testing for the spacecraft according to the overall design information.

In the traditional development process (as shown in Figure 10.3), the design of final assembly, structure and thermal control is in series with the downstream production, that is, only after all the design work is completed and the design state is controlled, the downstream process design and production can be started. This results in an overlong cycle of design and production & processing. In addition, if a process problem exists in the original design, it needs to be fed back to the designer for design change. As a result, the actual operation chain becomes more complex. All the design information is recorded and controlled in the form of documents and 2D drawings. The production plants need to fully understand and digest the contents of documents and drawings, which puts forward high requirements for the technological and processing personnel.

The same problem also exists between the final assembly design and the downstream assembly implementation, which are also in a serial process. All the system design information on equipment installation, pipeline welding and cable installation is delivered in the form of documents and drawings to the assembly implementation department. This will not only need a lot of manpower and material resources but also lead to the loss of part of the design information. Moreover, the spacecraft system designers are often required to follow the whole process of assembly implementation.

The collaboration mode pursued after the adoption of digital technology is shown in Figure 10.4. All the design information on mechanical system, structure and thermal

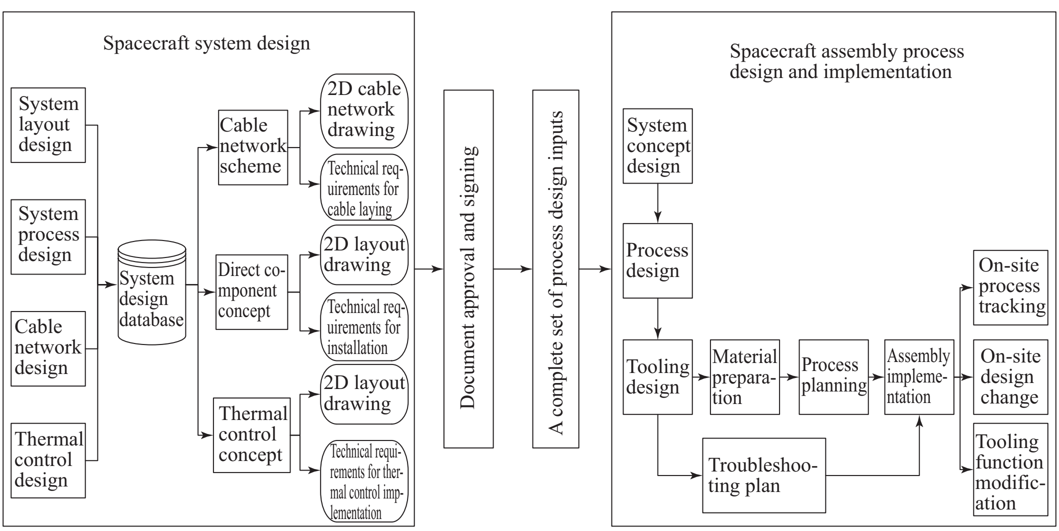

FIGURE 10.3 Traditional “design—final assembly” serial mode of a spacecraft.

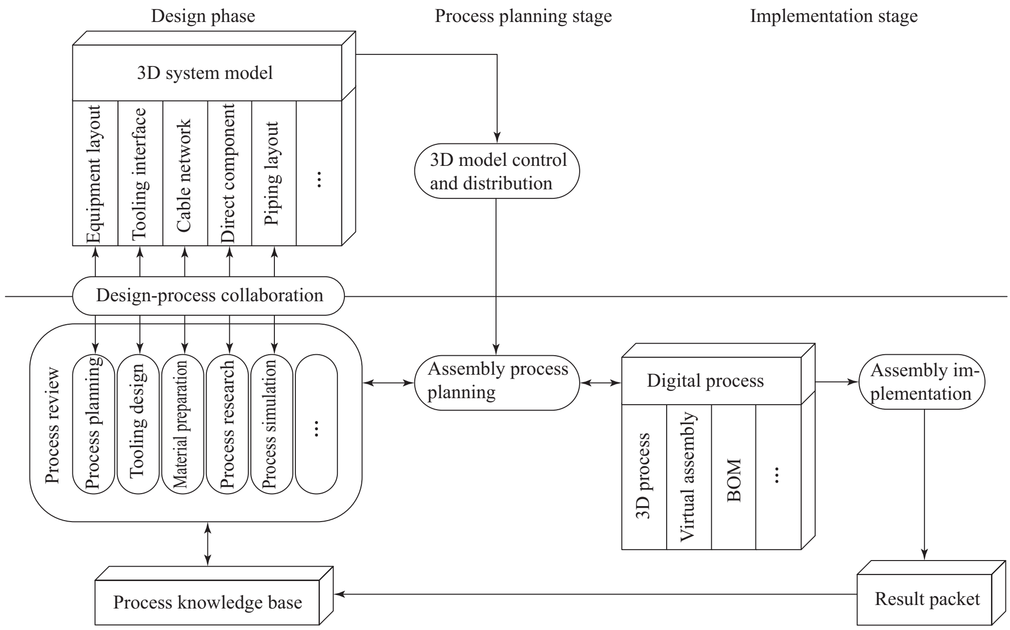

FIGURE 10.4 "Design—final assembly" collaborative design mode of a spacecraft.

control will be reflected in the 3D models, from which the downstream plants can directly obtain the required processing information without the need for converting or "translating" the design information into 2D drawings and documents. This ensures the same source for production information and design information and improves the design quality and working efficiency. In addition, the production technology personnel can intervene in the early design phase and timely recheck the process contents in the design, so the design iterations can be minimized. Meanwhile, the mechanical and thermal analysis models can be derived from the 3D models to guide the simulation tests on mechanics, thermal control, remanent magnetism and electromagnetic compatibility (EMC).

In the case of system assembly design, the sharing of 3D model can be realized through the data management system. Based on the assembly design model, the process design and material preparation can be carried out in advance, and the process design process, tooling design and key assembly process can be simulated and verified. On the one hand, the implementation plan of final spacecraft assembly shall be verified, and the technological problems in the system design shall be found and timely fed back to prevent the design defects from flowing into the final assembly site. On the other hand, the process design and planning shall be advanced, and the process design shall parallel the system design, so as to shorten the time of final assembly development.

10.3 MODEL-BASED 3D COLLABORATIVE DESIGN OF A SPACECRAFT

With the wide application of digital techniques (especially the popularization of 3D CAD technique and the maturity of CAD/CAM integration technique), the product development based on 3D models has become a trend. The 3D model has been gradually accepted by people because it is realistic and intuitive and can be directly used in numerical control processing. It challenges the traditional development model based on 2D drawings and has a profound impact on the development concept, process and tools of modern manufacturing industry.

Through the digestion and absorption of Chinese and foreign advanced concepts and the consideration of China's experience in 3D collaborative spacecraft design and development, this section presents the general requirements for 3D model classification and construction, the 3D collaborative design of spacecraft system, structure and thermal control and the design model construction oriented to manufacturing and final assembly.

10.3.1 General Requirements for 3D Model Classification and Construction

The 3D spacecraft models can be classified in different dimensions. According to the characteristics of spacecraft product development, they are usually classified by development stage and purpose:

10.3.1.1 Classification by Development Stage

According to the different development stages of a spacecraft, 3D spacecraft design models can be divided into the 3D model for conceptual design and the 3D model for detailed design. The 3D model for conceptual design is established in the phase of spacecraft concept design and is used to complete the technical research and conceptual demonstration and to determine the indexes, interfaces and composition of the spacecraft system and subsystems. The 3D model for detailed design is established in the phase of detailed design and is used to complete the subsystem and equipment design, system-level layout and optimization and to define the basis of physical production. The conceptual design model and the detailed design model constitute a process of information enrichment and have no boundary in between in the strict sense.

10.3.1.2 Classification by Purpose

According to the different purposes of spacecraft design and verification, 3D models can be divided into 3D system design model, 3D structural design model and 3D thermal control design model. The 3D system design model reflects the design intention for system configuration and equipment layout and shows the spacecraft equipment layout, large components installation, cable routing and pipeline connection. The 3D structural design model reflects the design intention for the structure and mechanism and shows the spacecraft structure configuration and partial-assembly connection method. The 3D thermal control design model reflects the design intention for thermal control and shows the type, specification and layout of thermal control measures.

The construction of 3D spacecraft models shall generally meet the following design requirements:

1. The dimensions reflecting the physical shape shall be modeled in a 1:1 scale, and the attributes of the physical product shall be expressed in the models.

2. With the deepening of the design work, the model contents are gradually enriched and refined from top to bottom, namely, from system level, cabin level, cabin panel level to equipment level.
3. Through the structured combination, the lower level models can turn into higher level models. The establishment of the models at each level shall follow the unified construction specifications and requirements.
4. The 3D design models of the spacecraft shall be simplified according to different stages and purposes while meeting the accuracy requirement and other requirements.

10.3.2 3D Collaborative Design of Spacecraft System, Structure and Thermal Control

The 3D collaborative design of spacecraft system, structure and thermal control uses spacecraft (satellite) framework model as the top level of system-level collaborative design and uses model and structured data as the carrier of collaborative information transfer to build a 3D design model oriented to spacecraft system, structure and thermal control.

10.3.2.1 Construction of Spacecraft (Satellite) Framework

The spacecraft (satellite) framework (as shown in Figure 10.5) is used to express the common information shared by the 3D models of spacecraft system, structure and thermal control, including the information on spacecraft (satellite) datum, cabin datum, basic size and lapping relationship. When the design is changed, the information can be transferred quickly to ensure the real-time state consistency of top-level design parameters of spacecraft system, structure and thermal control.

