Power supply system for a micro-satellite

Zheng, Chao

2008

Zheng, C. (2008). Power supply system for a micro-satellite. Master's thesis, Nanyang Technological University, Singapore.

<https://hdl.handle.net/10356/41402>

<https://doi.org/10.32657/10356/41402>

Power Supply System For A Micro-Satellite

Zheng Chao

School of Electrical & Electronic Engineering

A thesis submitted to the Nanyang Technological University
in fulfilment of the requirement for the degree of
Master of Engineering

2008

Acknowledgements

It is my pleasure to express my sincere appreciation to my supervisor, Professor Low Kay Soon, for his support, advice, encouragement and guidance over the past two years. Without his technical advice, this dissertation would not be possible.

I also appreciate the financial support provided by the DSO National Laboratories. Dr. Goh Cher Hiang from DSO National Laboratories has also provided invaluable advice during the course of this research.

Thanks are due to my colleagues in Satellite Engineering Center (SEC), Ms. Fu Ying and Mr. Ting Yeh, for their support and discussions on developing the system and conducting of the tests.

Last but not least, I am sincerely thankful to my family and friends, for their understanding, encouragement and support through all these times.

Table of Contents

Acknowledgements i

Table of Contents ii

Summary v

List of Figures vii

List of Tables xi

Chapter 1 Introduction 1

 1.1 Motivation and overview 1

 1.2 Objectives 5

 1.3 Main contributions 6

 1.4 Organizations of the Thesis 7

Chapter 2 A review of satellite power supply system 9

 2.1 Power system architectures 9

 2.2 Solar array 19

 2.3 Battery 27

 2.4 Power converters 32

 2.5 Conclusions 36

Chapter 3 X-SAT power supply system 37

 3.1 Satellite mission overview..... 38

 3.2 Sizing of solar panels and battery capacity 41

3.3 Power supply system design	47
3.4 Conclusions	59
Chapter 4 Lithium-ion battery modeling and state of charge	
estimation	61
4.1 Basics of lithium-ion battery	62
4.2 Proposed battery model	65
4.3 Conclusions	74
Chapter 5 Electrical Ground Support System	75
5.1 Virtual instrumentations	75
5.1.1 VI system architecture	76
5.1.2 System modeling	80
5.2 Conclusions	87
Chapter 6 Experiment results	88
6.1 Whole orbit test	88
6.2 System efficiency analysis	93
6.2.1 Power efficiency analysis	93
6.2.2 Solar energy utilization factor analysis	95
6.3 Power balance analysis	97
6.3.1 Launch and early operation stage	99
6.3.2 PPU operation in normal operation stage	101
6.4 Battery module analysis	103
6.4.1 Battery trickle charge observation	103
6.4.2 Battery latch-up analysis	104

6.4.3 Battery SOC estimation	106
6.5 Summary of the whole orbit tests results	109
6.6 Thermal cycling test	110
6.6.1 Operation scenario test	110
6.6.2 Functional test	115
6.6.3 Observations and discussions	116
6.7 Conclusions	126
Chapter 7 Conclusions and recommendations	127
7.1 Conclusions	127
7.2 Recommendations for future works	128
Author's Publications	130
References	131
Appendices	140

Summary

Power supply system is a key subsystem of a micro-satellite. Its performance and reliability greatly affects the satellite operation. In this dissertation, the development, testing and performance evaluation of the power supply system of X-SAT, which is a micro-satellite being developed by Nanyang Technological University and DSO National Laboratories, is presented.

In this thesis, the characteristics of critical satellite power supply system modules such as the solar array, rechargeable battery and voltage regulators are first investigated. The power system topologies such as the direct energy transfer and maximum power point tracking topology are then reviewed. Due to the various shortcomings of the existing topologies, a new satellite power supply system architecture has been proposed and developed. The new system adopts a modified direct energy transfer topology by eliminating the shunt regulators and adding a main bus regulator to increase the solar energy utilization as well as minimizes the battery latch-up effect in the cold environment.

From the instrument requirements, a new model is developed to estimate the solar cell parameters using the temperature and incident angle factor. Besides the solar array model, a combined lithium-ion battery model based on the current available battery models is also proposed. The proposed model is capable of estimating the battery real-time state of charge based on the knowledge of battery run-time output voltage and current values. The proposed models have been implemented into the integrated testing system. To test the system performance, an integrated system based on the virtual instrument concept has been developed to investigate the real-time satellite operations on the ground. The system

includes the effect of space environmental temperature profile, the orientation of the satellite as well as the dynamic loading during the mission operation. Whole orbit test results through the functional and thermal cycling tests are presented and discussed. Moreover, the performance of the power supply system is evaluated.

List of Figures

Fig. 2.1	DET un-regulated bus architecture	10
Fig. 2.2	DET sun-regulated bus architecture	12
Fig. 2.3	DET fully regulated bus architecture	13
Fig. 2.4	MPPT regulated bus architecture	15
Fig. 2.5	The basic, non-ideal circuit model of a solar cell	19
Fig. 2.6	Cell characteristics in sunlight and the maximum power point.....	22
Fig. 2.7	Solar cell I-V characteristics under full and partial irradianations.....	22
Fig. 2.8	Normalized solar cell output voltage and voltage characteristics under varying incident angles	22
Fig. 2.9	Solar cell I-V and P-V characteristics under different temperature..	25
Fig. 2.10	Basic electrical model of battery	28
Fig. 2.11	Battery discharge curves at constant temperature, different discharge ratings	30
Fig. 2.12	Battery discharge curves at different temperatures, constant discharge rating	30
Fig. 2.13	Battery charge characteristics at different charge ratings under constant temperature	31
Fig. 2.14	Battery charge (Buck) converter	33
Fig. 2.15	Inductor voltage and current over 3 periods	34
Fig. 3.1	X-SAT operation modes	38

Fig. 3.2	Power consumption at different modes in Normal Operation Phase	39
Fig. 3.3	Solar panel temperature histories in one orbit	40
Fig. 3.4	Solar array power curves at highest and lowest temperatures	41
Fig. 3.5	Satellite solar panel positioning under IRIS load	42
Fig. 3.6	Configuration of Power Control Module	49
Fig. 3.7	Fabricated power control module.....	50
Fig. 3.8	Battery latch-up problem during large load situation.....	52
Fig. 3.9	Battery latch-up problem during activated shunt regulator situation	53
Fig. 3.10	Block diagram of the power distribution module	56
Fig. 3.11	Fabricated power distribution module board 1 (PDM-1)	57
Fig. 3.12	Fabricated power distribution module board 2 (PDM-2)	57
Fig. 3.13	Fabricated ADCS interface module power (AIM-PWR) board	58
Fig. 3.14	Block diagram of the designed power system	58
Fig. 3.15	Experimental setup of the test system	59
Fig. 4.1	Sections through a lithium-ion battery	63
Fig. 4.2	X-SAT lithium-ion battery pack	64
Fig. 4.3	Electrical model for lithium-ion battery	65
Fig. 4.4	Battery discharge profile	67
Fig. 4.5	Battery voltage responses of 13.5A, 6.75A, 4.5A and 1.35A pulse discharging	67
Fig. 4.6	Battery voltage response at 60% SOC and C/2 (6/75A) current rate	69
Fig. 4.7	Battery internal resistances at different SOC during charge/discharge	71

Fig. 4.8	Battery voltage curves under 13.5A, 6.75A, 4.5A and 1.35A constant current discharging	72
Fig. 4.9	Battery discharge rate factors, with reference to C/3(4.5A) discharging current	73
Fig. 5.1	Block diagram of Power Supply System Electrical Ground Support System	77
Fig. 5.2	VI program flow chart	78
Fig. 5.3	Solar cell model	80
Fig. 5.4	Power consumption under idle and IRIS load	84
Fig. 5.5	Electrical ground support system instruments control interface	86
Fig. 5.6	Electrical ground support system PCM testing graphical interface .	87
Fig. 6.1	One orbit experiment results under IRIS load	89
Fig. 6.2	Power profiles of solar array, lithium ion battery and loads	90
Fig. 6.3	Equivalent PCM block diagram in eclipse stage	91
Fig. 6.4	Equivalent PCM block diagram in battery charging stage	92
Fig. 6.5	Equivalent PCM block diagram in battery discharging stage	92
Fig. 6.6	PCM input power, load power and system power efficiency	95
Fig. 6.7	Solar array output power under IRIS operating scenario and theoretical maximum power	96
Fig. 6.8	Solar array power, battery power and load power during LEOP phase	101
Fig. 6.9	Solar array power, battery power and load power under PPU scenario	102

Fig. 6.10	Battery voltage and current over one orbit period	103
Fig. 6.11	Experiment result showing battery latch-up phenomenon	104
Fig. 6.12	Estimated battery SOC under IRIS operating scenario for 5 orbits period	106
Fig. 6.13	Estimated battery SOC under PPU operating scenario for 5 orbits period	107
Fig. 6.14	Battery SOC under orbit number 1-4 of LEOP stag	108
Fig. 6.15	Battery SOC under orbit number 50-52 of LEOP stage	108
Fig. 6.16	Block diagram of the operation scenario test set-up	111
Fig. 6.17	Programmed and actual thermal cycle profiles	112
Fig. 6.18	Satellite load power profile under ‘eclipse data download’ scenario	113
Fig. 6.19	Load power profiles of IRIS and PPU operation scenario	115
Fig. 6.20	PCM voltages during 3 cycles of operation scenario tests	118
Fig. 6.21	PCM currents under 3 cycles of operation scenario tests	118
Fig. 6.22	Temperature profile under 3 cycles of operation scenario tests	119
Fig. 6.23	PDM DC-DC converters temperature profiles	124
Fig. 6.24	PDM switching MOSFETs and sensing resistors temperature profiles	125

List of Tables

Table 2.1 Comparison of power system architectures 18

Table 2.2 The Kelly cosine values of the photocurrent in silicon cells 24

Table 2.3 Temperature coefficients of a typical multi-junction
InGaP/InGaAs/Ge solar cell 24

Table 2.4 The radiation influence on the performance of solar array 26

Table 3.1 Characteristics of chosen solar cell 45

Table 3.2 Energy consumptions during eclipse period 46

Table 4.1 Energy and power densities of various battery cells 63

Table 4.2 Battery internal resistances at different charge/discharge rates and
SOC 70

Table 6.1 Power consumption for various operating scenarios 97

Table 6.2 Summary of the power supply system whole orbit tests results 110

Table 6.3 Summary of PCM board temperature profiles 120

Table 6.4 Summary of PDM board temperature profiles 123

Chapter 1

Introduction

1.1 Motivation and Overview

Micro-satellites are drawing increase interests of many research institutes and companies. The fast and cheap development of micro-satellites makes them a suitable platform for technology evaluation. They usually operate at Low Earth Orbit (LEO) of 200km to 2000km and weight less than 100kg. Typically, they carry a variety of payloads to perform scientific and engineering experiments. Tens of micro-satellites have been launched in the recent years [1- 5]. The challenge of designing a micro-satellite is to maximize the capacity and performance while meeting the limited space and mass constraints.

Nanyang Technological University and DSO National Laboratories are collaborating to design a micro-satellite, X-SAT. It will be launched into a LEO at an orbit height of approximately 685 km. The satellite will have a regular eclipse season of 35 minutes with a 100 minutes orbit period. It is a multi-mission micro-satellite that will support 3 payloads, namely the IRIS Camera payload (also the primary load), the Parallel Processing Unit (PPU) payload, and a Global Positioning System Receiver (GPS). The orientation of the satellite is provided by the Attitude Determination and Control System

(ADCS) by means of interaction of 3-axis coils magnetic field with earth's magnetism. Supervision and command of all the subsystems is done by the On Board Data Handling Subsystem (OBDH). All the subsystems will be supplied with proper voltages by the Power Supply System (PSS). The PSS consists of solar array, secondary batteries, Power Control Module (PCM), and Power Distribution Module (PDM).

Being one of the most critical systems of the micro-satellite, the power supply system must be able to support all the mission operations throughout the designed mission life with minimum number of solar and battery cells. Power system always carries serious mass, size and cost issues to small satellites which are constrained in volume and mass. In this research, we focus on the power system topology design, integrated testing environment development, data analysis and performance evaluation of the X-SAT power supply system.

Over the years, different power system topologies have been developed [6-11]. These topologies can be classified into two categories, namely the Direct Energy Transfer (DET) topology and Maximum Power Point Tracking (MPPT) topology [12-17]. Conventional DET topology has a rather simple design but suffers the disadvantages of wide output voltage range and battery latch-up problem. The MPPT topology fully utilized the solar energy with a narrow output voltage range, but the design and construction is usually complex. Thus, heat dissipation, cost and mass issues have to be taken into consideration during the design process. The use of digital controls and embedded systems in MPPT is under research and development to solve some of these problems [15].

In the X-SAT power system design, the modified DET topology is adopted for its simple structure advantage that affects the mass and space. Different from the conventional DET sun-regulated topologies, the shunt regulator structure is eliminated to further reduce mass and prevent battery latch-up problem in cold temperature situations. Moreover, a main bus regulator has been introduced to ensure the main bus voltage meets the voltage range requirements. Both the main bus regulator and battery charge regulator are designed such that they work at 100% duty cycle to minimize the power loss and maximize the solar cell utilization factor.

To thoroughly evaluate the performance of the PSS, a test and measurement system has been developed. Different techniques under different software environments have been used to develop the power system model and simulate its performance [18-21]. These virtual-prototypes provide useful information for the hardware construction. However, the simulation results are not accurate enough for a complex and high cost satellite due to the assumptions made and software limitations. In this research, an integrated test system is developed using the Virtual Instrument (VI) concept. A VI is a computer based software driven instrument that interacts with external systems. VI has been used in various applications [22-29] such as measurement of flicker [22], testing of smart transducer interface [23], bearing condition monitoring [24] and intranet-controlled power monitoring [25] etc. All these reported results have demonstrated the usefulness of the VI system.

In this study, we have developed a VI based system that enables the PSS to be continuously tested in different combination of simulated satellite operating scenarios. The system interacts with instruments such as the solar array simulator and the electronic loads to simulate the solar panel performance and loading scenario on the ground. A novel solar array model that accounts for the solar cell degradation factor, solar incident angle and temperature effect has been proposed and implemented into the control algorithm. The system also performs data acquisition and data logging, which enables the evaluation of the PSS performance.

Among the performance parameters, the battery state of charge (SOC) is one of the most critical parameters. In the X-SAT design, the lithium-ion battery is chosen to be the secondary power supply as it provides a significant weight advantage over typical nickel-hydrogen batteries. Moreover, it has been evaluated and proven for space application [30-33]. Physical-base battery models have been built to study the internal dynamics [34-37]. However, these models are not suitable for system level design and testing process. Besides the physical models, both analytical and electrical models for lithium-ion battery have also been developed over the past few years [38-41]. The analytical model, which predicts the remaining capacity of lithium-ion battery based on the current and voltage measurements, was presented in [38]. This method requires a period of constant current discharging before the estimation for a variable load, which is not applicable to a satellite such as X-SAT whose currents are varying. A dynamic electrical model has been developed and verified in [39]. This model could account for the charging/discharging rate factors and temperature dependencies. However, the internal resistances of the

battery need to be manually tuned. Thus, errors are generated during the first few cycles. Another electrical battery model was first introduced in [40] for NiMH battery. This model has been proven to work well for the lithium-ion battery [41]. From this model, the battery internal resistances and capacitances can be estimated accurately.

In this research, a combined electrical battery model has been developed based on two battery models that have been introduced. With the proposed model, the battery internal parameters at different current ratings and its remaining capacities can be estimated. Moreover, the battery state of charge can be calculated simply based on the output voltage and current in real-time without a long resting period. The experimental results have validated that the model is quite accurate as compared with the manufacturer's data.

1.2 Objectives

1. To design a power control module of power supply system for X-SAT that meets the proposed X-SAT mission operation requirements, under the constraints of mass, area and the payload. The module is used to generate, store and regulate the power to power distribution module, which distributes power to all the other satellite subsystems.
2. To develop an Electrical Ground Support System (EGSS) for the PSS. This system is an integrated testing platform. It is designed to simulate the satellite operation scenarios on ground and to monitor the PSS state of health in real-time. Proper models of solar array and electronic loads need to be proposed. The

proposed models are to be implemented into the instruments control algorithm. Graphical user interface will be designed for control, monitoring and historical data visualization purpose.

3. To develop method for lithium-ion battery State Of Charge (SOC) estimation. The method is to be integrated into the satellite EGSS to estimate the battery SOC in real-time scenario tests. This is motivated by the system performance evaluation purpose. Battery state of charge is one of the most critical parameters in the evaluation process. Lower battery SOC results in shorter satellite mission life.
4. To analyze the experiment data and evaluate the PSS performance with respect to satellite mission operations criteria. The designed PSS characteristics are to be confirmed and verified.

1.3 Main Contributions

The major contributions of this thesis are to be summarized as follows:

- A new power supply system topology for a micro-satellite has been designed. The design takes advantage of the convention DET sun-regulated topology with the shunt regulator being eliminated. This further improves the mass deduction and elimination of the battery latch-up problem in cold temperature. Experimental and simulation results have shown that the system meets all the satellite mission operation requirements.

- An integrated PSS test system has been developed using virtual instrument environment. A novel solar cell model has been developed to define the solar cell parameters in space environment. The model enables the test system to generate solar power for mission operations on ground. Experimental results show that the system provides useful information and is a valuable tool for the satellite testing and development.
- The conventional lithium-ion battery SOC methods require long battery resting period, or need manually tuning of internal parameters. These are not acceptable for a satellite testing and evaluation system. In this research, a new battery model is proposed to overcome the limitations and enables the battery SOC to be estimated in real-time. The analysis based on the experimental results shows that the proposed model provides accurate information as compared to the manufacturer data.

1.4 Organization of the Thesis

The rest of the thesis is organized as follows:

Chapter 2 reviews related literature on satellite power supply systems, including solar array, rechargeable battery, voltage regulators, as well as different system topologies. Solar cell and rechargeable battery characteristics are studied from experiments. Practical design considerations are discussed based on the characteristics. Pros and cons of different topologies are also compared.

Chapter 3 discusses the design consideration and construction of X-SAT power supply system architecture. From the proposed architecture, the solar array and battery capacity are optimally sized.

Chapter 4 concentrates on the study and modeling of lithium-ion battery. The X-SAT lithium-ion batteries internal parameters at different current rating and remaining capacity are determined.

Chapter 5 describes the design concept of a virtual instrument for testing the PSS. In the virtual instrument environment, the solar cell and electronic load models are proposed. The models estimate the solar array output using temperature and incident angle factors. Then the virtual instrument controls the solar array simulator and the electronic load based on the parameters estimated.

Chapter 6 presents the test results of X-SAT power supply system. In this chapter, we evaluate the performance of power supply system as well as the lithium-ion battery state of charge. From the test results, the power supply system characteristics during whole orbit operation scenario as well as thermal cycling tests are discussed.

Finally, the conclusions are presented in Chapter 7. Possible plans and directions for future research work are also included.

Chapter 2

A review of satellite power supply system

The satellite power system is designed to generate, store, regulate and distribute power required by the various subsystems of the satellite. In modern systems, photovoltaic (PV) cells and secondary batteries are commonly used as the power source in the satellite [11]. This chapter reviews the solar array, the secondary batteries and the commonly used satellite power system architectures.

2.1 Power system architectures

Typically, the power system for a satellite consists of solar array, rechargeable battery and power regulators. Various power supply system topologies for satellites have been developed over the years. These systems are designed to support payloads power ranging from several watts to several kilo-watts [3-10, 64-69]. Among these systems, conventional power system architectures for LEO satellite can be classified into the following categories:

1. Direct Energy Transfer (DET)
2. Maximum Power Point Tracker (MPPT)

In the DET approach, the power from the solar array is transferred directly to the main bus without passing through any series component. In the MPPT architecture, a series element in the form of maximum power point tracker is present. The DET un-regulated

bus, DET sun-regulated bus, DET full regulated bus and MPPT topology will be discussed in the following sections.

a. DET un-regulated bus

The DET un-regulated bus is developed to minimize complexity when all the loads are able to tolerate large main bus voltage variations [11]. Fig. 2.1 depicts the DET un-regulated bus architecture. As shown in the figure, the load is connected directly to the solar array output. The solar array charges the battery through a battery charge regulator.

Fig. 2.1 DET un-regulated bus architecture.

The advantage of DET un-regulated bus is its simplicity. However, this configuration also has several disadvantages:

1. Wide output voltage range

As the system has no component to regulate the solar array and battery voltage, the main bus voltage can range from the minimum battery voltage to the open circuit voltage of a cold solar array. For X-SAT, this voltage range is from 21.8 Volts to 60 Volts. This can be a problem if the subsystems can not tolerate such a wide voltage range.

2. Dynamic loading problem

In sunlight, the stable operating point of the solar array is at the descending side of the power curve [11]. During dynamic loading when the load power exceeds the maximum power that the solar array can generate, the solar array operating point is clamped to the battery output voltage minus the diode voltage drop. For X-SAT, this voltage is in the range of 21.8 Volts – 28.7 Volts. If such high power demand condition persists, the battery discharge will lead to a drop in the main bus voltage. Thus, the power output of the solar array also decreases. Hence, a margin has to be given for such instances when calculating the power budget.

3. Battery latch-up problem

The third problem of this configuration persists during the transition from eclipse to sunlight. The solar panel is designed such that its maximum power point voltage is always higher than the battery voltage in order to fully charge the battery. Thus, both the solar array bus and the main bus voltage are locked at the increasing side of the power curve during this period of time. Unless this operation is somehow perturbed, the solar array will not be able to operate beyond the full charged battery voltage. The detailed reason and solutions of this problem will be discussed in Chapter 3.3.

b. DET sun-regulated bus

The block diagram of DET sun-regulated bus is shown in Fig. 2.2. As shown in the figure, the shunt regulators are introduced in this architecture to regulate the main bus voltage

during sunlight. Same as the un-regulated bus, the battery still discharges through a simple diode. The battery discharge diode only allows discharge current from the battery. It blocks any uncontrolled charge current coming directly to the battery, leaving the charging function to the charge regulator. Therefore, the main bus voltage falls as the battery discharges during eclipse, and rises as it gets recharged during sunlight. The main bus voltage variations are the same as the battery voltage. When the solar array generates more power than the load and the battery charge power requirements, the extra power are shunted by the shunt regulators. The main bus voltage drops as the solar array power reduces. The accuracy of the regulated voltage is dependent on the number of shunts. Various models have been proposed to study the performances and characteristics of shunt regulators [42-46]. In general, more shunt regulators produce less variation of the main bus voltage.

Fig. 2.2 DET sun-regulated bus architecture.

This configuration of sun-regulated bus is still relatively simple although it is more complex than the un-regulated bus. The problem of large main bus voltage range exists in the un-regulated bus can be solved with the introduction of shunt regulators. Narrower operating range during sunlight can be achieved by introducing more shunts. However, problems that associated with the dynamic loading during sunlight and the battery latch-

up problem remain present in the sun-regulated configuration. Moreover, when the solar array is cold and the solar array output voltage is high, it is possible that all the shunts are shunted to regulate the main bus voltage. During this stage, there will be no power generated. This will greatly reduce the power efficiency and increase the battery depth of discharge during cold array as well as launch and early operation stage.

c. DET fully regulated bus

In a fully regulated energy transfer bus, commonly known as the regulated bus, the main bus voltage is controlled within a few percent during the entire orbit period. The typical main bus voltage is ± 2 to 5% of the nominal voltage. The architecture of this system is shown in Fig. 2.3. It includes shunt regulator, battery charge regulator and battery discharge regulator (BDR).

Fig. 2.3 DET fully regulated bus architecture.

The battery charge and discharge regulators provide interfaces between the main bus and the battery. The discharge regulator boosts the battery voltage to the main bus voltage during the eclipse and the charge regulator bucks the solar array voltage to the battery voltage during sunlight. The only difference between the sun-regulated bus and fully

regulated bus is the presence of a battery discharge regulator. The sun-regulated bus replaces the battery discharge regulator with a diode.