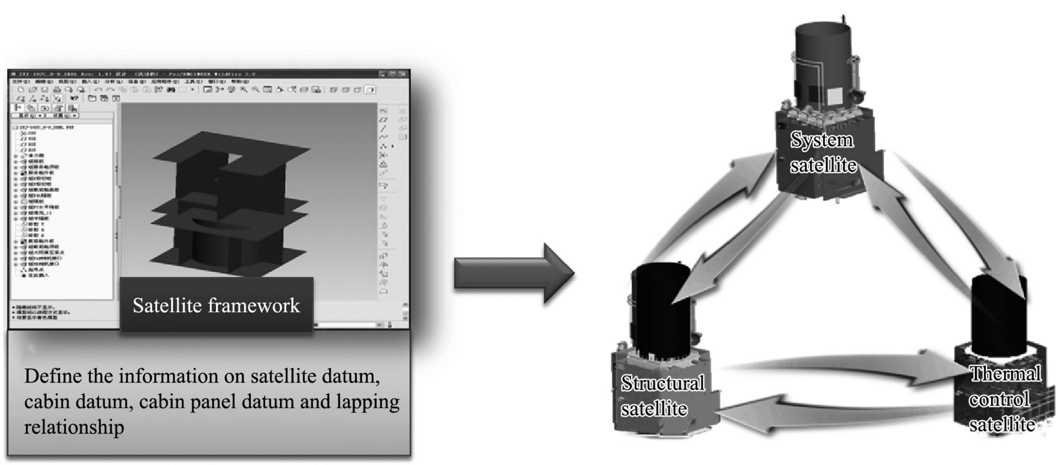

FIGURE 10.5 Satellite framework construction.

10.3.2.2 Construction of 3D System Design Model

The establishment of 3D system design model shall meet the following requirements:

1. Accurately reflect the size of the mechanical installation interface between the spacecraft and the carrier rocket.
2. Contain the 3D models of spacecraft cabins, cabin panels, equipment, cable network, pipelines and grounding connections that can fully express the spacecraft composition.
3. Accurately reflect the geometric shape, layout, position and direction of the cabins, cabin panels, equipment, pipelines and cable network.
4. Adopt a structure with several levels according to the requirements of model assembly.
5. Provide necessary process information.

The 3D system design model can be constructed according to the following steps (as shown in Figure 10.6):

1. Carry out the lightweight equipment modeling based on interface data.
2. Arrange the equipment configuration layout on cabin panels by using the spacecraft (satellite) framework as datum. Complete the configuration layout of each cabin panel to obtain the spacecraft (satellite) configuration layout model.
3. Analyze the mass characteristics, field of view (FOV) occlusion, solar wing occlusion and mass surface density based on the configuration layout design model.
4. Adjust the equipment layout according to the analysis results and optimize the configuration layout of the whole spacecraft (satellite). The above third and fourth steps

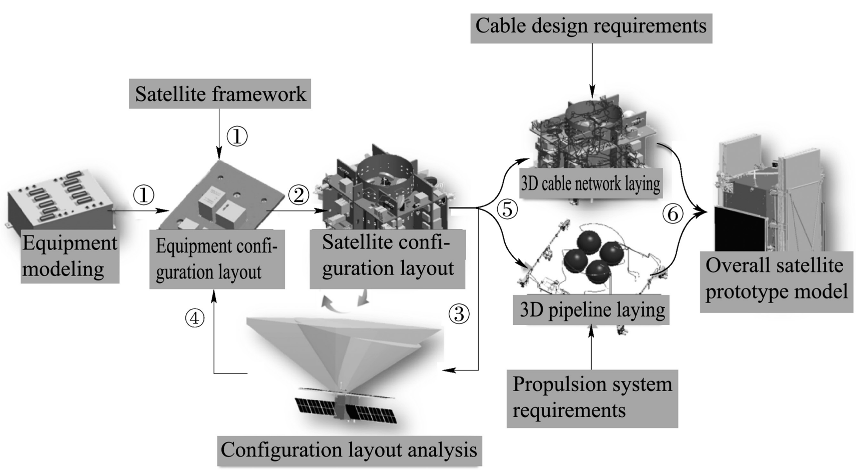

FIGURE 10.6 3D modeling of overall spacecraft system design.

constitute a process of cyclic iteration. After several rounds of iteration and adjustment, the configuration layout scheme of the whole spacecraft is finally shaped.

5. After the completion of equipment layout, extract the location information of equipment installation holes and the information on cabin panel openings and transmit it to the structural designers in the form of information flow.

6. Use the propulsion system requirements provided by control propulsion system as input to carry out the 3D design of pipeline system based on the configuration layout results of the whole spacecraft (satellite); and use the cable design requirements and cable contact table provided by power supply and distribution subsystem as input to carry out the 3D laying design of cable network.

7. After completing the above design work, the antennas, solar wings and other large components are installed to obtain the 3D system model design.

10.3.2.3 3D Modeling of a Spacecraft Structure

The 3D modeling of a spacecraft structure shall meet the following requirements:

1. Reflect the real shape and actual composition of the spacecraft structure and express the connection relationship between components and their connection method.
2. Contain the properties and parameters necessary for mechanical analysis.
3. Contain the collaborative data interfaces with system design and thermal control design.
4. Contain necessary process information.

The 3D design model of the spacecraft structure can be constructed according to the following steps (as shown in Figure 10.7):

1. The preliminary configuration of 3D structural design model is defined based on the spacecraft (satellite) framework. This configuration is the 3D substantialization of spacecraft framework, in which the cabin panels have no detail features such as holes, openings and embedded parts. However, the relative position and lap joint between the panels can be seen from the model.

2. The structural designer extracts the outer contour of panels through "Publish Geometry" to define special features.

3. The system designer introduces the "Publish Geometry" of structural panels to the space under each panel and uses the outer contour of "Publish Geometry" as the modeling reference to build the overall spacecraft panels. The overall spacecraft panels constructed in this way will have the same contour with structural panels and can be updated automatically with the change of "Publish Geometry" of structural panels.

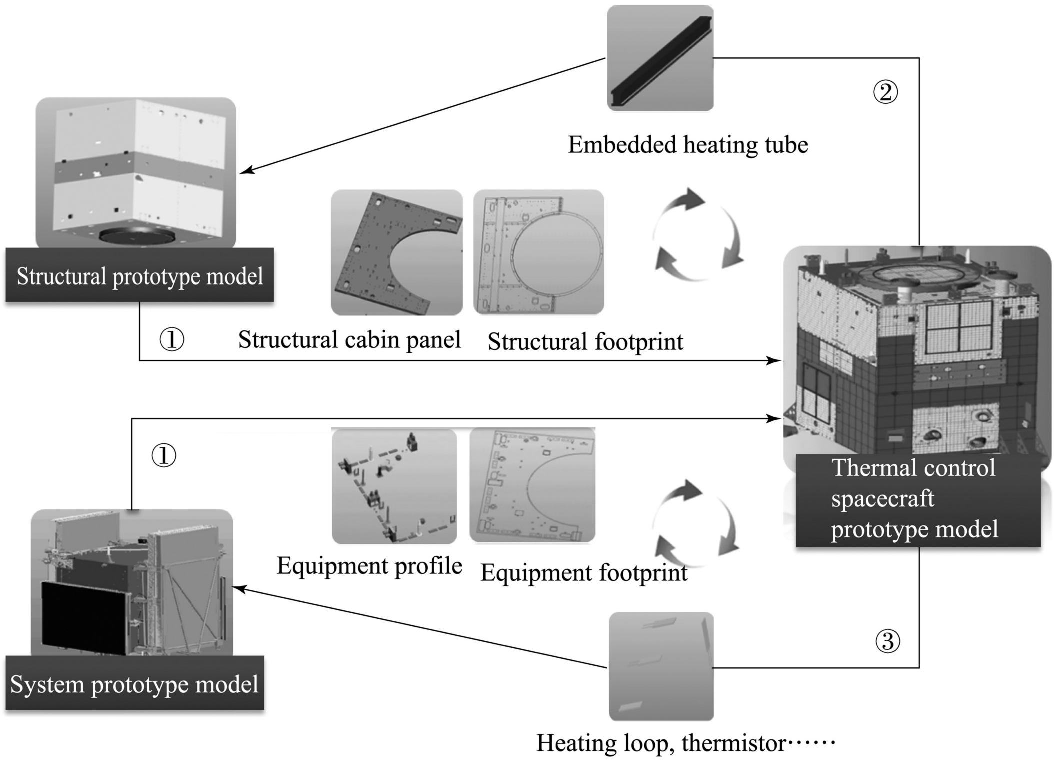

FIGURE 10.7 3D modeling of structural design of a spacecraft system.

4. When the system designer arranges the equipment layout, the structural designer can obtain the hole table and opening information at any time. He can use the design tools to create the holes and openings on structural panels in order to assemble the corresponding embedded parts.
5. After the completion of the above design work, the angle beads, stiffening beams and other direct subassembly components can be installed to obtain a 3D structural design model.

10.3.2.4 3D Modeling of Spacecraft Thermal Control

The 3D modeling of spacecraft thermal control should meet the following requirements:

1. Reflect the specification, number, shape and installation position of thermal control facilities, as well as the connection relationship between them.
2. Reflect the heat consumption information of spacecraft equipment.
3. Contain the collaborative data interfaces with the system design and structural design.
4. Contain necessary process information.

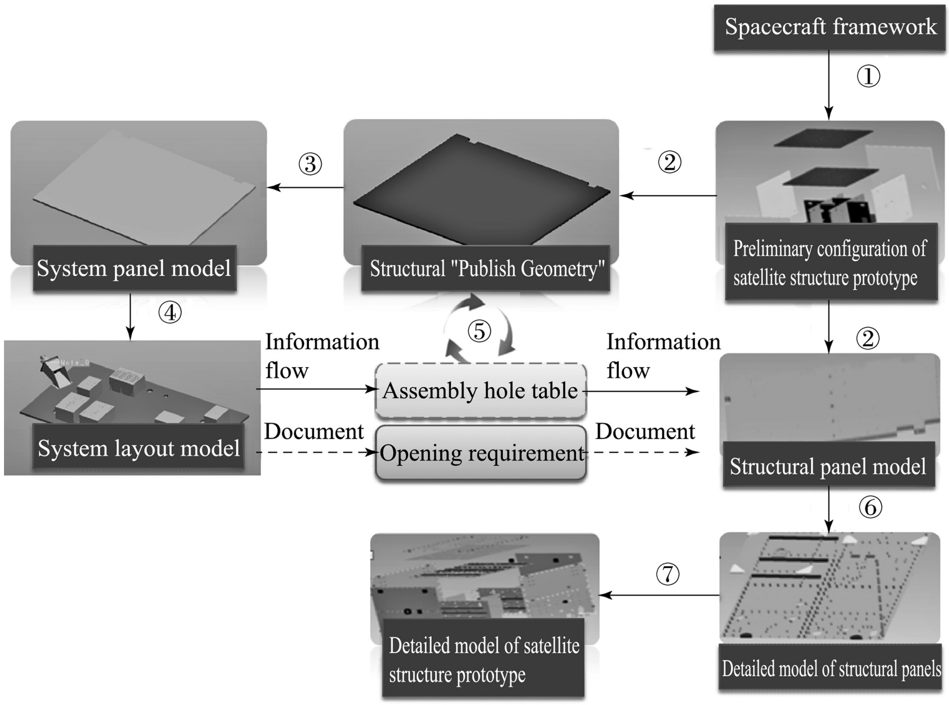

FIGURE 10.8 3D modeling of thermal control design of a spacecraft system.