In normal operation, the DET fully regulated bus maintains the main bus voltage between the specified upper and lower limits. It can operate in the following three modes:

1. Shunt mode: In this mode, the batteries are fully charged and the generated solar array power exceeds the satellite load power. Thus, the excess power from the solar array is shunted through the shunt regulators.
2. Battery charging mode: In this mode, the batteries are being charged. Besides supplying power to all the mission payloads, the solar array charges the battery in constant current constant voltage mode. The battery charging voltage is regulated by the battery charge regulator.
3. Battery discharging mode: this mode is operating throughout the whole eclipse and part of the sunlight when solar array power is not adequate for the loads. The batteries are discharging to provide the power needed by the load. The main bus voltage is regulated by the battery discharge regulator.

When the battery is in the discharging mode, the battery is effectively decoupled from the main bus. This solves the problems associated with battery latch-up and dynamic loading as the voltage is not clamped during the battery discharge. This also reduces the voltage range of the main bus as it does not have to be the same as the battery voltage as it

discharges. However, the problem of all shunts activated during cold array that reduces the power generation is still present. Moreover, a large number of shunts are needed if tight main bus voltage control is desired.

d. Maximum power point tracker (MPPT)

The solar array generates more power at higher voltage at the beginning of life and when it just comes out from the eclipse. However, the load power must be supplied at the same voltage, generally lower than the maximum power point voltage of the solar array. A suitable switching regulator between the solar array and the load, remedies the disparity between the maximum power point voltage and the load supply voltage. The series regulator input voltage is then maintained at the maximum power producing level by the maximum power point tracker, and the output voltage is stepped down to the constant load voltage.

Fig. 2.4 MPPT regulated bus architecture.

The conventional architecture of a MPPT regulated bus is shown in Fig. 2.4. The maximum power point tracker fully utilizes the solar array by maintaining the operating point of the solar array at its maximum power point. The main bus regulator regulates the solar array output voltage. The battery discharge regulator boosts the battery voltage to a

regulated main bus voltage level. This configuration gives the optimized power output from the solar array irrespective of the temperature fluctuation. The main bus voltage is also being regulated at a required voltage above the fully charged battery voltage. However, the system requires additional electronic circuits to implement the MPPT. There are also additional power losses incurred by the MPPT and the regulators.

e. Comparison of the different topologies

Based on the discussions of the various power supply system topologies, a comparative study is made and shown in Table 2.1. For selecting a particular power supply system topology, there is a trade-off between simplicity/cost and optimized use of power. In a LEO satellite like the X-SAT, the dynamic solar array operating range is large due to the temperature variation. Furthermore, it has to meet the dynamic loading during sunlight. The solar array power should be fully utilized due to the limited area of solar panels and cost. Moreover, the voltage range of the main bus has to meet the input requirements of all subsystems as well. The characteristics of the solar array, battery and the power converters will be studied in the following sections to further understand the differences between these power system architectures.

	DET un-regulated bus	DET sun-regulated bus	DET fully regulated bus	MPPT
Voltage range	22V - 48V	22V – upper threshold voltage (Upper limit depends on the number of shunts)	30V – upper threshold voltage (Upper limit depends on the number of shunts)	Slightly higher than the battery fully charged voltage, fully regulated
pros	<ul style="list-style-type: none">• Simple implementation;• Low cost.	<ul style="list-style-type: none">• Simple implementation;• Reduced voltage range.	<ul style="list-style-type: none">• Fully regulated main bus voltage;• No battery lock-up problem due to the use of BDR.	<ul style="list-style-type: none">• Optimized solar power output can be obtained irrespective of array temperature;• Regulated operating voltage.
cons	<ul style="list-style-type: none">• Wide operating voltage;• Potential battery lock-up problem after eclipse.• During high power	<ul style="list-style-type: none">• All shunts activate to regulate the voltage just after eclipse when array is cold resulting in no power generation;• Narrow operating voltage range achievable only with large	<ul style="list-style-type: none">• Complex implementation;• All shunts activate to regulate the voltage just after eclipse when array is cold resulting in no power generation;	<ul style="list-style-type: none">• Regulator and MPPT power losses resulting in low efficiency than DET in many cases;• More heat dissipation inside the satellite body;

	<p>dynamic operation in sunlight, efficiency of solar array drops dramatically as solar array bus voltage follows battery discharge voltage.</p>	<p>number of shunts and control circuits;</p> <ul style="list-style-type: none">• Battery latch-up concern;• During high power dynamic operation in sunlight, efficiency of solar array drops dramatically as solar array bus voltage follows battery discharge voltage.	<ul style="list-style-type: none">• Narrow operating range achievable only with large number of shunts and control circuits;• BDR increases complexity and decreases efficiency.	<ul style="list-style-type: none">• Complex implementation.
<p>Best application in mission with these features</p>	<ul style="list-style-type: none">• Small load variations;• Small variations in illumination for most of sun period.	<ul style="list-style-type: none">• Small load variations;• Small variations in illumination for most of sun period.	<ul style="list-style-type: none">• Loads requiring close regulation;• Large solar array output voltage variations.	<ul style="list-style-type: none">• Loads requiring close regulation;• Large variation in solar array input energy throughout the mission.

Table 2.1 Comparison of power system architectures.

2.2 Solar array

The solar array is made of a number of photovoltaic (PV) cells connected in a series-parallel combination to obtain the required voltage and current. A photovoltaic cell converts sunlight into direct current electricity. In terms of the conversion efficiency, there are several types of PV technologies available such as single-crystal silicon, gallium arsenide and thin film [53-57]. Among them, silicon is the most commonly used element in the fabrication of solar cells. To evaluate the performance of the solar cells, various models of solar cells have been proposed [58-60]. These models describe varying electrical and physical characteristics of the solar cells such as the AC parameters and the series resistance.

a. General Characteristics

Fig. 2.5 The basic, non-ideal circuit model of a solar cell.

Among the various functional solar cell models, the most practical and commonly used model is shown in Fig. 2.5. As shown in the figure, the series and parallel resistances R_{sh} and R_s are the internal resistances that limit the output of the cell due to the manufacturing method and type of materials used. Generally, the internal resistance R_{sh} is

much greater than the external load resistance. Thus, most of the available current is delivered to the load. In addition, the internal resistance R_s is usually much less than the external resistance and the internal power dissipation within the cell is small.

Using Kirchhoff's current law, the load current I can be expressed as:

$$I = I_{ph} - I_D - \frac{V + R_s I}{R_{sh}} \quad (2.1)$$

where I_D is the diode current, I_{ph} is the photocurrent and V is the applied voltage.

Since R_{sh} is much larger than R_s , (2.1) can be simplified as

$$I = I_{ph} - I_D \quad (2.2)$$

From the diode equation, the diode current I_D can be expressed as:

$$I_D = I_{rs} \left[\exp \left(\frac{q(V + R_s I)}{kTA} \right) - 1 \right] \quad (2.3)$$

where I_{rs} is the cell reverse saturation current, A is an empirical constant depending upon the solar cell material, q is the electronic charge, k is Boltzmann's constant and T is the junction temperature.

The solar cell I-V characteristics equation can be expressed by substituting (2.3) into (2.2),

$$I = I_{ph} - I_{rs} \left[\exp \left(\frac{q(V + R_s I)}{kTA} \right) - 1 \right] \quad (2.4)$$

From (2.4), it is observed that if the solar cell is shorted, i.e. $V=0$, the solar cell output current I equals the photocurrent I_{ph} .

b. I-V and P-V characteristics

The electrical characteristic of a PV cell for a given irradiation and temperature in sunlight is represented by the I-V curve as shown in Fig. 2.6 [11]. In the figure, the four most important parameters for describing the cell electrical performance are the open circuit voltage V_{oc} , the short circuit current I_{sc} , the maximum power point voltage V_{mp} , and the maximum power point current I_{mp} . The short circuit current can be measured by shorting the output terminals of the cell, and measuring the terminal current under full illumination. If we neglect the diode and the ground leakage currents under zero terminal voltage, the short circuit current I_{sc} is the photocurrent I_{ph} . The current under this condition is the maximum current the cell can deliver. I_{mp} and V_{mp} are the current and voltage generated by the solar array when it is operating at the Maximum Power Point (MPP). This is the point P of Fig. 2.6 (a). This diagram shows the solar cell I-V characteristics. The corresponding cell power curve is shown in Fig. 2.6 (b). When a solar cell is connected to a constant resistance load, the operating point of the solar cell resides at the intersection point of the constant resistance load line and the I-V curve. As shown in Fig. 2.6, when the load resistance increases, the load line shifts from line 1 to line 2. Consequently, the solar cell operating point moves from point P to a higher voltage point Q . Moreover, it is observed from Fig. 2.6 (b) that the output power of the solar cell decreases. The satellite power system should be designed such that the solar cell operating point is close to its maximum power point to fully utilize the solar energy.

Fig. 2.6 Cell characteristics in sunlight and the maximum power point --- left: cell output current vs. cell output voltage; right: cell power vs. cell output voltage.

Figure 2.7 Solar cell I-V characteristics under full and partial irradiancies.

Fig. 2.8 Normalized solar cell output voltage and voltage characteristics under varying incident angles

The general characteristics of the solar cell under full and partial sun irradiations are shown in Fig. 2.7 [11]. As shown in the figure, the solar array output current is directly proportional to the irradiation level of the sunlight. The output of the solar cell is also influenced by the incident angle. Fig. 2.8 shows the effect of the incident angle on the solar cell voltage and current output characteristics specified by the manufacturer. From the figure, it is observed that the solar cell output current is greatly influenced by the incident angle. Generally, when the incident angle changes, the solar cell short circuit and the open circuit voltage can be expressed as [11]:

$$V_{oc(\theta)} \approx V_{oc(\theta=0)} \quad (2.5)$$

$$I_{sc(\theta)} = I_{sc(\theta=0)} \cdot \cos \theta \quad (2.6)$$

where θ is the incident angle between sunlight and the normal of solar panel and $I_{sc(\theta=0)}$ is the photocurrent when $\theta=0^\circ$. It is observed from (2.5) that the short circuit current I_{sc} follows cosine law as the incident angle changes. However, the actual power-angle curve of the PV cell is called Kelly cosine [11], which is given in Table 2.2. From Table 2.2, it is observed that the cosine law describes the solar cell characteristics accurately for sun angles ranging from 0° to 50° .

With increasing temperature, the short circuit current of the cell increases, whereas the open circuit voltage decreases. Table 2.3 shows the temperature coefficients of a typical multi-junction InGaP/InGaAs/Ge solar cell. From the table, it is observed that the voltage temperature coefficients are about 1000 times larger than the current temperature coefficients. Thus, the increase in current is much less than the decrease in voltage. Consequently, the net effect is a decrease in power when the temperature increases.

Sun angle (degrees)	Mathematical cosine value	Kelly cosine value
30	0.866	0.866
50	0.643	0.635
60	0.500	0.450
80	0.174	0.100
85	0.087	0

Table 2.2 The Kelly cosine values of the photocurrent in silicon cells.

Temperature coefficients			
$I_{sc} (\mu A/^{\circ}C)$	$V_{oc} (mV/^{\circ}C)$	$I_{mp} (\mu A/^{\circ}C)$	$V_{mp} (mV/^{\circ}C)$
5.3	-5.9	1.2	-6.5

Table 2.3 Temperature coefficients of a typical multi-junction InGaP/InGaAs/Ge solar cell.