The 3D design model of thermal control can be constructed according to the following steps (as shown in Figure 10.8):

1. The information on panel shape and panel-lapping position is extracted from the 3D model of structural design, and the information on equipment profile and position is extracted from the 3D model of system design. The two information sources are both used as the design input of 3D thermal control model. Then the thermal control designer creates the preliminary model of thermal control design based on the input conditions through substantialization and other methods and carries out thermal control painting, optical solar reflector (OSR) element layout, film sticking, pin layout, heating element layout and other work.
2. The heat pipe information is fed back, in the form of 3D model, to the structural design, and the interference between the heat pipes and the structural openings and openings is checked.
3. The information on the discrete wires of heating loop and thermistor is fed back, in the form of structured table, to the system designer, as one of the inputs of the 3D laying of cable network.
4. The above steps constitute a process of cyclic iteration. After several rounds of iteration and adjustment, the 3D design model of thermal control design is finally shaped.

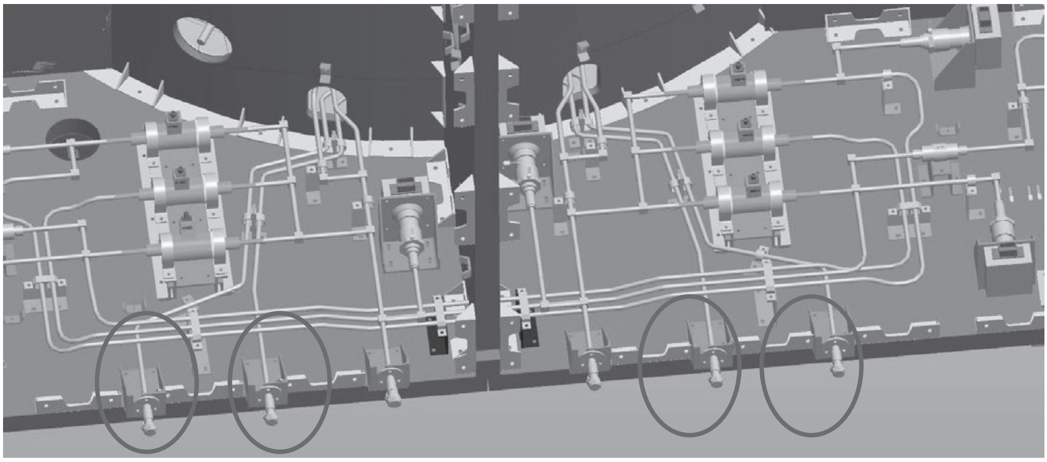

FIGURE 10.9 Iterative adjustment of design models according to interference checking.

10.3.2.5 Iterative Adjustment of 3D Design Model

In the collaborative design process, the detailed designs of the spacecraft system, structure and thermal control are carried out in parallel. The interference check (as shown in Figure 10.9), assembly check, as well as rapid adjustment, confirmation and update of the models are completed collaboratively. The dynamic correlation mechanism of the models is utilized to realize rapid design iteration and reduce multi-disciplinary coupling errors.

The 3D real-time collaboration of the system design, structural design and thermal control design can open up the information links between the system and the structure, between the system and thermal control and between the structure and thermal control. They have not only a clear division of labor and data interface but also share information in real time and carry out parallel design work. Through the correlative mechanism and secondary development of the 3D design tool itself, the real-time notification of input changes is realized and the changes are quickly reflected on the correlation model, so as to ensure the real-time design state consistency of the system, structure and thermal control.

10.3.3 3D Model Construction Oriented to Manufacturing Assembly

As shown in Figure 10.10, after the completion of 3D spacecraft co-design, the 3D design models of the system, structure and thermal control are obtained, respectively. To meet the model application requirements oriented to manufacturing and final assembly, the above design models need to be adapted in four aspects: model simplification, 3D annotation, information extraction and model reorganization.

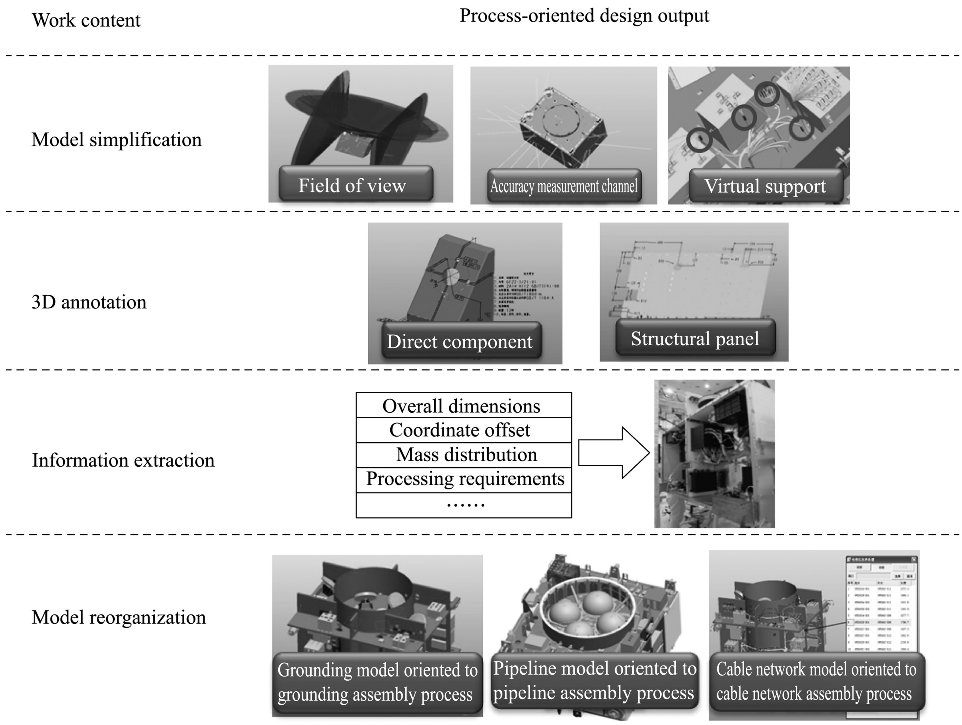

FIGURE 10.10 Construction of spacecraft system design model oriented to manufacturing assembly.

1. Model simplification: Add or delete something to or from the 3D design model. For example, delete the design contents unrelated to production (such as field of view, accuracy measurement channel, virtual support) and add the detailed features needed for production and processing to meet the requirements of process design.

2. 3D annotation: Through 3D annotation, the manufacturing requirements are expressed on the models. Through the information structurization (especially for assembly), the model attribute information required by the process can be expressed in a way that can be recognized by the computer.

3. Information extraction: Extract the corresponding manufacturing information (such as overall dimensions, coordinate offset, mass distribution and processing requirements) from the models and submit the structurized information to the downstream organizations.

4. Model reorganization: Reorganize the models according to the production type. For example, the final assembly models of a spacecraft need to be divided into multiple models for equipment installation, grounding, cables, pipelines, painting and multilayer, which, in turn, are released to the production units for the convenience of subsequent production and assembly.

10.4 COLLABORATIVE SPACECRAFT DESIGN BASED ON EQUIPMENT INTERFACE DATA

In the design process of a spacecraft system, a large amount of coordination data will be generated between the system and subsystems, including optical, mechanical, electrical, thermal and dynamical requirements. These data are of a large quantity and various types and are interrelated. They often need negotiation and adjustment and cannot be fully reflected in the technical specification. Therefore, the quantitative description and process control of the above interfaces need to be carried out through special interface documents.

The existing interface documents used in spacecraft development are mainly Interface Control Sheet (ICD) and IDS. The ICD is generally used to define the interfaces between large systems (such as spacecraft, launch vehicle and ground TT&C system). In the process of spacecraft design, the IDS is mainly used as the basis for the control and coordination of the interfaces between the system and subsystems/equipment.

Compared with the traditional design based on documents, the multidisciplinary design based on interface data has unique advantages in the control degree of technical state, the granularity of interface management as well as the support for the subsequent application design process.

This section focuses on the evolution of IDS, as well as its role in the thermal control design and in the information flow design of cable network.

10.4.1 Role and Evolution of IDS

The IDS contains the mechanical, electrical, thermal, telemetry and telecontrol data interface relations between the spacecraft system and subsystems/equipment. Whenever technical indexes of the system are changed or technical indexes of subsystems/equipment are adjusted, relevant contents in IDS should be modified and jointly confirmed by the designers of the system and subsystems/equipment before taking effect. It can be said that the IDS is the legal basis for the parallel collaborative development of the spacecraft system and subsystems/equipment and is also the most important data link bridging the development efforts for the spacecraft system, subsystems and equipment (as shown in Figure 10.11).