The temperature effects on solar cell current and voltage are quantitatively evaluated in the following section. Suppose I_{mp} and V_{mp} are the maximum power point current and voltage at reference temperature T . Let α and β be the temperature coefficients in $A/^{\circ}C$ and $V/^{\circ}C$, respectively. If the operating temperature is increased by ΔT , then the new current and voltage is given by

$$I_{mp(T+\Delta T)} = I_{mp(T)} + \alpha \cdot \Delta T \tag{2.7}$$

$$V_{mp(T+\Delta T)} = V_{mp(T)} - \beta \cdot \Delta T \tag{2.8}$$

Thus, the maximum solar power at the new temperature is

$$P_{mp(T+\Delta T)} = V_{mp(T+\Delta T)} I_{mp(T+\Delta T)} = (I_{mp(T)} + \alpha \cdot \Delta T) \cdot (V_{mp(T)} - \beta \cdot \Delta T) \quad (2.9)$$

By neglecting the term containing the products of α and β in (2.9), we have the following expression,

$$P_{mp(T+\Delta T)} = V_{mp(T)} I_{mp(T)} + \alpha \Delta T V_{mp(T)} - \beta \Delta T I_{mp(T)} \quad (2.10)$$

Further simplification of (2.10) yields,

$$P_{max(T+\Delta T)} = P_{max(T)} - \left[(\beta I_{mp(T)} - \alpha V_{mp(T)}) \Delta T \right] \quad (2.11)$$

Based on the temperature coefficients given in Table 2.3, and (2.7)-(2.11), the solar cell I-V and P-V curves at different temperatures are determined and shown in Fig. 2.9. From the figures, it is observed that I_{sc} and I_{mp} are not sensitive to temperature variation. However, V_{oc} and V_{mp} are decreasing when the temperature is increasing. Thus, the maximum power generated by the solar cell becomes lower. As shown in Fig. 2.9 (b), the maximum power of each solar cell drops from 1.12W by approximately 25% to 0.84W when temperature increases from -50°C to 80 °C.

Fig. 2.9 Solar cell I-V and P-V characteristics under different temperatures.

The temperature effect of solar cells influences both the solar array sizing and the instrument modeling, which will be discussed later in Chapters 4 and 5.

c. Design and operational considerations

Photovoltaic cells are always combined into serial and parallel configurations to meet the power requirements. The voltage output of a string of cells is proportional to the number of cells in series. The current generated by a series of solar strings is proportional to the number of strings in parallel. The required number of series cells per string and the number of parallel strings for a satellite need to be calculated based on the required voltage, current and load power consumption.

Radiation Influence	Remaining Factor				
	I _{sc}	V _{oc}	I _{mp}	V _{mp}	P _{mp}
LEO	0.996	0.954	0.999	0.967	0.966

Table 2.4 The radiation influence on the performance of solar array..

At the satellite end of life (EOL), the solar array output voltage must be greater than the battery fully charged voltage. The energy that the solar array can generate must be greater than the sum of maximum loads power and the battery charge energy by a certain margin. The margin is to cover all the circuit boards power losses as well as possible failure of solar array string. Due to the orbital environment, the solar array power degrades over the satellite’s lifetime. Based on the manufacturer data, Table 2.4 shows the radiation influence on the performance of a solar array after 3 years of mission life. From the table,

it is observed that the maximum solar power drops by 3.4% after 3 years of LEO satellite mission. Since the solar array is to be sized to fulfill the EOL power requirements, the satellite system is designed to generate greater power at the beginning of life (BOL).

2.3 Lithium-ion Battery

For the satellite, the energy storage is required to meet the satellite load demand not only during the eclipse, but also when the power demand exceeds the solar power generation at any time. The most widely used energy storage technology is the battery. There are several types of rechargeable batteries such as Nickel Cadmium (NiCd), Nickel Hydrogen (NiH₂) and Lithium Ion (Li-ion). The electrochemical reaction of a rechargeable battery is reversible. It can be recharged from an external source after being discharged. The battery charge and discharge rates are specified in terms of the battery capacity, C . For example, the charging of a 100-Ah battery at 10-Amps rate is said to be charging at $C/10$ rate. Discharging this battery at $C/2$ rate means the drawing current is 50 Amps. At this rate, the battery will be fully discharged in 2 hours. The state of charge (SOC) of the battery at any instant is defined as

$$SOC = \frac{\text{Ah capacity remaining in the battery}}{\text{rated Ah capacity}} \quad (2.12)$$

The battery depth of discharge (DOD) is defined as

$$DOD = \frac{\text{Ah capacity drained from fully charged battery}}{\text{rated Ah capacity}} \quad (2.13)$$

From (2.12) and (2.13), it is observed that,

$$DOD = 1 - SOC \quad (2.14)$$

a. General Characteristics

All the rechargeable batteries including Li-ion battery have similar basic electrical models. They can be modeled as a voltage source with a small series internal resistance. Fig. 2.10 shows the basic electrical model of a battery. In the figure, E_i is the internal electrochemical voltage and R_i is the internal resistance. When the battery is in operation, the value of E_i decreases and R_i increases when the battery remaining capacity drops [11].

Fig. 2.10 Basic electrical model of battery.

From Fig. 2.10, the battery output voltage V can be obtained as,

$$V = E_i - IR_i \quad (2.15)$$

E_i can be expressed as a function of battery state of charge (SOC) [71]. From (2.15), it is observed that the values of E_i equals to the open circuit voltage V when $I = 0$. The internal resistance R_i is influenced by a number of factors including the battery SOC, current I and the temperature T . Thus, the voltage V is a nonlinear function of the battery SOC, discharge or charge rate I , and temperature T , which can be observed from the experimental results in the following section. The SOC is dependent on the battery rated capacity, which changes with the discharge rate and temperature also.

b. Charge/Discharge characteristics

The charge/discharge characteristics of the li-ion battery depend on the charge rate, discharge rate, temperature, and the age of the cell.

Two sets of experiments are conducted to study the X-SAT li-ion battery discharge characteristics. In the first set of experiment, the battery is discharged at different current rates under constant room temperature (20°C). The second set of experiments is conducted inside a thermal chamber. The battery is discharged at a constant current rating of 1.4 Amps under different temperatures. The constant current load is realized by using a HP E6050A electronic load. The battery output voltage is continuously monitored and logged during the experiments. The results are shown in Figs. 2.11 and 2.12. From Fig. 2.11, it is observed that the total capacity delivered by the battery decreases as discharge current rating increases. It also decreases with increasing battery temperature as can be observed from Fig. 2.12. In both of these figures, when the battery capacity is nearly drained, the voltage drops off rapidly towards the cutoff voltage of 2.5 Volts. At this endpoint, the battery capacity is depleted. Further operation beyond this point can cause the battery to reverse polarity and self-destruct.

Fig. 2.11 Battery discharge curves at constant temperature, different discharge ratings.

Fig. 2.12 Battery discharge curves at different temperatures, constant discharge rating.

Charging tests have also been conducted using the X-SAT li-ion battery. For this test, the battery is charged to its full charged voltage (4.2 Volts) using a power supply. Battery output voltage, current and the delivered capacity during the charging process are shown in Fig. 2.13. From the figure, it is observed that the charging current remains constant initially when the battery output voltage is lower than its full charged voltage. Once the battery reaches 4.2 Volts, its charging current starts to decrease until the battery is fully charged. Greater charging current rating results in shorter charging period. Similar to the battery during discharge, if the battery is overcharged beyond the full capacity, excess heating and internal gas buildup can destroy the battery.

Fig. 2.13 Battery charge characteristics at different charge ratings under constant temperature.

The battery charge and discharge curves as shown in Figs. 2.11 – 2.13 will be used for the battery internal parameters estimation in Chapter 4.

c. Design and operational considerations

Similar to the solar cells, the li-ion battery cells are also arranged in series and parallel combinations to provide the required output voltage and current. More battery cells in series results in higher output voltage. More battery strings in parallel gives higher capacity.

The li-ion battery has a limited number of charge cycles for its lifetime. The depth of discharge (DOD), the capacity drained before recharge, limits the number of possible charge cycles. In general, a greater DOD will result in a shorter battery lifetime [11]. For X-SAT battery to support the three years mission life, the battery DOD shall not exceed 25% throughout the satellite mission operations.

2.4 Power converters

The power converter converts one voltage level into another at the interface of two power components. In a typical satellite power system, the power converters are used in the battery charge regulator, battery discharge regulator and main bus voltage regulator systems. For the battery charge regulator, it uses the buck converter topology. For the battery discharge regulator, the boost converter topology is used. For the main bus voltage regulator, it can use either the buck or buck-boost topology depending on the

main bus voltage range requirements. The buck converter topology is described in the following section since only buck converter is used in the X-SAT power system design.

During sunlight, the main bus voltage needs to be regulated from the solar array output voltage which has a very wide voltage range. For X-SAT, the solar array output voltage ranges from 30 Volts to 60 Volts as the environment temperature changes. The battery is charged only if the battery voltage is lower than the solar array bus voltage. A DC-DC converter is required to step down the solar array bus voltage to the battery voltage during charging.

Fig. 2.14 Battery charge (Buck) converter.

Fig. 2.14 shows the circuit of the battery charge converter topology. When the converter is on, i.e., the switch is closed, the DC input source charges the capacitor and supplies power to the load via the inductor. Thus, the inductor current I_L increases linearly and the inductor voltage $V_L = V_{in} - V_{out}$. When the switch is open, the converter is switched off. Consequently, the power drawn from the DC source is zero. However, the load power is supplied by the energy stored in the inductor and capacitor, while the diode carrying the

return current. Since the inductor is discharging, its current flow decreases and the voltage $V_L = -V_{out}$. The voltage and current waveforms of the inductor over three complete cycles are displayed in Fig. 2.15. From the figure, it is observed that the converter works in continuous mode when the inductor current is always greater than 0.

Fig. 2.15 Inductor voltage and current over 3 periods.

Under the steady-state condition, the average voltage across the inductor equals zero.

Thus, we have the following expression:

$$(V_{in} - V_{out}) \cdot t_{on} = V_{out} \cdot t_{off} \quad (2.16)$$

From (2.16), the duty cycle D can be obtained as,

$$D = \frac{t_{on}}{T_s} = \frac{V_{out}}{V_{in}} \quad (2.17)$$

From (2.17), it is observed that we can vary the duty ratio to control the output voltage.

Typically, this is done in a feedback control loop with the required battery charge current

as the reference. The control loop output modulates the pulse width of t_{on} to vary the duty ratio.

The power efficiency of the voltage converters is calculated as follows. During the on time, the input voltage, less the voltage drop in the transistor switch, supply power to the

load. The energy transfer efficiency is therefore $\frac{(V_{in} - V_s)}{V_{in}}$, where V_{in} is the input voltage

and V_s is the voltage drop in the switch. During the off period, the inductor voltage, less the diode voltage drop, is the output voltage. The efficiency during this period is

therefore $\frac{V_{out}}{(V_{out} + V_d)}$, where V_{out} is the output voltage, and V_d is the voltage drop in the

diode. In addition, there is some losses in the inductor, the efficiency of which is denoted by η_i . The losses in the capacitor and the wires are relatively small and can be ignored for simplicity. The overall power efficiency of the buck converter is, therefore, the product of the above three efficiencies, namely

$$\eta_c = \left(\frac{V_{in} - V_s}{V_{in}} \right) \left(\frac{V_{out}}{V_{out} + (1 - D)V_d} \right) \eta_i \quad (2.18)$$

With commonly used devices, the transistor switch and the diode voltage drops are typically 0.6 Volts each. Thus, (2.18) can be expressed as

$$\eta_c = \left(\frac{V_{in} - 0.6}{V_{in}} \right) \left(\frac{V_{out}}{V_{out} + 0.6} \right) \eta_i \quad (2.19)$$

It is observed from (2.19) that the buck converter efficiency decreases with decreasing output voltage and/or increasing input voltage, as the switch and the diode voltage drops remain constant.

Since the buck converter in a LEO satellite handles high power and operates in long periods, they must be designed to achieve high efficiency to minimize the power loss as well as the heat generated. In X-SAT power system design, this factor has been taken into consideration and the detailed design principle will be explained in Chapter 3.3.

2.5 Conclusions

This chapter reviews the characteristics of the satellite power system modules namely the solar array, rechargeable battery and voltage converters. Characterization experiments have been conducted on the X-SAT batteries. Based on the characteristics of the module, the actual system design considerations have been discussed. Conventional power supply system topologies have been introduced. Advantages and disadvantages of existing power system topologies have also been compared.