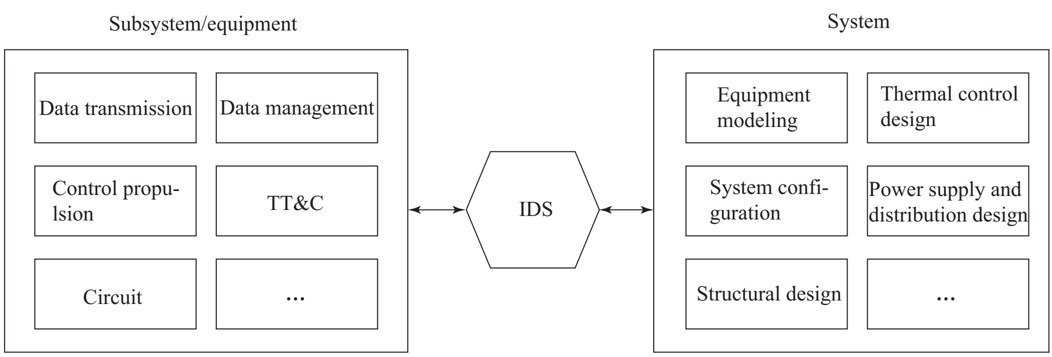

FIGURE 10.11 Role of IDS in the co-development of spacecraft system and equipment.

The IDS documents have long been presented in the form of text documents or tables and controlled by version management in PDM system. Their technical state is controlled in the same way as that of ordinary drawings. Although this method can strictly control the technical state of IDS documents and trace the history of changes back, document is still the smallest management unit and the data in IDS is not structured. When a designer wants to use the data in IDS, he must open the IDS file for manual filtering and identification. When contents of an IDS file are changed, the transfer of the changes still needs to be guaranteed by the designer. When the IDS data transmission chain is too long or the designer's post is adjusted in the development process, the timely transfer of the changes can't be guaranteed so that delays or even errors in design adjustment can be caused. Therefore, although the IDS has established its position as a unified data source for the spacecraft system and subsystems/equipment at this stage, its effect is seriously diluted due to poor data accessibility and difficult change transfer.

With the technological development, the granularity of IDS system management has been refined from the file level to the parameter level. Therefore, parameter is the smallest management unit, and its storage and technical state control is achieved through database. Meanwhile, the design knowledge is integrated into the IDS system, in which the data types are specified and most of the contents are filled in by using the pull-down lists to ensure the standardization and consistency of different IDS files. In addition, the IDS system provides a wide range of APIs so that data can be directly extracted and applied to the relevant operations. Due to these changes, the IDS files have become a truly unified data source for the system and subsystem/equipment co-development (see Figure 10.12).

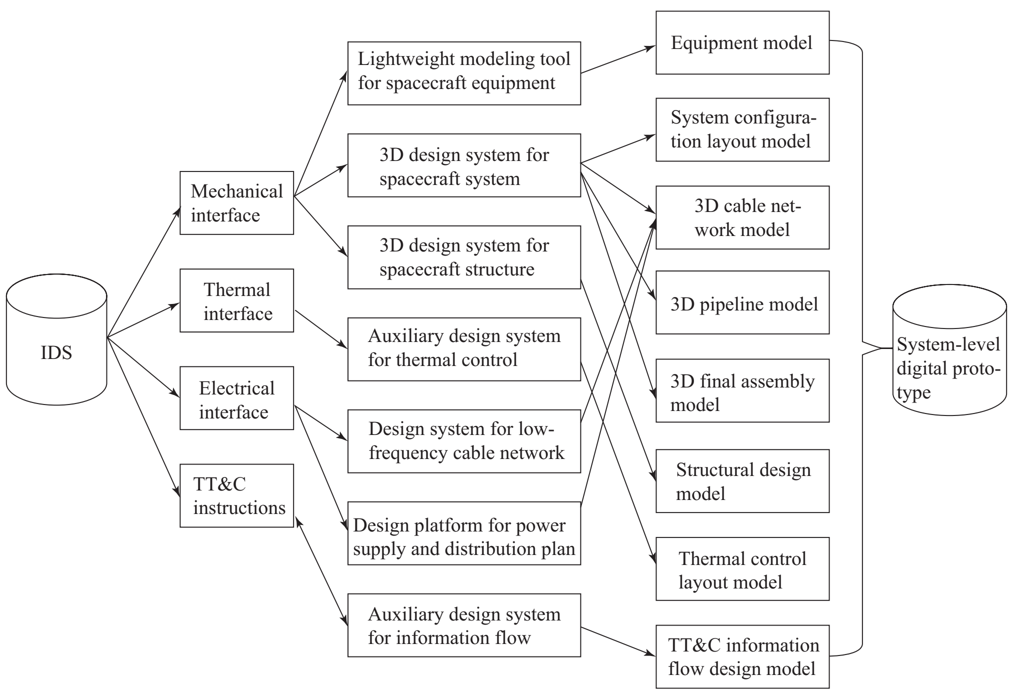

FIGURE 10.12 Unified data application based on IDS.

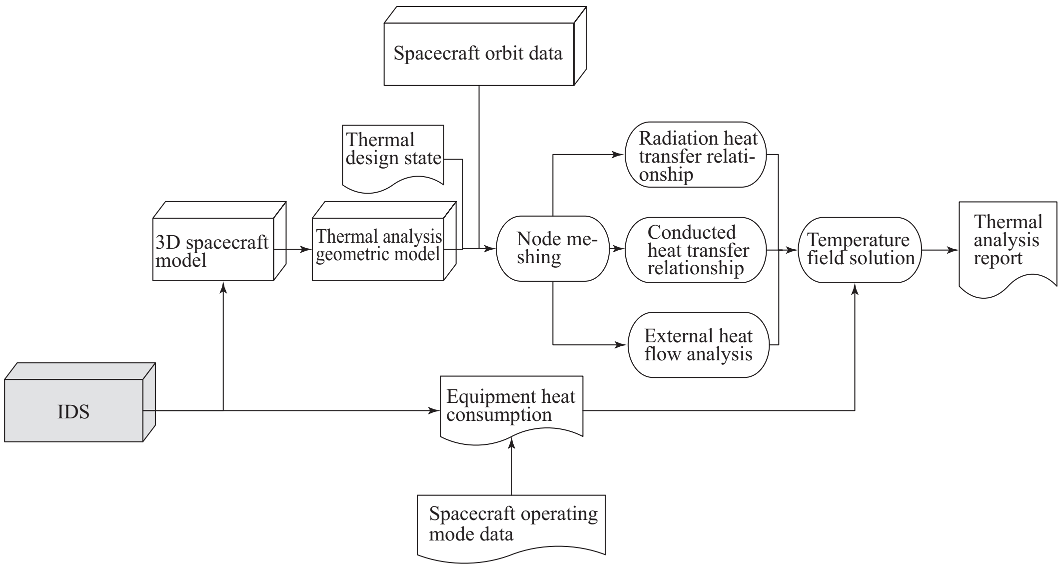

FIGURE 10.13 Digitalization process of thermal analysis of a spacecraft system.

10.4.2 Application of IDS in Thermal Control Design

The thermal characteristics analysis of a spacecraft is mainly to establish a thermal analysis model based on the preliminary thermal control design, analyze and calculate each operating condition, and sort out and analyze the thermal analysis results to verify the correctness of thermal design. In the thermal control design based on IDS (as shown in Figure 10.13), the heat consumption values of the equipment in different operating modes can be directly obtained from IDS as an important input of the system-level thermal analysis. Through the combination of those values with the 3D design model of spacecraft system, an initial thermal analysis model can be shaped quickly, and the efficiency and quality of thermal analysis modeling can be greatly improved.

10.4.3 Application of IDS in Cable Network Design

The cable network design based on IDS (as shown in Figure 10.14) can directly extract the connection relationship between the electrical connectors of different devices of the same model from IDS, as well as the voltage, current, polarity, shielding requirement and twisted-pair requirement of each pin of each electrical connector of each device. It can be used as a direct basis for cable EMC analysis, cable thermal analysis, cable branch design, cable length design, cable production and other operations. Combined with other digital software tools, it can greatly improve the development efficiency of cable network.

10.4.4 Application of IDS in TT&C Information Flow Design

The IDS gives the equipment telemetry parameters and telecommand information necessary for the information flow design of a spacecraft system and provides direct input for the design of telemetry program, telecontrol channel allocation, telemetry channel allocation,

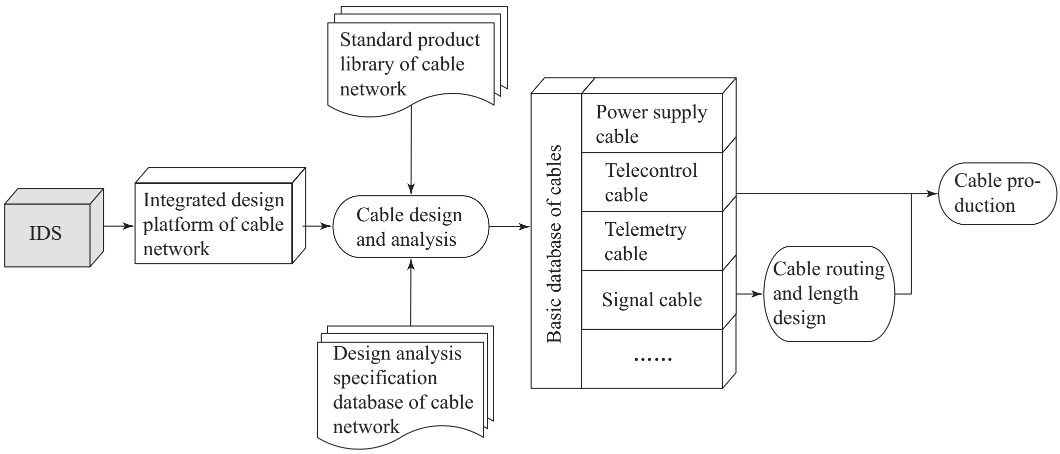

FIGURE 10.14 Digitalization process of cable network of a spacecraft system.