Chapter 3

X-SAT power supply system

The characteristics and design considerations of the satellite power supply systems have been discussed in the previous Chapter. From the comparison of existing power system topologies, it is observed that both the DET fully regulated system and the MPPT systems carry mass and size issues which are critical for a micro-satellite. The main bus voltage range of DET un-regulated system is unacceptable for most of the X-SAT subsystems. For the DET sun-regulated system, it is able to meet the main bus voltage range requirements and its design is fairly simple. However, the associated battery latch-up problem greatly reduces the solar utilization factor. This is especially significant for a low earth orbit satellite such as X-SAT, as more than one third of the satellite operation is in eclipse. Moreover, the lowest temperature of the solar array can be as low as -50°C . The low temperature greatly increases the solar array output voltage. Thus, more shunts are activated and the satellite takes a longer period to get out of the battery latch-up problem. To overcome these issues, a new power system architecture is proposed in this chapter to increase the solar energy utilization factor during the cold solar array temperature periods. Moreover, the optimal sizing of the solar panel and lithium-ion battery based on the satellite mission operation conditions are also discussed in this chapter.

3.1 Satellite mission overview

There are several operation modes that the X-SAT can operate. The operation modes are the Launch and Early Operation Phase (LEOP), Normal Operation Phase (NOP) and Safehold Operation Phase. Each phase of operation shall be mutually exclusive and non-overlapping, that is, the satellite can only operate in one phase at any one time.

Fig. 3.1 X-SAT operation modes.

The system logic, with transitions between operation phases, is shown in Fig. 3.1. As shown in the figure, the very first operation phase that the X-SAT will enter is the LEOP. In this phase, the satellite will be tumbling with the spin rate determined by the momentum imparted by the separation system of the launch vehicle. The satellite shall carry out the de-tumbling operations first and then followed by the sun acquisition. After the satellite stabilizes and has the first successful contact with the ground station, it enters into the NOP. There are several operating modes in the satellite NOP. They include the IRIS operating mode, data download operating mode, parallel processing unit (PPU)

operating mode, and GPS operating mode. Different modes shall be carrying on at different time and position. The total power consumption of the satellite at different modes is shown in Fig. 3.2. In the figure, the satellite comes out from the eclipse at 1350 seconds. The sunlight period lasts for 65 minutes. At 5250 seconds, the satellite enters the eclipse again. From the figure, it is observed that the peak load power during sunlight is 207 watts under the IRIS operation. During the eclipse, the peak load power is 179 watts under the data download operation.

Fig. 3.2 Power consumption at different modes in Normal Operation Phase.

In the space, the satellite will experience large temperature variations. For the X-SAT, it has 35 minutes in eclipse for every orbit, which is 35% of the 100 minutes orbit time. The temperature variations are simulated by the colleagues of thermal group using SINDA/G

software. The simulation is conducted in a time intervals of 50 seconds at both extreme hot and cold cases based on the following assumptions:

1. The solar flux is normal to the solar panels.
2. For the hot case, the solar cells are assumed to be switched off. Thus, there is no power output from the solar array and all energy absorbed is converted into heat.
3. For the cold case, 26% efficiency of solar cells is assumed.

Fig. 3.3 Solar panel temperature histories in one orbit in time intervals of 50 seconds.

The simulated temperature profile is shown in Fig. 3.3. From the figure, we observed that the temperature varies from -50°C to 90°C in the hot case, and -60°C to 50°C in the cold case.

3.2 Sizing of solar panels and battery capacity

From (2.11) and the temperature coefficients given in Table 2.3, the X-SAT solar array output power curves at the highest and lowest temperatures for various numbers of solar strings are calculated and shown in Fig. 3.4. From the figure, it is observed that the peak power reduces from 325W to 210W for 18 solar strings when the temperature increases from -50°C to 90°C. Moreover, the open circuit voltage decreases from 48V to 33V.

Fig. 3.4 Solar array power curves at highest and lowest temperatures.

Both the solar array and rechargeable battery are sized based on the load power consumptions as well as the temperature profiles. The solar array shall generate sufficient power to meet the highest power requirement in all of the operations. In order to maintain orbital power balance, the power generated during the sunlight is used to sustain sunlight

operation as well as to replenish the battery energy used during the previous eclipse operation, i.e.

$$\begin{aligned} P_{solar-min} &= P_{average-sun} + P_{bat-charge} \\ &= P_{average-sun} + P_{average-eclipse} \cdot T_{eclipse} / T_{sunlight} \end{aligned} \quad (3.1)$$

where $P_{average-sun}$ and $P_{average-eclipse}$ are the average load power during sunlight and eclipse period respectively. $T_{eclipse}$ and $T_{sunlight}$ are the length of eclipse and sunlight period of X-SAT, which are 35 and 65 minutes respectively.

The minimum size of the solar array can be calculated from the minimum solar array power output required under all the operations. From (3.1) and Fig. 3.2, it is observed that the minimum solar array power required is 173.5 watts under IRIS load. Moreover, from the IRIS mission requirements, the satellite changes to nadir pointing 700 seconds before and after equatorial plane. Fig. 3.5 illustrates the X-SAT solar panel positioning under IRIS load over one orbit period.

Fig. 3.5 Satellite solar panel positioning under IRIS load.

As defined by the mission operation, the satellite starts the IRIS operation at 700 seconds before it enters the equatorial plane. The IRIS operation lasts for 1400 seconds. During this period of time, the solar panel is commanded to be nadir pointing instead of sun tracking. As shown in Fig. 3.5, the incident angle θ keeps changing and the power factor equals $\cos(\theta)$. Thus, the satellite sunlight average power factor S_{avg} over one orbit period during IRIS scenario can be calculated as follows:

$$\theta_1 = \theta_2 = \frac{\pi}{2} - \varphi = \frac{\pi}{2} - 2\pi \cdot \frac{700}{6000} = 0.2667\pi \quad (3.2)$$

$$S_{avg} = \frac{700 \cdot \int_0^{0.2667\pi} \cos\theta \cdot d\theta + 700 \cdot \int_{-0.2667\pi}^0 \cos\theta \cdot d\theta + (3900 - 1400)}{3900} = 0.908 \quad (3.3)$$

Moreover, the solar cells are not always operating at its maximum power point in DET system. The average power generated as a percentage of the peak power is defined as the solar energy utilization factor. Assume the solar energy utilization factor is 80%, which will be verified from the experimental results in chapter 6. Then the minimum power that the solar array is capable to generate at EOL can be calculated as:

$$\text{Minimum solar array output power} = \frac{173.5}{0.8 \times 0.908} = 238.90 \text{ watts} \quad (3.4)$$

From the solar array datasheet, the maximum power of each solar cell is 1.04 watts at 29.2°C. Moreover, the degradation factor is 0.966 after three years of mission life. The temperature factor of the cell is -2.37 mW/°C. Thus, the minimum number of solar cell can be calculated as:

$$\begin{aligned}
 \text{Minimum number of solar cell} &= \text{ceil} \left[\frac{238.90}{[1.04 - (90 - 29.2) \times 2.37 \times 10^{-3}] \times 0.966} \right] \\
 &= \text{ceil}(277.18) \\
 &= 278
 \end{aligned} \tag{3.5}$$

Thus, a total number of 278 solar cells are required to satisfy the load power requirement at end of life (EOL).

The minimum voltage produced by the solar array at the highest temperature and EOL should be at least equal to the battery full-charge voltage plus all the diode drops across the solar array bus. It is taken to be

$$29.4\text{V} + 0.8\text{V} = 30.2\text{V} \tag{3.6}$$

The characteristics of the solar cell are shown in Table 3.1. From the table, the maximum power point voltage V_{mp} of the solar cell is 2.3 Volts at 29.2°C. The temperature coefficient of V_{mp} is $-6.5 \text{ mV}/^\circ\text{C}$. Thus, the minimum number of solar cells in series is calculated as,

$$\begin{aligned}
 \text{Number of cells in series} &\geq \text{ceil} \left[\frac{30.2}{[2.3 - 6.5 \times 10^{-3} \times (90 - 29.2)] \times 0.967} \right] = \text{ceil}(15.98) = 16
 \end{aligned} \tag{3.7}$$

Hence, a minimum of 16 solar cells is required.

Maximum power point voltage @29.2°C (mV)	Open circuit voltage @29.2°C (mV)	V _{mp} Temperature coefficient (mV/ °C)	V _{mp} end-of-life remaining factor
2300	2593	-6.5	0.967

Table 3.1 Characteristics of chosen solar cell.

The minimum number of strings can be calculated by dividing the minimum number of cells by the number of cells in series. From (3.5) and (3.7), we have,

$$\text{Number of strings} \geq \text{ceil}\left(\frac{278}{16}\right) = \text{ceil}(17.375)=18 \tag{3.8}$$

Thus, the solar array is configured as 18 strings connecting in parallel with 16 solar cells in series of each string. The solar array output power can be calculated from its voltage and current output. As shown in Table 2.4, the solar array current and voltage degradation coefficients are 0.999 and 0.967 respectively after 3 years mission. Moreover, the voltage and current temperature factors can be observed from Table 2.3. Thus, the solar array output power at the highest temperature of 90°C and satellite EOL is calculated as,

$$\begin{aligned} P_{\text{Solar array}}^{90^{\circ}\text{C}} &= I_{\text{Solar array}}^{90^{\circ}\text{C}} \cdot V_{\text{Solar array}}^{90^{\circ}\text{C}} \\ &= 0.967 \times 16 \times \left[2.3 - (90 - 29.2) \times 6.5 \times 10^{-3} \right] \\ &\quad \times 0.999 \times 18 \times \left[0.455 + (90 - 29.2) \times 1.2 \times 10^{-6} \right] \\ &= 241.17 \text{ watts} \end{aligned} \tag{3.9}$$

To size the battery capacity, it is important to ensure the battery has enough capacity to sustain the highest power needed for eclipse operation. From the load power curves

shown in Fig. 3.2 and taking into account of power losses due to the regulators, the energy consumption for all the operating scenarios are calculated as shown in Table 3.2 .

	LEOP	Normal Operating Phase				
		Idle	IRIS	PPU	Data Download	GPS
Energy for eclipse (Wh)	42.3	48.1	49.4	50.8	59.48	48.9

Table 3.2 Energy consumptions during eclipse period.

From Table 3.2, it is observed that the maximum energy consumption during the eclipse period is 59.48 Wh during the data download operation. From the battery specifications, the maximum depth of discharge must not exceed 25% throughout the 3 years satellite mission operations. Thus, the minimum battery capacity at EOL is calculated as:

$$\frac{59.48}{0.25} = 237.92Wh \tag{3.10}$$

Thus, a lithium ion battery pack of 13.5 Ah capacity is chosen for X-SAT. At a nominal voltage of 25.2 Volts, the battery capacity at beginning of life is 340.2 Wh. The amount of battery capacity fade over life is dependent upon a number of parameters including depth of discharge (DOD), battery temperature and discharge rate. From the manufacturer data, the X-SAT battery experiences a worst case of 27.9% capacity fading after 3 years of mission cycling. Thus, the battery EOL total capacity is calculated as:

$$\text{Battery Capacity at EOL} = 340.2 \times (1 - 27.9\%) = 245.28Wh \tag{3.11}$$

3.3 Power supply system architecture design

Top level satellite design requirements can be derived by mission profile and general satellite configuration. The relevant criteria for power supply system design are first discussed in this section, and then followed by the system architecture design.

In the LEOP, the deployable solar panels are not deployed after successful separation of the satellite from the launch vehicle. The satellite is tumbling before it enters into the sun acquisition state. The spinning satellite will have two effects on the solar array output:

1. Since the satellite is tumbling, the solar panel is not normal to the sunlight. Thus, the solar array temperature will be lower than the normal operation mode. This will cause the solar array output voltage to be higher.
2. As discussed in Chapter 2, the solar array output current decreases in a cosine factor manner when the incident angle is increased. Thus, the output power of the solar array will be much lower than the normal operation phase.

Due to the high voltage and low power output nature of the solar array during LEOP, the proposed power system must be designed such that it is capable of regulating the high solar array voltage to the desired main bus voltage. It is also required to fully utilize the solar array output power such that the battery can be adequately charged to maintain the power balance.

During the eclipse period of normal operation, no power will be generated from the solar array. Thus, all the loads shall be supplied by the battery. As mentioned in Chapter 3.2, a maximum of 25% DOD is required to ensure the battery life is not compromised. During sunlight period of normal operation, the solar array output power must be able to supply all the loads as well as recharging the lithium-ion battery. Finally, solar cell degradation and battery capacity loss have to be taken into consideration. The criteria for the power supply system can be summarized as follows:

- Continuous operation of the satellite to carry on de-tumbling operation and first ground contact in LEOP;
- Continuous operation of all the payloads. The battery maximum DOD must be less than 25% during NOP;
- No impact of these mission criteria by aging and degradation effects.

Based on these criteria, the configuration of the power control module is proposed as shown in Fig. 3.6. The fabricated engineering model circuit board is shown in Fig. 3.7. As shown in Fig. 3.6, the design uses the direct energy transfer (DET) technique with a main bus regulator (MBR). As compared to the conventional DET approach as shown in Fig 2.2, the shunt regulators have been eliminated in this design. Thus, there will be no power shunted to the ground during the low temperature and high solar array output voltage situation. The battery charge regulator (BCR) is used to regulate the battery charging voltage, while the main bus voltage is regulated by the MBR. With this configuration, the solar energy can be fully utilized during the LEOP and when the satellite just comes out from the eclipse.

Fig. 3.6 Configuration of Power Control Module.

In this system, both the MBR and the BCR use the buck converter topology. To avoid the increasing loss due to decreasing input and output voltage(?), both the MBR and the BCR are designed such that they operate at 100% duty cycle when the input voltage is smaller than the designed threshold voltage. The threshold voltages of the main bus buck converter and the battery charge regulator are designed as 30.5 Volts and 29.4 Volts respectively. This is determined based on the requirements that the upper limit of the required main bus voltage is 30.5 Volts and the battery fully charged output voltage is 29.4 Volts. The operation can be expressed as:

$$\begin{cases} V_{out} = V_{reg} & \text{when } V_{in} \geq V_{reg} \\ V_{out} = V_{in} & \text{when } V_{in} < V_{reg} \end{cases} \quad (3.12)$$

where V_{reg} are 30.5 volts and 29.4 volts for the MBR and the BCR respectively. Thus, the buck converter functions as a simple diode to maximize the conversion efficiency when $V_{in} < V_{reg}$.

From (3.12), we calculate the minimum conversion efficiencies of the MBR and the BCR as:

$$\begin{cases} \eta_{MBR-\min} = \frac{30.5-0.6}{30.5} \cdot \frac{30.5}{30.5+0.6} \cdot \eta_i = 0.961 \cdot \eta_i \\ \eta_{BCR-\min} = \frac{29.4-0.6}{29.4} \cdot \frac{29.4}{29.4+0.6} \cdot \eta_i = 0.960 \cdot \eta_i \end{cases} \quad (3.13)$$

Fig. 3.7 Fabricated power control module.

Since the battery discharges power only through a discharge diode, the solar array bus voltage is the same as battery output voltage when the battery is discharging. Thus, the dynamic loading problem still exists in the designed system. This problem can be

observed in the results as shown later in Chapter 6. However, it still has the advantages over typical sun-regulated bus design using shunt regulators.

The issue of battery latch-up problem has been reported in [73]. As discussed in [11], the bus remains latched to the battery voltage during the battery latching-up. Moreover, the battery continues to discharge power instead of entering into charge mode although the solar array is capable of delivering full power. This problem is essentially caused by the discharging battery during sunlight period. The battery output voltage is designed to be smaller than the EOL solar array maximum power point voltage so that the battery can be fully charged during sunlight period. In a sun-regulated Direct Energy Transfer (DET) system, the solar array bus, the battery and the system main bus are simply connected through diodes when the battery is discharging. Thus, when the battery latch-up happens in a sun-regulated DET system, both the solar array output voltage and the main bus voltage will decrease to the battery output voltage. The battery latch-up problem happens in two situations:

- 1). The solar array latch-ups in battery discharge mode when its maximum power is insufficient to supply the load. This situation is illustrated in Fig. 3.8. As shown in the figure, the red line describes the constant load power, while the blue line shows the solar array V-I curve. Since the load power is larger than the maximum solar array power, the battery starts to discharge. When the battery discharges, the solar array output voltage decreases to the battery voltage of 22-29.2 Volts. Thus, the operating point of the solar array moves to point 2 from its maximum power point 1. The corresponding solar array current and the battery discharging current are indicated by I_{SA} and I_{Bat} respectively in Fig.

3.8. The solar energy utilization factor decreases since the solar array is not operating at its maximum power point. Removing the shunt regulators does not help to solve the latch-up problem under this situation. It can only be solved by sizing the solar panel to meet the EOL load requirements with sufficient margin.

Fig. 3.8 Battery latch-up problem during large load situation.

2). The battery latch-up problem can also happen when the satellite transits from eclipse to sunlight. Under this situation, the solar panel temperature is low and its output voltage increases as described in Chapter 2.2. Suppose the sun regulated power control module is implemented using 9 shunt regulator with each shunt regulator controlling two parallel solar strings. As shown in Fig. 3.9, the solar array’s open circuit voltage reaches 48 volts when the satellite just comes out from the eclipse. This is much higher than the main bus upper voltage limit (30.5 volts). Thus, some of the shunt regulators are activated to regulate the main bus voltage. As a result, the operating point of the solar array moves

from point 1 to 2 as shown in Fig. 3.9 with only 4 strings remain active. Consequently, the solar array current that supplies the system decreases linearly with the increase of the activated shunt regulators. As a result, all the solar power from the other strings are shunted and dissipated. At this instant, the solar array output voltage is 45 volts and is still higher than 30.5 volts. Thus, one more shunt regulator is activated and only 2 solar strings are connected to the power system.

Fig. 3.9 Battery latch-up problem during activated shunt regulator situation.

As shown in Fig.3.9, 2 strings of solar cell can not provide enough power to supply the load. Thus, the battery discharges and the operating point of the solar array stabilized at point 3. At point 3, the solar array only provides 25 watts of power to the load even though the maximum solar power is 360 watts. As a result of the battery latch-up, the solar energy utilization factor decreases and the battery maximum DoD increases.

By eliminating the shunt regulators in the power system, the solar power will not be shunted even when the solar array voltage is high. Thus, the solar array operates at point 1 as shown in Fig. 3.9. While the solar array output voltage is approximately 47 volts, the main bus voltage is regulated to 30.5 volts by the main bus regulator. The solar panel supplies sufficient power to the load as well as recharges the battery. Thus, the battery latch-up problem can be avoided during eclipse to sunlight transition period.

Avoiding the battery latch-up problem is extremely important for the X-SAT during the LEOP phase. During the de-tumbling stage, the satellite is tumbling and the solar panels are not deployed. Since the solar panels are not facing the sun at the normal angle for the whole sunlight period, its temperature will be much lower than the generated temperature profile as shown in Fig. 3.3. As described earlier, the low temperature environment will cause the battery to latch-up. Consequently, the battery can not be charged during the sunlight period throughout the de-tumbling stage. Consequently, the battery will be over-discharged before the first successful ground contact, which will cause series consequences to the satellite mission operations. By eliminating the shunt, the solar array will not be shunted when the satellite just come out from the eclipse. Consequently, all the available power could be used to supply the load and recharge the battery. This advantage can be seen clearly from the battery performance results in whole orbit tests, which is discussed in detail in Chapter 6.4.2.

From the discussions, it is clear that the battery latch-up will still happen under the first situation when the peak solar power is lower than the load power. This problem can be

avoided by sizing the solar array to meet the end of life load power with sufficient margin or adding more battery cells in series to increase the battery discharging voltage. However, both the methods require extra mass and cost which is critical for micro-satellite. Furthermore, adding more battery cells in series will also cause the main bus voltage to exceed the designed voltage range and the converters work inefficiently at low duty ratio. Thus, the battery latch-up problem during extremely high load power period is not considered. As will be shown from the results in Chapter 6, battery still discharges during sunlight period for a very short period of time only when the IRIS load is operating.

The resultant main bus voltage range is 22V – 30.5V, which is much narrower than the un-regulated bus voltage range (22V-54V). The power consumptions of both the converters are negligible when the input voltage is low. Thus, the power efficiency of the developed system is expected to be high. However, the system also locks the output voltage of the solar array to the battery voltage when the battery is discharging.

The regulated output voltage of PCM supplies power to the power distribution module (PDM). Subsequently, the PDM distributes power to all the payload systems according to their requirements. The PDM controls the power switch of all the payloads according to the mission operation scenario. Moreover, it provides over current protection of all the modules. The PDM contains 3 circuit boards, namely the power distribution module board 1 (PDM-1), power distribution module board 2 (PDM-2) and ADCS interface module power board (AIM-PWR). PDM-1 distributes power to modules such as the rate

sensor and parallel processing unit. PDM-2 supplies power to modules such as the on board computer (OBC) and X-band transmitter. AIM-PWR converts the 22V – 30V power from PDM-1 and supplies power to the ADCS module. The block diagram of the PDM is shown in Fig. 3.10. The fabricated PDM circuit boards are shown in Figs. 3.11 - 3.13.

Fig. 3.10 Block diagram of the power distribution module.

Fig. 3.11 Fabricated power distribution module board 1 (PDM-1).

Fig. 3.12 Fabricated power distribution module board 2 (PDM-2).

Fig. 3.13 Fabricated ADCS interface module power (AIM-PWR) board.

Fig. 3.14 Block diagram of designed power system.

Fig. 3.14 shows the block diagram of the completed power supply system including both the PCM and PDM. As shown in the figure, the regulated main bus voltage supplies power to the PDM. Subsequently, the PDM converts the main bus voltage to various voltage levels, such as +12 Volts, +5 Volts and +3.3 Volts, based on the payloads requirements.

The actual experiment set-up of the power system is shown in Fig. 3.15. From the figure, it is observed that the solar array is simulated by a solar array simulator. Payloads are

simulated using an electronic load. During power supply system testing, all the payloads are represented by resistive loads. The OBC commands are simulated using 5 Volts digital inputs to control the power switching of the modules. Moreover, all the telemetry signals are collected by the data acquisition system. The detailed testing set-up information will be given in Chapter 5.

Fig. 3.15 Experimental setup of the power supply system.

3.4 Conclusions

A new power system architecture for a micro-satellite has been proposed and developed. The new architecture adopts the direct energy transfer topology with the shunt regulators

being eliminated. A main bus regulator has been introduced to regulate the main bus voltage. Both the battery charge regulator and main bus regulator operate at 100% duty cycle when the input voltage is lower than the designed threshold voltage. Moreover, the new design eliminates the battery latch-up problem in cold temperature environment. Performance of the designed system will be evaluated in Chapter 6.

Chapter 4

Lithium-ion battery modeling and state of charge estimation

Lithium-ion battery has attracted more attentions for satellite applications due to the constant needs of reducing mass [30-32, 47-49]. Batteries that display higher gravimetric and volumetric energy densities, longer cycle life, wider operating temperature range and better discharge rate capability can all translate into increased payload and satellite capability. Both simulation models and analytical models of lithium-ion battery have been proposed to study its electrical performances [38-41, 50-52] including battery run time behavior [50], battery life estimation [51] and battery characterization [52]. Lithium-ion battery has been implemented into quite a number of satellites including the mars surveyor program [30].

The battery state of charge (SOC) is one of the most critical parameters that determine the satellite operation life. For X-SAT that has the designed three years mission life, the lithium-ion battery must be kept above 75% SOC. However, the battery SOC can not be measured directly. It is a complex function of battery material, charge/discharge rate, operating temperature as well as the battery service history. Several methods for estimating the lithium-ion battery SOC have been proposed. These methods can be classified into two categories. The first category is the analytical method. In [38], an

analytical model has been proposed to predict the lithium-ion battery remaining capacity with a support of system management bus for the battery. This method has simple analytical expressions and small number of parameters. However, this method may produce large errors in an environment where the battery works with a variable load. The second category is the electrical model [39-40]. In [39], an electrical battery model has been developed by introducing the battery charge/discharge rate factors. The rate factor describes the relationship between the battery capacity and the charge/discharge current rating. However, the battery electrical parameters need to be manually tuned during the simulation process in this model. This is not suitable in our automated system. Another electrical model was introduced in [40] to predict the battery internal parameters based on the battery SOC. In this chapter, we develop a combined lithium-ion battery model based on the model introduced in [39] and [40]. With the proposed model, the battery SOC can be estimated based on the battery output voltage and current values. Battery charge/discharge rate factor is considered during the SOC estimation. Moreover, the proposed model is suitable to be implemented into a fully automatic satellite testing system.