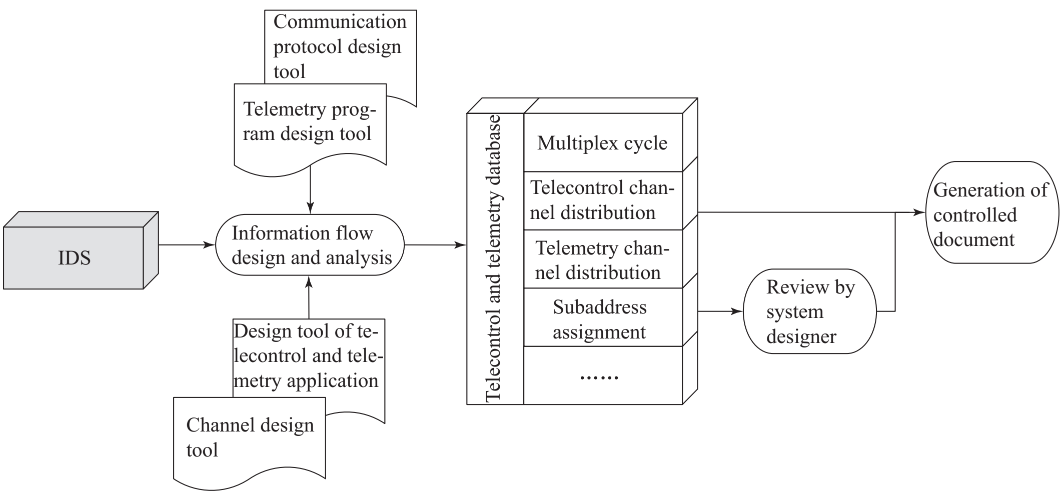

FIGURE 10.15 Design process of information flow of a spacecraft system.

telemetry parameter processing method and telecommand application criteria. On this basis, the above digital design activities (as shown in Figure 10.15) can be carried out with the help of design tools of telemetry program, telecontrol channel allocation, telemetry channel allocation, telemetry parameter processing method and telecommand application criteria. After design activities are completed, controlled documents will be automatically generated as the input and basis for the subsequent development.

10.5 MDO BASED ON MODEL

MDO is a methodology to fully explore and utilize an interactive synergistic mechanism in engineering systems to design complex systems. Its purpose is to obtain the overall optimal solution or engineering satisfactory solution of the system by making full use of the system effects produced by the interaction between various disciplines (subsystems). The general mathematical expression of MDO is:

$$\text{Objective function: } \min J = F(X, R)$$

$$\text{Constraint: s.t. } g(X, R) <= 0, h(X, R) = 0$$

$$\text{Design variables: } X = [x_1\ x_2 x_n]^T,\ R = [r_1\ r_2 r_m]^T,\ L_{j<=} x_{j<=} U_j,\ j = 1,2,\ldots, n$$

where X is the design variable vector, that is, the variable parameter in the optimization process; U_i and L_i are the upper and lower boundaries of the design variable vector element x_i; R is the parameter that affects the objective function but remains unchanged in the optimization process, also known as the system parameter; J is the objective function; g is the inequality constraint function; and h is the equality constraint function. If the objective function is to get the maximum, then it can be expressed as $J = -F(X, R)$.

It can be seen that the MDO is mainly to solve the problem of interdisciplinary coupling and collaborative optimization. As the model-based system engineering (MBSE) becomes an inevitable trend in the development of system engineering, the model-based MDO (MB-MDO) has also become a hot and difficult point in the research and application of engineering problems.

This section will focus on the connotation and characteristics of MB-MDO and the working process of MB-MDO.

10.5.1 Connotation of MB-MDO

MB-MDO is a method of multidisciplinary simulation analysis and design optimization based on model-based spacecraft system engineering. As a model covering the overall spacecraft system design, the MB-MDO includes not only the traditional design, analysis and simulation models in the disciplines such as mechanical engineering, electronics, thermal control and orbit, but also the system models describing the system-level requirements, parameters, structure and behavior and the relationship between a system model and a discipline model.

The implementation process of MB-MDO mainly includes the building of spacecraft system mathematical model, the building of system simulation model, the division of system (spacecraft) and subsystems (disciplines), the decomposition of system tasks and indexes, the building of mathematical model for each discipline, the building of simulation model for each discipline, and the multidisciplinary analysis and optimization. The division of system (spacecraft) and subsystems (disciplines) is an important basis for MDO and an important part of system modeling. The system model generated by system modeling is verified by simulation analysis, covering the overall technical requirements of spacecraft system, the system's technical requirements for each subsystem and the system concept. In short, the MB-MDO in essence is to establish the system model through cyclic iterative simulation by verifying the rationality of system decomposition; to build the mathematical model for each discipline based on system model; and to integrate the design and analysis tools of each discipline with the knowledge on each discipline to carry out multidisciplinary comprehensive simulation analysis and optimization.

MB-MDO is the inheritance and development of MDO, that is, the simulation, analysis and optimization of multi-disciplinary mathematical model is the inheritance of MDO. Model-based discipline decomposition is a development of MDO.

10.5.2 Spacecraft Design Process Based on MB-MDO

The MB-MDO is based on system model to model the design variables, objective functions and constraints in the mathematical models of MDO engineering problems. Disciplinary-level simulation analysis and optimization is the bridge between multidisciplinary demand analysis and design optimization. It is mainly to simulate, analyze and optimize the mathematical models of various disciplines in the system model, involving the sensitivity analysis of each discipline, the establishment of agent model, the selection of optimization algorithm, and the encapsulation and calling of simulation analysis and optimization tools. MDO is the ultimate engineering goal of MB-MDO. An MDO and simulation model adapting to the spacecraft system-level optimization can be established to obtain the system-level optimal or feasible solution set through combing the coupling relationship between disciplines, analyzing the sensitivity of multiple disciplines (subsystems) to the system, building the approximate agent models of complex discipline/system models, selecting the MDO optimization algorithm and strategy, and establishing the MDO framework software. Through the comparison between the optimal or feasible solution set and the parameter model of system model, the improvement and perfection mechanism of system requirement model is proposed to drive the change of system model through the change of requirement model, so as to realize the cyclic iteration of system model, discipline-level analysis and optimization simulation model and multidisciplinary design and optimization model. The detailed process is shown in Figure 10.16.

10.6 SPACECRAFT LIFECYCLE DATA MANAGEMENT AND CONFIGURATION MANAGEMENT

The lifecycle of an aerospace model product is generally divided into the following phases: project approval, feasibility demonstration, conceptual design, prototype design, flight model design, launch and in-orbit flight. In the whole process of aerospace model development, the organization, management and transfer of product data run through all the key development steps. The PDM provides a fundamental guarantee for the smooth implementation of technical state management in the aerospace model development.

The rapid development of aerospace technology has increased the complexity of aerospace models and the content of new technologies, further shortened the development cycle, strengthened the coupling, synergy and parallelism, and increased the difficulty of configuration control. It has become a consensus among advanced military development organizations at home and abroad that the information tools shall be used for the management of a large amount of data generated in the process of product development, so as to realize accurate configuration description, correlative evolution and lifecycle traceability.

Based on the application practice of lifecycle data management and configuration control of aerospace models, this section systematically studies the lifecycle data management for aerospace models, the technologies for their configuration management and the

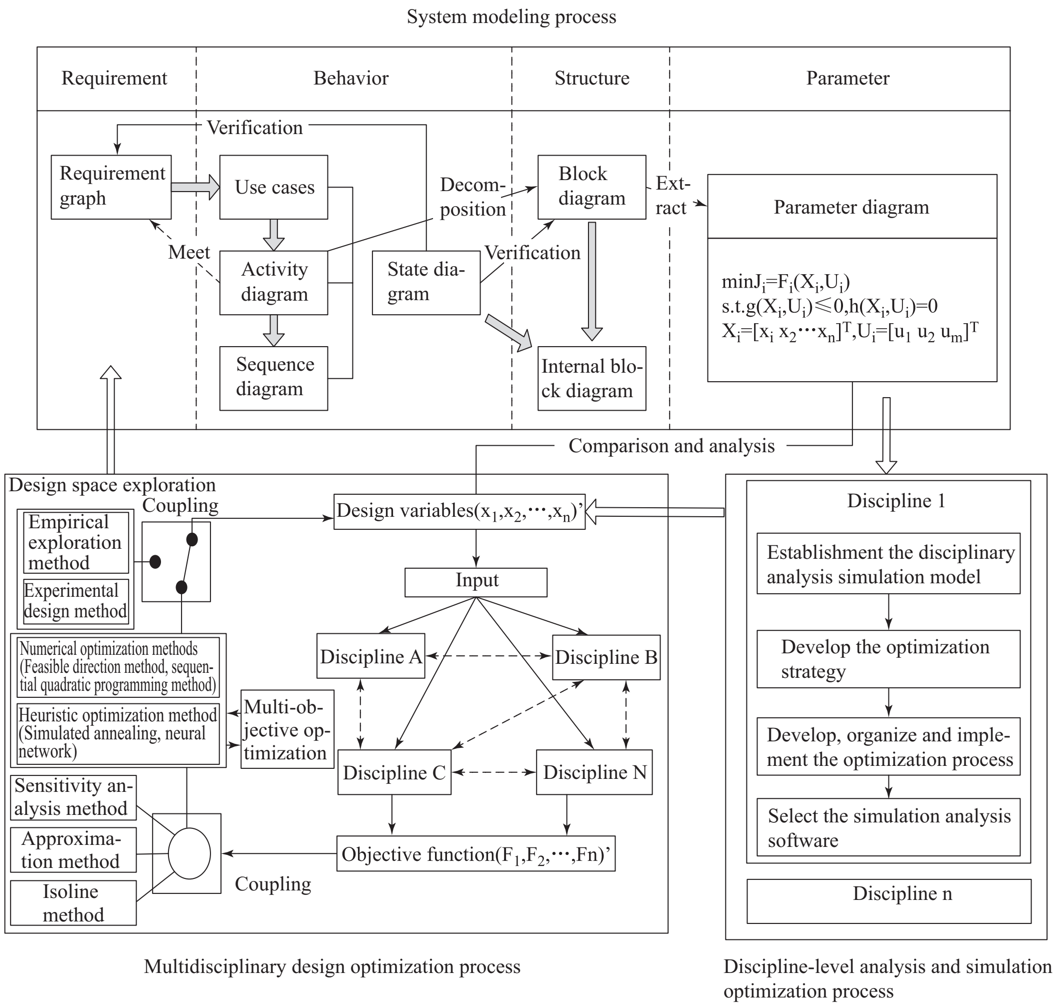

FIGURE 10.16 MB-MDO process.