4.1 Basics of lithium-ion battery

Fig. 4.1 shows the sections of a lithium-ion battery. As shown in the figure, the battery is constructed by using a lithium oxide cathode and a carbon compound anode, with a polymer separator and a non-aqueous electrolyte between the poles [70]. Minute spaces are designed between the electrode materials to allow the lithium-ions to enter. Upon charging, the lithium ions are extracted from the positive material and inserted into the negative electrode material. Upon discharging, the reverse process is taking place. Hence,

the basic electrochemistry of the cell involves only the transfer of lithium ions between the two insertion electrodes. The energy and power densities of various battery cells are shown in Table 4.1 [11]. From the table, it is observed that the specific energy and energy density of the lithium ion battery is quite high as compared to other battery cells. Thus, the lithium-ion battery is a very favorable the secondary battery system for satellite.

Electro-chemistry	Specific Energy (Wh/kg)	Energy Density (Wh/l)	Specific Power (W/kg)	Power Density (W/l)
NiCd	40-50	50-100	150-200	300-500
NiH ₂	45-65	35-50	150-200	200-300
NiMH	50-70	140-180	150-200	300-500
Li-ion	90-150	150-250	200-220	400-500
Lithium-Polymer	100-200	150-300	>200	>400

Table 4.1 Energy and power densities of various battery cells.

Fig. 4.1 Sections through a lithium-ion battery.

Fig. 4.2 X-SAT lithium-ion battery pack.

Fig. 4.2 shows the picture of the X-SAT lithium-ion battery pack. The X-SAT battery consists of three units. Each unit has three parallel strings, and each string consists of seven battery cells connected in series. The overall battery capacity is 13.5 Ah (or 340 Wh at a nominal voltage of 25.2 Volts). The battery output voltage range is from 17.5 Volts to 29.4 Volts. However, the battery cut-off voltage is designed at 21.4 Volts to avoid over-discharge situation. According to the manufacturer data, the X-SAT lithium-ion battery has a life time of three years under the condition that its state of charge is kept above 75%.

The battery state of charge is a clear indicator of the energy balance of a satellite's power system. Thus, the estimation of the battery state of charge based on the data collected

such as the battery output voltage, current and temperature is essential to understand the battery system. A lithium-ion battery model will be proposed to estimate the battery SOC in real-time in the following section.

4.2 Proposed battery model

Fig. 4.3 Electrical model for lithium-ion battery.

In X-SAT, the lowest battery temperature is 3°C based on the analysis results from the thermal group. As the battery kinetic reactions become significant at low temperature environment. Thus, the kinetic resistance must be included in the battery model. In this research, the electrical model of the lithium-ion battery shown in Fig. 4.3 is used. A test scheme is developed to estimate the real-time battery state of charge (SOC). In Fig. 4.3, I_O and V_O denotes the battery output voltage and current; V_{OC} denotes the battery open circuit voltage; R_i is the constant part of battery internal resistance; R_D and C_D describe the effect of diffusion process; while R_K and C_K represent the kinetic reaction. The battery internal resistance consists of three components R_i , R_D and R_K . Moreover, the capacitances C_D and C_K characterize the transient response of the battery.

From Fig. 4.3, the relationship between battery open-circuit voltage and terminal voltage can be described based on Kirchhoff's voltage law:

$$V_{OC} = V_O + I_O \cdot (R_i + R_D + R_K) - I_O \cdot R_D \cdot \exp(-t/\tau_D) - I_O \cdot R_K \cdot \exp(-t/\tau_K) \quad (4.1)$$

where $\tau_D = R_D \cdot C_D$, $\tau_K = R_K \cdot C_K$.

To determine the battery model parameters including R_i , R_D , R_K and charge/discharge rate factors, a series of battery pulse charging/discharging experiments at different battery SOC have been conducted. The battery charging/discharging profile is shown in Fig. 4.4. The battery is charged with constant current initially till the trickle current reaches C/50 rating, which is 0.27A. The battery SOC is considered to be 100% at this time instant. Subsequently, the battery is continuously discharged at 6.75A (C/2) till the system consumes 20% of the battery full capacity. This takes about 1440 seconds. This is followed by 1800 seconds rest period. At the end of every rest periods when the battery EMF is stabilized, which are points 1-5 as shown in Fig. 4.4, the pulse charging and discharging experiments are conducted.

In this study, the charging/discharging current is a 5 seconds duration pulse which is generated by programming a HP 6050A electronic load that has a slew rate of 5 MA/s. The values of the current sets are 13.5A (1C), 6.75A (C/2), 4.5A (C/3) and 1.35A (C/10). The rest period between current pulses is 30 seconds each.

Fig. 4.4 Battery charging/discharging profile.

Fig. 4.5 Battery voltage responses of 13.5A, 6.75A, 4.5A and 1.35A pulse discharging.

Fig. 4.5 shows a particular case with the battery discharging at 60% SOC. The battery voltage responses during the impulse discharging can be divided into three periods. The first period is the initial voltage drop period, which is from t_0^- to t_0^+ . The following 1 second period from t_0^+ is termed as the short-term transient period; while the remaining 4 seconds period is considered as the long-term transient period. The voltage drop during the first period is $V_i = I_0 \cdot R_i$. For the short and long term transient periods, the voltage responses are expressed as $V = a \cdot \exp(-(t - t_0)/b)$, where $a = I_0 \cdot R_D$, $b = \tau_D$ in short-term transient, and $a = I_0 \cdot R_K$, $b = \tau_K$ in the long-term transient. From these expressions, the parameters R_i , R_D , C_D , C_K can be determined. Taking the voltage response at 60% SOC and C/2 (6.75 Amps) discharging rate as example, which is shown in Fig. 4.6, we observed that during short term transient (45 – 46 second), $V_R = 0.79V$, $V_D = 0.04V$ and $\tau_D = 0.38s$.

From (4.1), the parameters are calculated as:

$$R_i = \frac{V_R}{I} = \frac{0.79V}{6.75A} = 0.117\Omega \quad (4.2)$$

$$R_D = \frac{V_D}{I} = \frac{0.04V}{6.75A} = 5.93m\Omega \quad (4.3)$$

$$C_D = \frac{\tau_D}{R_D} = \frac{0.38s}{5.93m\Omega} = 64.13F \quad (4.4)$$

Fig. 4.6 Battery voltage response at 60% SOC and C/2 (6/75A) current rate.

From Fig. 4.6, it is observed that the long-term transient response can be represented by the remaining part of the transient response (46 – 50 seconds). Applying the values of V_K and τ_K to (4.3) and (4.4), we have $R_K = 16.85m\Omega$ and $C_K = 336.46F$.

The lithium-ion battery steady state internal resistance can be calculated as:

$$R_{int} = R_i + R_D + R_K \tag{4.5}$$

From (4.2) – (4.5), the values of R_{int} at different discharge ratings and state of charge are calculated and shown in Table 4.2. From the table, it is observed that the lithium-ion battery charging/discharging internal resistances are almost constants as the

charge/discharge rating varying. However, the value increases as the battery SOC decreases. When the battery SOC drops below 40%, the internal resistances experience a sharp change.

Battery internal resistances ($m\Omega$)		Battery state of charge				
		100%	80%	60%	40%	20%
Battery Discharge Rates	13.5A	132.62	138.04	139.3	206.94	223.81
	6.75A	132.84	137.92	139.78	208.72	224.56
	4.5A	133.91	137.78	140.0	207.3	225.76
	1.35A	131.27	139.21	140.74	208.25	228.30
Battery Charge Rates	13.5A	212.28	217.64	221.29	258.72	272.46
	6.75A	212.76	218.05	222.02	258.37	272.68
	4.5A	212.95	217.36	221.65	258.64	273.21
	1.35A	211.92	217.69	220.85	257.53	273.06

Table 4.2 Battery internal resistances at different charge/discharge rates and SOC.

Taking the average resistance values at different charging/discharging ratings, the lithium-ion battery internal resistances at different state of charges are shown in Fig. 4.7. From the figure, it is observed that the battery internal resistances increase as its state of charge decreases in both charging and discharging situations. These average internal resistances will be used to determine the battery charging/discharging rate factor lately in the following discussion.

Fig. 4.7 Battery internal resistances at different SOC during charging/discharging.

As discussed in Chapter 2, the battery discharge tests can be conducted using an electronic load as the constant current sink. The data acquisition system logs the battery output voltages under different discharging ratings at a frequency of 1 sample/second. The experimental results are shown in Fig. 4.8. Consider the discharge current rating of $C/3$ (4.5 Amps) as reference and shift other voltage curves along the y -axis by a value of $(I_{ref} - I) \cdot R_{int}$. The values of R_{int} at different battery SOC can be obtained from Fig. 4.7. A set of new battery discharging curves are formed and shown as the dash-dotted curve in Fig. 4.8. From the figure, the battery discharge rate factor is estimated as:

$$R_{discharge} = \frac{a}{b} \tag{4.6}$$

where b is the total capacity released; and a is the x -axis intersection by shifting the battery discharge curve at the reference current I_{ref} for a level of $(I_{ref} - I) \cdot R_{int}$ along the y direction.

Fig. 4.8 Battery voltage curves under 13.5A, 6.75A, 4.5A and 1.35A constant current discharging.

From Fig. 4.8 and (4.6), the discharge rate factors at different current levels are determined and shown in Fig. 4.9, with a reference discharging current of $C/3$.

To determine the battery charge rate factor, the lithium-ion battery is charged at constant current levels till its output voltage reaches 29.4 Volts. The current will eventually decrease to trickle current level of $C/50$. Dividing the designed battery full capacity by the total capacity delivered at different current ratings, the battery charge rate factor can be calculated. From the experiments, it is approximately a constant at 0.905. The

temperature effect is neglected since the battery in the satellite will be kept within a narrow range of 3°C-18°C.

Fig. 4.9 Battery discharge rate factors, with reference to C/3(4.5A) discharging current.

With the knowledge of battery charge and discharge rate factors, the battery state of charge can be calculated as:

$$SOC_{(t_1)} = SOC_{(t_0)} + \frac{\int_{t_0}^{t_1} V_{O(t)} \cdot I_{O(t)} \cdot dt}{R} \tag{4.7}$$

where R is the charge/discharge rate factor, $V_{O(t)}$ and $I_{O(t)}$ are the battery output voltage and current. Moreover, $I_{O(t)}$ is negative when the battery is discharging, and positive otherwise.

The estimated battery SOC based on whole orbit simulation results will be compared with the manufacturer data later in Chapter 6. From the results, we can observe that the estimated results match manufacturer data well.

4.3 Conclusions

A lithium-ion battery model has been proposed to estimate the battery state of charge simply based on the battery output voltage and current ratings. This model allows continuous varying load condition. Moreover, this model requires no manual tuning of parameters. It has been implemented in the test system. From the model, the battery SOC data can be estimated in real-time during the satellite whole orbit test. Estimated results will be compared with manufacturer data in the following Chapter based on the results.

Chapter 5

Electrical Ground Support System

Power supply system architecture for X-SAT has been designed in the previous Chapter. Besides the normal functional tests, there are many test such as the power budget test, voltage range test and battery performance test etc. To conduct these tests, a test environment that is able to simulate the real satellite mission operations is to be developed. This environment is required to fully automatically simulate the solar panel as the input to the PSS boards, the loading power as the output according to the mission operations. Data acquisition function and real time data analysis function are also needed.

The concept of Virtual Instruments (VI) has demonstrated its usefulness in many automated testing system and has been implemented in various industrial applications [22-29]. Thus, the virtual instruments concept is adopted for the development of X-SAT integrated testing system. In this chapter, an integrated testing system is presented. For the developed system, a new model of solar cell is proposed to estimate solar cell parameters and control the instrument output.

5.1 Virtual instrumentations

Virtual instruments are new generation of instruments that leverage the computational power, digital I/O, memory and storage, and sophisticated graphical programming

capabilities of the modern computers. VIs are different from traditional instruments both outwardly and inwardly. Outwardly, VIs are computer-based instruments that consists of computers and instrument I/O devices. Inwardly, the functionality of VIs are written in software programs, while traditional instruments are built almost exclusively into hardware and dedicated firmware [61-63]. Currently, LabVIEW from National Instruments and VEE from Hewlett Packard are the most widely used VI languages.