Aerospace Vehicles Integrated Design and Manufacturing (AVIDM)-based lifecycle data management and configuration control of aerospace models.

10.6.1 Lifecycle Data Management for Aerospace Models

The overall characteristics of aerospace model development can be summarized from different dimensions such as organization, product and process, as shown in Figure 10.17.

1. Organizational dimension: the participation of many organizations, and the prominence of distributed collaboration

   Due to the high complexity of aerospace model products, many development organizations in different regions need to work together. Relying on aerospace system engineering, all the subsystem and equipment developers need to cooperate with each other under the control of system developer to jointly complete the

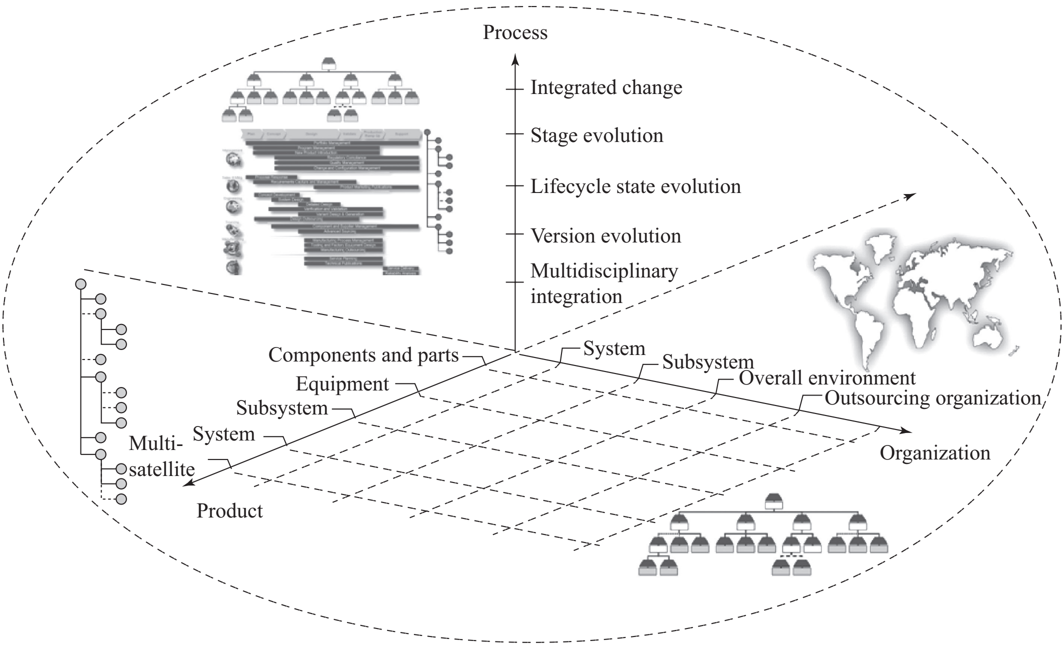

FIGURE 10.17 Characteristics of spacecraft system model development.

product development. The whole collaboration process is characterized by distributed collaboration.

2. Product dimension: the presence of multiple configurations and the parallelism of multi-satellite states

   Aerospace model products are divided into systems, subsystems, equipment, components and parts at various levels, involving mechanical, electrical, software, optical and thermal disciplines and covering a wide range of data to be controlled. The model development process is generally divided into the following stages: M (model sample), C (prototype sample) and Z (flight sample). The configuration of each stage needs to be solidified and traced. In addition, one model may have multiple configurations. Each configuration may correspond to multiple in-orbit model products. The configuration of each product needs to be managed and traced.

3. Process dimension: the complex development process characterized by spiral iteration

   The model development process is typically an iterative process of "improvement—testing" with the characteristic of phased development. Such a long-time model development process is inundated with a lot of innovations and iterations, such as: multidisciplinary integration, version evolution, lifecycle state evolution, development phase evolution, and closed-loop change.

The spacecraft development process can be divided into several stages, such as conceptual design, prototype design and flight model design. It is implemented at three levels, project management, configuration management and collaborative R&D, as shown in Figure 10.18:

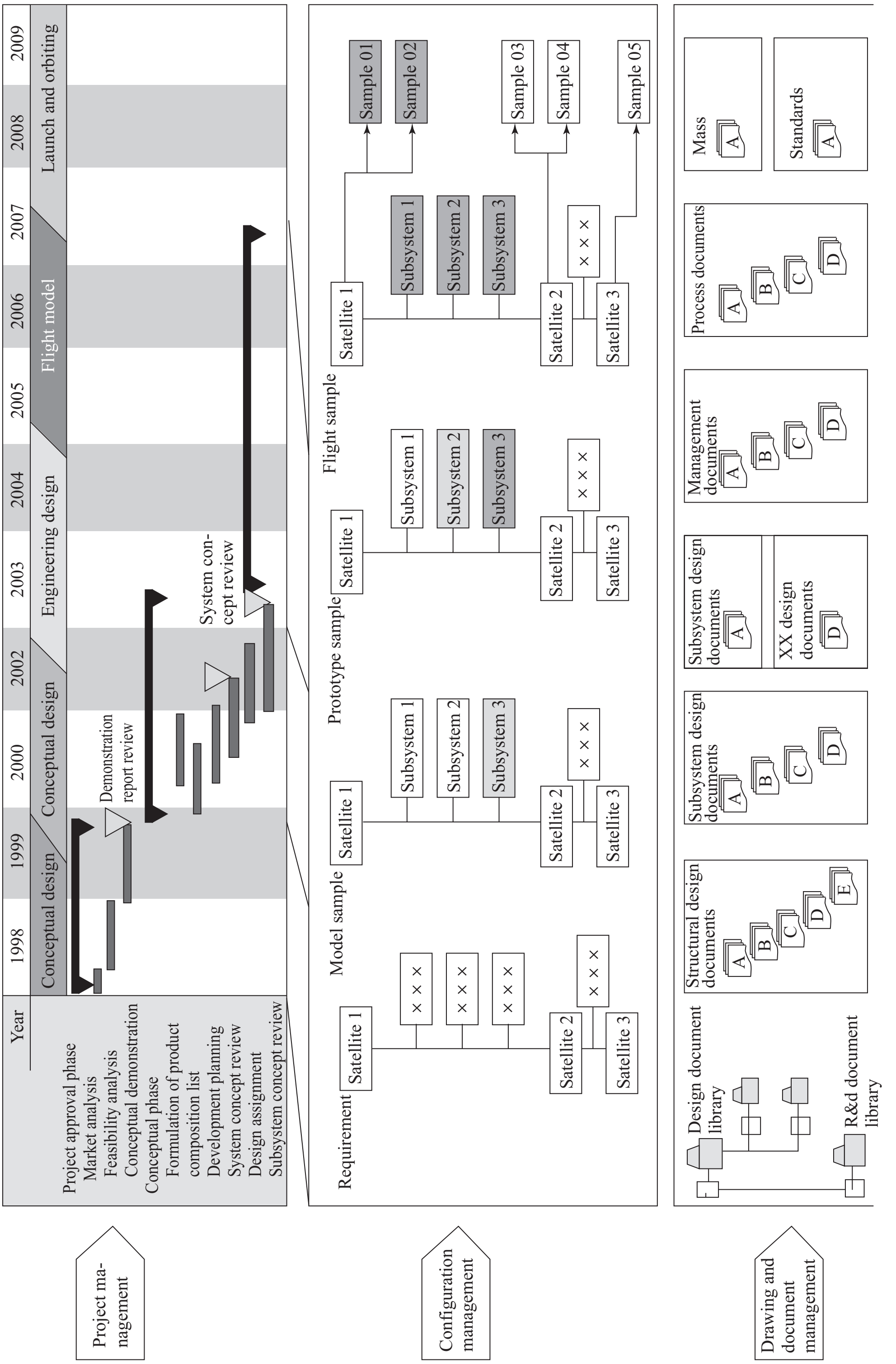

FIGURE 10.18 Overall process of aerospace model development.

Level 1 (model development project management): to manage the whole development process according to the requirements of aerospace development phase and the Work Breakdown Structure of model development work. Generally speaking, the work at level 1 is divided into technical and command management lines. The task division, scheduling and implementation of the project constitute the top level, that is, the project management level. The whole model project management process runs through the conceptual, model sample, prototype sample, flight sample and in-orbit management stages.

Layer 2 (configuration management): to manage the configuration at each stage of the model development process. The composition of a model product at each stage and all associated drawings and documents as the solidification result at the transition stage are referred to as specific configuration. The related configurations at different stages are M, C, Z and others.

Level 3 (collaborative R&D level): to carry out collaborative R&D among various disciplines and departments. At present, the development and production mode of aerospace models is gradually changing. The models are increasing. Moreover, the one-to-one relationship among one model, one set of drawings and one physical object has gradually changed into the many-to-many relationship among multiple models, multiple sets of drawings and multiple physical objects.

It can be seen from the perspective of model application that the spacecraft lifecycle data management has organically integrated the product data, configurations, engineering changes, and R&D collaboration into a whole (as shown in Figure 10.19).

10.6.2 PLM-based Spacecraft Configuration Management Technology

Configuration management is a set of activities to guide, supervise and control the determination and change of product configuration in the whole product lifecycle by organically combining technical, administrative and management means in order to ensure that the product meets the requirements of applicability and conformity and the user's requirements.

The configuration management in aerospace model development is to, based on the structure tree of a product model, integrate the management of quality and reliability data packets of the model, subsystems and equipment with the related operation system and tool software, provide the product structure management, validity management, baseline management, multi-view management, change management and other functions, and thus ensure the integrity and traceability of quality and reliability data during development. For a model in batch production, the unique correspondence between the model data package (including product structure, design data, process data and all kinds of bills) and the spacecraft batch shall be defined, and the batch application scope of the changed object shall be recorded in various change documents. In this way, the unified control of configuration objects (data) of the same model and different batches in different stages of batch production can be lowered from model level to batch level. It is necessary to ensure the legibility, accuracy and consistency of configuration identification of a batch model in different life stages, processes and batches as well as the integrity and traceability of its quality and reliability data during development.

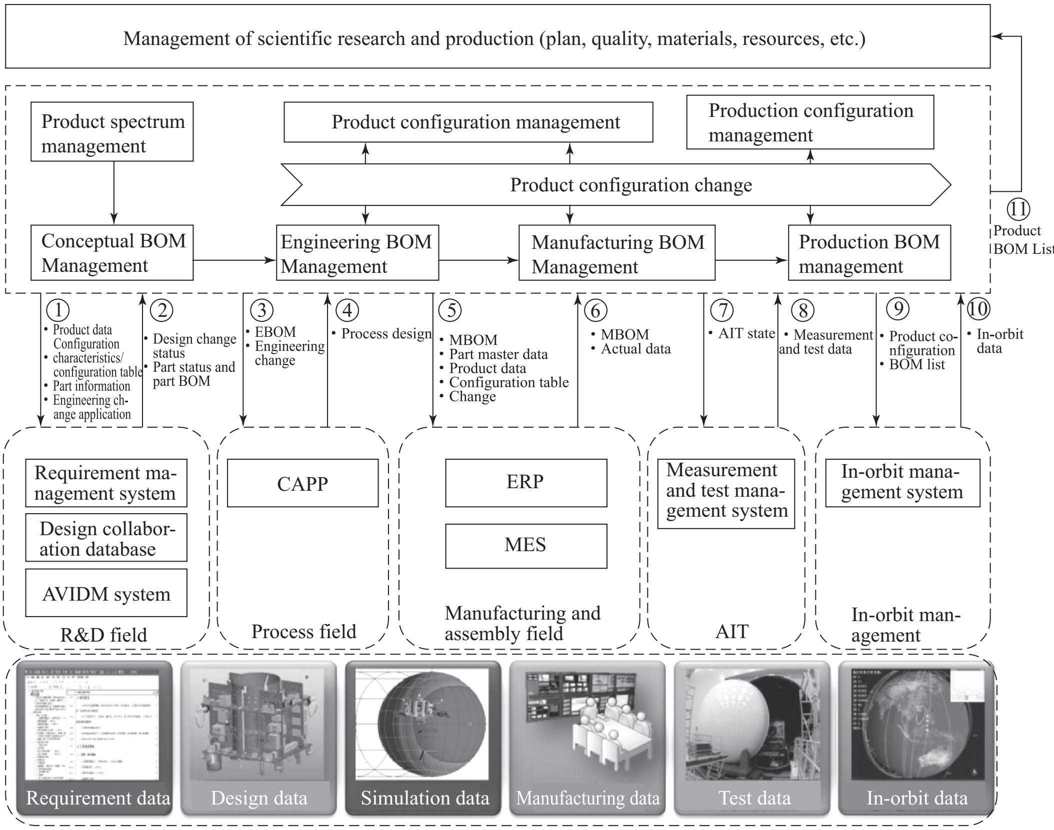

FIGURE 10.19 Spacecraft system lifecycle data management.

10.6.2.1 Product Structure Management

The 3D models, 2D drawings, technical documents, process information and other data are organized with product structure as the core. The product structure is managed by several levels, including model (stage) level, subsystem level, sub-subsystem or multi-function assembly level, equipment level, component level and part level. On the basis of product structure management, the technical documents, test reports, requirement reports, design data and process data of any structure tree nodes are effectively correlated to achieve the unique unified and centralized management of model development data based on product structure, support the fast data operation based on product structure, and provide a complete data base for the sharing of peripheral system data.

10.6.2.2 Data Packet Baseline Management

Based on the unified product structure, the unified state calibration of data packet versions is realized for the product in a certain milestone (such as test, factory integration test, formal delivery). That is to say, a unique mapping relationship is established between the data packet version and the part/component/assembly version or product structure version or technical document version to form the functional, distribution, product and other data baselines. In this way, the baselines can be compared for the same product at different stages and for different products at the same stage to track, trace and control the configuration changes of products. Finally, a management approach is established for product quality tracking so that the quality defects in the delivered products can be quickly located and found by the design department and the product quality can "return to zero" easily.

10.6.2.3 Validity Management

By defining the validity of product structures/product documents, the product structures, 3D models/2D drawings, technical documents, process information and other data of any batch of aerospace products can be accurately recorded to ensure that the users can access the existing valid data.

10.6.2.4 Multi-view Management

Based on the product structure, the bill-of-material (BOM) structures of multiple views are established from different perspectives, and their evolution processes and relations are managed. On this basis, the BOM structures of different views can be reconstructed, the product structure conformance of multiple BOM views can be verified, and the change results of upstream view BOM can be automatically transferred to the downstream view BOM.

10.6.2.5 Change Management

Based on the product structure tree combined with version management and configuration management, the history, impact scope, reason for change, and other information on the changed object can be accurately recorded to realize the traceability of the change process and change data, so that the configuration evolution can be effectively documented. At the same time, through the message service management, the product changes can be timely notified and fed back.

10.6.2.6 Integrated Management

One of the bases for technical state management in the whole process of model development is the single data source of multi-disciplinary integration. Through the integration interfaces, a data integration link can be established from technical state management to spacecraft co-design system, CAPP system, Kanban management system, measurement and test management system, batch-production project management system and other operation systems as well as the related structural design, electronic design and other tool software.

10.6.3 AVIDM-based Aerospace Model Lifecycle Data Management

After 20 years of development, the AVIDM[6] has become the cross-regional and cross-place core support platform with Chinese aerospace characteristics for model PDM and configuration control (as shown in Figure 10.20). In the space industry, a product family represented by aerospace vehicle product data management (AVPDM)/ aerospace vehicle plan (AVPLAN)/ aerospace vehicle test data management (AVTDM) and other products with fully independent intellectual property rights has taken shape. It has become an important proprietary technology and product supporting the information construction

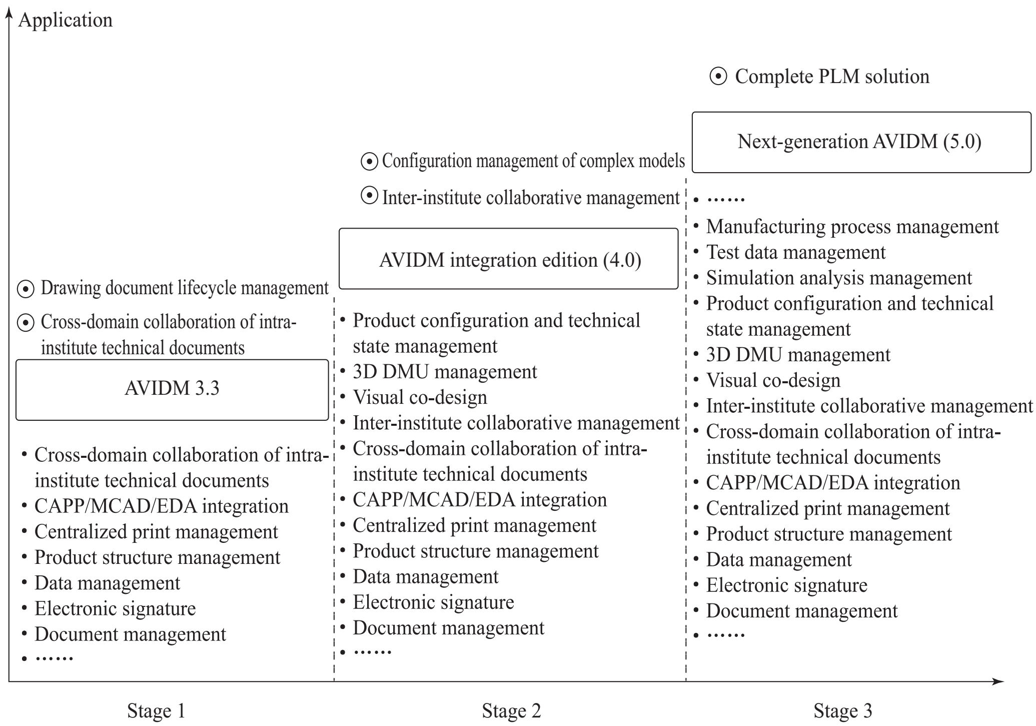

FIGURE 10.20 AVIDM development path of a spacecraft product.

of space engineering and a representative proprietary product in the field of Chinese industrial software. Based on AVIDM system, the drawing documents, 3D models and other development data of aerospace products can be managed, and the continuous, complete and traceable configuration control oriented to the whole lifecycle can be realized.

In China's aerospace industry, the wide application of aerospace vehicle integrated design and manufacturing (AVIDM) system starts from AVIDM 3.3. According to the development orientation of AVIDM products, AVIDM3.3 is mainly orientated to the product data and process management based on document mode, with the basic idea of solidifying and applying the original working mode. AVIDM4.0 is orientated to the model data organization and management based on product structure, focusing on solving the problems in 3D data management, configuration management and inter-institute collaboration in order to realize structured data management and operation delicacy management in the model development. AVIDM5.0 will extend the capabilities of simulation analysis, test data management and manufacturing process management support to obtain a PLM solution needed for supporting the aerospace model development.

The functional framework of AVIDM is shown in Figure 10.21. The main functions of AVIDM include system management, data management, workbench management, product structure and configuration management, product management, repository management, change management, interdomain collaboration as well as the integration of AVIDM with other software and tools.

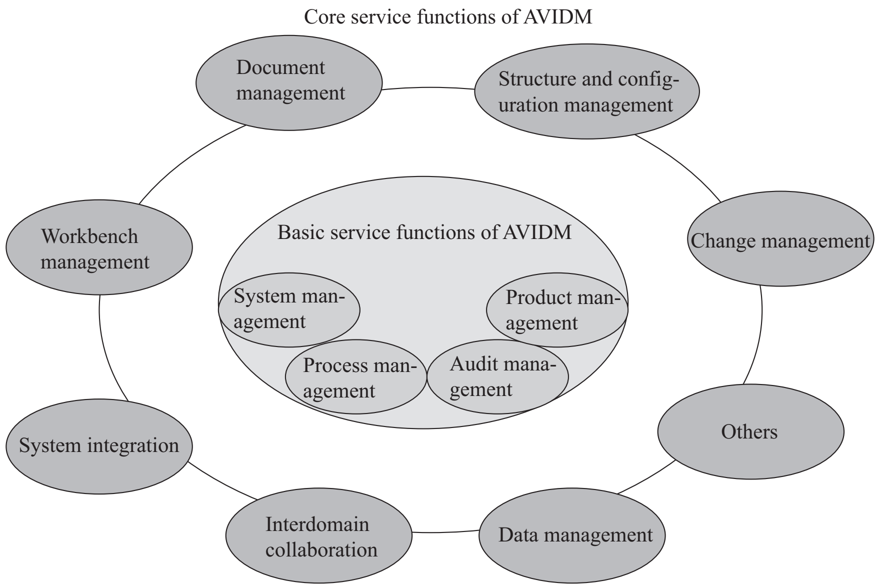

FIGURE 10.21 Spacecraft AVIDM function framework.

10.7 SPACECRAFT CO-DESIGN ENVIRONMENT

The spacecraft product development is characterized by complicated conceptual design and detailed design, extensive participation, high technical index requirements, a high degree of design coupling and a lot of collaborative discussion and analysis. However, the traditional collaboration mode is temporary and difficult to meet the need for the collaboration among various disciplines. By establishing a professional collaborative design environment equipped with appropriate software and hardware, the designers in different disciplines can work collaboratively in the unified software and hardware environment, data environment and visual environment to support the development of the concept, prototype and flight model of complex products so as to improve the collaborative efficiency and quality.

The basic elements of spacecraft co-design environment include the following.

10.7.1 Process of Collaborative Design

According to the different stages of spacecraft development process, different collaborative design activities can be carried out. The typical collaborative design process includes three stages:

1. Preparation
2. Centralized design discussion and analysis
3. Data organization and report writing

10.7.2 Team of Collaborate Design

At the beginning of collaborative spacecraft development, an IPT composed of multidisciplinary professionals (including the user) shall be established. The team is a group of system-level and subsystem-level professionals. Among them, the system-level professionals include those engaged in system design and mission analysis, while the subsystem-level professionals include those engaged in control propulsion, structure, thermal control, power supply and distribution, TT&C and payload. The whole team works in a fixed office space during the collaborative design, and each member has a fixed workstation. All technicians in different disciplines and all the managers can work collaboratively in the unified software and hardware environment, data environment and visual environment.

10.7.3 Areas of Collaborative Design

In order to meet the requirements of co-design activities in the development of complex spacecraft products, a certain space should be occupied and generally divided into the following areas with different functions:

1. Area of centralized design
2. Area of grouped design
3. Area of visual simulation
4. Area of service assurance

10.7.4 Basic Software and Hardware Environment for Collaborative Design

The collaborative design environment must be equipped with necessary hardware and software resources to ensure both resource sharing and real-time communication.

10.8 PROSPECT OF DIGITAL SPACECRAFT DEVELOPMENT

Digitalization is an important support means to improve the efficiency and quality of spacecraft development. The deep integration of digital technology and development process is an important direction of future spacecraft development. The digital spacecraft development is transitioning from improving the design efficiency to changing the spacecraft development mode. With the rapid development of digital technology, a number of new techniques and methods have emerged and profoundly revolutionized the spacecraft development mode.

10.8.1 Application of Cutting-Edge Digital Techniques

10.8.1.1 3D Printing [7]

The3D Printing, also known as additive manufacturing or incremental laminar manufacturing, is a rapid forming technique invented by the Massachusetts Institute of Technology in the 1990s. It can turn design ideas into real product models automatically, directly, quickly and accurately without the need for machining or molding, so as to rapidly evaluate, modify and functionally test the products at the design stage and effectively shorten the product lead time. The 3D printing, directly driven by 3D CAD model, can directly output the products, without the need for intermediate processes (such as blank preparation, parts machining and assembly) and expensive tools and molds. In addition, it is not affected by the shape and structure of the parts, which makes it possible to directly manufacture complex models, further eliminate the gap between design and manufacturing and provide unlimited creative space for designers.

In the aviation field, Boeing has used 3D printing to make about 300 kinds of different aircraft parts. The European company Airbus has laid out a 3D printing roadmap, starting with printing the small parts of an aircraft and ending up with printing the entire aircraft by 2050. In the spaceflight field, the NASA put forward the "Made in Space" program in 2010. It will launch a specially developed 3D printer into space to directly print the parts of a spacecraft and a space station in space, which will then be assembled in gravity-free condition and directly applied. While reducing the development cost, this method can avoid a series of risks brought by the spacecraft development and launch. As a cutting-edge and pioneering emerging technology, 3D printing is making profound changes in the traditional production mode and production process and has good development potential and broad application prospects.

10.8.1.2 Virtual Reality/Augmented Reality[8]

Virtual reality (VR) is a technology that integrates computer graphics and various interface devices (such as reality and control) to provide an immersive feeling in an interactive 3D environment generated on a computer. Augmented reality (AR) is an emerging technology that has gradually evolved from VR. It is based on computer display and interaction, network tracking and positioning and other techniques. In the AR technology, the virtual information generated by computer is superimposed and fused into real scenes as a supplement to the real world in order to provide users with an enhanced visual, auditory and tactile experience of the real world.

The difference between VR and AR is that, the VR presents virtual things and scenes, while the AR is a combination of "real+virtual" by superimposing virtual information on real scenes.

Large international aerospace enterprises are actively exploring the application of VR/AR technologies in their own domains. Boeing, Airbus, Lockheed Martin, NASA and other aviation giants have set up the VR/AR labs. Among them, Lockheed Martin took the lead in applying the VR technology to F-22 and F-35 projects. The immersive engineering alone saved more than $100 million for F-35, with an ROI (return on investment) of 15 times. NASA is also actively exploring the VR/AR applications in space, including the use of immersive headsets to remotely control the robots in the missions and the use of VR to train the astronauts on the ground before going into space so as to improve the efficiency of their missions.

10.8.1.3 Digital Twin [9–11]

Digital twin (DT) is to digitally present physical objects in virtual space, that is, to create virtual models for physical objects by digital means to simulate their behavior characteristics in the real environment. DT is the bridge between the real world and virtual world.

In the process of product development (including design, manufacturing and testing), DT can be used to early plan, deploy, simulate and verify the products in the virtual world and to effectively find and timely avoid various problems in the development process so as to greatly improve the production efficiency and quality.

In terms of DT application, General Electric (GE) uses big data, Internet of Things and other advanced technologies to realize the DT-based real-time monitoring, timely inspection and predictive maintenance of the engines through Predix, a cloud service platform built by GE. Similarly, the US Department of Defense uses DT to maintain and guarantee the aircraft health, that is, to establish digital aircraft models in the virtual space. Through sensors, the real aircraft state is acquired in real time and fed back to the digital model to realize the complete synchronization of virtual and real products. After each flight, the existing situation of a structure and its past load are based on to timely analyze and evaluate whether the maintenance is needed.

10.8.2 MBSE

With the constant increase of system scale and complexity, the problems faced by document-based system engineering are becoming more and more prominent. For example, the information representation is inaccurate and ambiguous. It is difficult to find the required information from the massive documents. Logic is lacked between the documents and reports. The correlation analysis of problems is difficult and cannot be connected with the designs in other engineering fields.

Owing to the advantages such as perceptual intuition, unambiguity, modularization and reusability, the MBSE has quickly covered the software, electronics and other engineering fields.[11,12] According to the MBSE definition given by International Council on Systems Engineering (INCOSE), the MBSE is to "support the system requirements and the design, analysis, verification and validation activities through formalized modeling from the conceptual design phase and throughout the development process and subsequent lifecycle phases". In the INCOSE's "Vision 2020", MBSE is identified as an important future development direction of system engineering.

NASA has applied MBSE to a number of projects with the aim of improving the project affordability, shortening the development time, effectively managing the system complexity, and improving overall system quality. NASA Jet Propulsion Laboratory (JPL) has been planning and implementing the MBSE application strategy "Integrated Model-centered Engineering" since 2009 to apply MBSE in all stages of the spacecraft lifecycle. Boeing's MBSE application is based on an integrated product architecture combined with Boeing product features. Chinese companies in the aviation, aerospace and weapon sectors are strongly interested in the MBSE research and application, but they are mostly in the exploratory stage. They lack an MBSE-based collaborative design process, a modeling method or specification for designers' reference, and an integrated modeling simulation platform and development environment.

10.8.3 Model-based Enterprises (MBEs)

How to seize the opportunity in the increasingly fierce competitive environment is an important challenge faced by every manufacturing enterprise. With the wide application of MBD, the model-based thinking has been deeply rooted. In order to improve competitiveness, the industry must fundamentally build a digital mindset and must construct the MBEs to promote the strategic transformation of enterprises. MBE[13] is a manufacturing entity with model-based thinking, which uses modeling and simulation for the thorough improvement, seamless integration and strategic management of all technologies and business processes in the design, manufacturing and product support, and uses models to define, execute, control and manage the whole process of the enterprise in order to radically reduce the time and cost of product innovation, development, manufacturing and support.

MBE has become the embodiment of modern advanced manufacturing system and the representative of future digital manufacturing system. MBE is no longer the pure application and promotion of a new technology or method, but the incarnation of national strategy and future advanced manufacturing technology. Its definition and connotation are constantly enriched and improved. So far, more and more manufacturing enterprises are participating in the MBE capacity building.

Spacecraft development is a large-scale multidisciplinary system engineering. With the continuous increase of spacecraft development projects, the continuous reduction of lead time and the continuous improvement of performance indexes, advanced digital technologies must be relied on to transform the traditional development mode, further improve the efficiency and quality of spacecraft development, and innovate the development of aerospace products.

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