LabVIEW is a graphical programming language used for creating test, measurement and automation applications. It uses icons instead of lines of text to create applications. Unlike text-based programming languages, LabVIEW uses dataflow programming, where the data flow determines the flow of execution of the program. In the program, the user interface is built by a set of tools and objects provided. The user interface is called the front panel. Then the code using the graphical representations of functions, operators and controls is added to the front panel objects.

5.1.1 VI system architecture

The block diagram of the test system is shown in Fig. 5.1. As shown in the figure, a personal computer (PC) is used to serve as the VI host and intranet client. It connects to the file server through transmission control protocol/internet protocol (TCP/IP) and downloads the operating scenario data such as the satellite orbital path, the orientation of the satellite and the temperature of the solar array etc in real time. These information determines the expected solar power received by the satellite in the space.

In the laboratory, a solar array simulator (Agilent E4350B) is used to simulate the primary power source. Its output voltage is programmed to vary depending on the operating scenario that determines the solar incidence angle and the operating temperature. An electronic load (HP E6050B) is also connected to the output of the PSS to provide dynamic loading according to the operation of various satellite payloads. Voltage and current sensing circuits incorporated in the system are used for continuously monitoring the battery, the main bus and the solar array. These detected signals are transmitted to the data acquisition system. The virtual instrument platform controls the solar array simulator and electronic load using the GPIB interface. A National Instrument

SCXI data acquisition system comprises of model SCXI-1104C multiplexer input module, model 1102 thermocouple input module and SCXI-1600 USB data acquisition module are used for signal acquisition.

Fig. 5.2 VI program flow chart.

The virtual instrument platform is programmed to support a graphical user interface for setting the instrument parameters for different test scenarios. It also performs data display, data analysis and data logging to show the system performance characteristics. The flow chart of the various functions performed by the VI program is shown in Fig. 5.2.

As shown in the figure, there are mainly 5 function loops in the system:

- Instruments control loop – controls and updates instrument parameters every 50 seconds base on the operating scenario data. With Attitude Determination and Control System (ADCS) integrated together with PSS, this loop is able to work in real-time when ADCS is generating and supplying operating scenario data.
- PCM data acquisition loop – controls and monitors battery, main bus and PCM board at a 1 Hz frequency since the voltage and current of solar array and battery are changing at a low speed.
- PDM command & acquisition loop – commands and detects the status of every individual payload. The sub-system status is scanned at 20 Hz frequency. All the analog current values are scanned at a much higher frequency of 750 KHz to 1MHz, since the in-rush current and settling time are critical factors for PDM design and testing.
- Data analysis loop – scales and error compensates the collected raw data and performs data analysis.
- Data logging loop – logged test results into database for historical results viewing and analysis.

The VI platform is built based on state machine architecture. The design goal is to decompose the complex testing system into a set of states for the convenience of both module level testing and system level testing. Every individual function loop can be enabled or disabled without affecting other loops depending on the need of the user-defined tasks.

5.1.2 System modeling

Instruments control algorithm is critical for the PSS integrated testing system. The system uses a solar array simulator to serve as the primary power source and an electronic load for the real-time loading power. The modeling algorithms for these instruments are described in the following section.

5.1.2.1 Solar array simulator

To properly test and validate the satellite operations on earth, the solar array simulator is programmed in the curve mode that simulates the power generated by solar panels based on four parameters: open circuit voltage V_{oc} ($=2.593\text{V}$), short circuit current I_{sc} ($=465\text{mA}$), maximum power voltage V_{mp} ($=2.3\text{V}$) and maximum power current I_{mp} ($=455\text{mA}$). These parameters are affected by factors such as the I - V characteristics of the solar cell, the operating temperature and the incidence angle.

As described in Chapter 2, the typical model of a solar cell is shown in Fig. 2.5 and repeated in Fig. 5.3 for convenience.

Fig. 5.3 Solar cell model.

From (2.4), the short circuit current I_{sc} is calculated as:

$$I_{sc} = I_{ph} - I_{rs} \left[\exp \left(\frac{q(V + R_s I_{sc})}{kTA} \right) - 1 \right] \quad (5.1)$$

Moreover, the solar cell output voltage V can be expressed as:

$$V = \frac{kTA}{q} \cdot \ln \left(\frac{I_{ph} - I}{I_{rs}} + 1 \right) - R_s \cdot I \quad (5.2)$$

Thus, the solar array output power is calculated as:

$$P = I \cdot \left(\frac{kTA}{q} \cdot \ln \left(\frac{I_{ph} - I}{I_{rs}} + 1 \right) - R_s \cdot I \right) \quad (5.3)$$

At maximum power point,

$$\frac{dP_{\max}}{dI_{mp}} = 0 \quad (5.4)$$

Substitute (5.3) into (5.4), the maximum power point current I_{mp} can be calculated from

(5.5),

$$\frac{d \left[I_{mp} \cdot \left(\frac{kTA}{q} \cdot \ln \left(\frac{I_{ph} - I_{mp}}{I_{rs}} + 1 \right) - R_s \cdot I_{mp} \right) \right]}{dI_{mp}} = 0 \quad (5.5)$$

Expressing (5.2) and (5.5) using Taylor series up to second order, we have:

$$I_{sc} = I_{ph} - I_{rs} \cdot \left(\frac{q \cdot R_s I_{sc}}{kTA} + \frac{1}{2} \left(\frac{q \cdot R_s I_{sc}}{kTA} \right)^2 \right) \quad (5.6)$$

$$-\frac{3}{2I_{rs}^2} I_{mp}^2 + \left(\frac{2I_{ph}}{I_{rs}^2} - 2 \frac{1}{I_{rs}} - 2R_s \right) \cdot I_{mp} + \left(\frac{I_{ph}}{I_{rs}} - \frac{I_{ph}^2}{2I_{rs}^2} \right) = 0 \quad (5.7)$$

For simplicity, assuming $R_s \rightarrow 0$, from (5.6) and (5.7), we get that the value of $\frac{I_{mp}}{I_{sc}}$ is

independent of the incident angle θ .

When the incident angle is not significant ($<60^\circ$, which is applicable to X-SAT), the solar cell open-circuit voltage V_{oc} is approximately constant, and the solar cells short-circuit current changes in a cosine factor manner as discussed in Chapter 2, which can be described as:

$$V_{oc(\theta)} \approx V_{oc(\theta=0)} \quad (5.8)$$

$$I_{sc(\theta)} = I_{sc(\theta=0)} \cdot \cos \theta \quad (5.9)$$

As discussed, the value of $\frac{I_{mp}}{I_{sc}}$ is independent of the incident angle θ . Thus, from (5.9),

$$I_{mp(\theta)} = I_{mp(\theta=0)} \cdot \cos \theta \quad (5.10)$$

As a measure of solar cell quality, the Fill Factor (FF) is used and expressed as:

$$FF = \frac{I_{mp} \cdot V_{mp}}{I_{sc} \cdot V_{oc}} \quad (5.11)$$

The solar cell fill factor is a function of the open circuit parameters and can be calculated as follows[72],

$$FF \approx \frac{v_{oc} - \ln(v_{oc} + 0.72)}{v_{oc} + 1} \quad (5.12)$$

where

$$v_{oc} = V_{oc} \cdot \frac{q}{mkT} \quad (5.13)$$

where V_{oc} is the open circuit voltage, m is the diode ideality factor whose value is between 1 and 2.

From (5.11) - (5.13), we observe that FF is a function of temperature T only. Substitute (5.8) - (5.10) into (5.11), we get maximum power point current is a function of temperature T only, which is:

$$V_{mp}(\theta) \approx V_{mp}(\theta=0) \quad (5.14)$$

Taking temperature effects into consideration, and from (5.8) - (5.10) and (5.14), the instantaneous parameters of solar array in series-parallel configuration are calculated as:

$$V_{oc}(\theta, T) = [V_{oc}(0, T) + K_v \times (T - T_o)] \times n_s \quad (5.15)$$

$$I_{sc}(\theta, T) = [I_{sc}(0, T) + K_i \times (T - T_o)] \times \cos \theta \times n_p \quad (5.16)$$

$$V_m(\theta, T) = [V_{mp}(0, T) + K_{vm} \times (T - T_o)] \times n_s \quad (5.17)$$

$$I_m(\theta, T) = [I_{mp}(0, T) + K_{im} \times (T - T_o)] \times \cos \theta \times n_p \quad (5.18)$$

where T_o is the reference temperature ($=29.2^\circ\text{C}$), n_s ($=16$) and n_p ($=18$) are the number of solar cells in series and strings in parallel, and the thermal coefficients of the solar cell are $K_i = 5.3 \mu\text{A}/^\circ\text{C}$, $K_v = -5.9 \text{ mV}/^\circ\text{C}$, $K_{im} = 1.2 \mu\text{A}/^\circ\text{C}$, $K_{vm} = -6.5 \text{ mV}/^\circ\text{C}$.

5.1.2.2 Electronic load

The satellite has several operating modes. In the typical operation, only one payload is activated in each orbit. Different tasks are carrying up at different time and orbital position. Take the IRIS and Idle operation modes for example, whose required power consumption in one orbit period is shown in Fig. 5.4.

Fig. 5.4 Power consumption under idle and IRIS load.

From Fig. 5.4, the power consumption for one orbit period can be described using step function as:

$$\begin{aligned} P_{IRIS(t)} &= P_{IRIS(100-t)} \\ &= 72.5 \cdot u(t) + 18 \cdot u(t-t_1) + 7.1 \cdot u(t-t_2) \\ &\quad + 61.6 \cdot u(t-t_3) + 66.8 \cdot u(t-t_4) \quad \text{when } 0 < t \leq 50 \end{aligned} \tag{5.19}$$

where $t_1=37.5$, $t_2=41.7$, $t_3=43.3$ and $t_4=47.5$ in minutes.

$$P_{Idle(t)} = 70.4 \cdot u(t) + 41.8 \cdot u(t-t_1) \quad \text{when } 0 < t \leq 50 \tag{5.20}$$

where $t_1=41.7$ in minutes.

The electronic load is used in the system setup to program the dynamic loading for the payload that is being used. It is programmed as a current sink while the current is changed and updated in real-time. The instantaneous main bus voltage $V(t)$ of the PSS is measured and feedback. With the load power consumption $P(t)$ and the output voltage $V(t)$, the required instantaneous current for the electronic load is calculated as

$$i(t) = \frac{P(t)}{V(t)} \quad (5.21)$$

5.1.3 System interface

The VI front panel graphical user interface is shown in Figs. 5.5 and 5.6. The front panel manipulations can be categorized into the following functions:

- System initialization & control – the system initialization resets all the hardware and software status. Instruments address, solar array simulator parameters, electronic load operating mode and data acquisition system configurations can be changed and updated.
- PCM data acquisition & analysis – voltage and current from battery pack, solar array output and main bus are acquired and displayed in graphs. While digital status and command are shown as LEDs. System performance data such as battery status, satellite position and power efficiency are analyzed and displayed on the panel in real time. Alarming and emergency reset functions are also available in the user interface.
- PDM command & data acquisition – PDM command function switches on/off every individual subsystem either manually or automatically, while PDM data acquisition monitors the working condition of all the subsystems, including

current levels and over current protection activation status. A custom signal conditioning hardware has been designed and implemented to amplify the signals, provide isolation, along with synchronized sample and functionality.

- Data logging – All the raw data from the DAQ devices are scaled and error compensated prior to updating the front panel and logging to the database. Moreover, we can take advantage of configurable data extraction and viewing with historical, real-time trend graphs and tabular data display through the VIs.

Fig. 5.5 Electrical ground support system instruments control interface.

Fig. 5.6 Electrical ground support system PCM testing graphical interface.

5.2 Conclusions

A new approach has been proposed to estimate the solar array parameters including the open circuit voltage, short circuit current, maximum power point voltage and current. An integrated testing system has been developed successfully in virtual instrument environment. By implementing the proposed modeling, the EGSS has proven to be capable of simulating the satellite operations realistically.

Chapter 6

Experiment results

In the previous chapters, the X-SAT power supply system boards and electrical ground support system have been designed and integrated. In order to evaluate the power system performance during mission operations, whole orbit test needs to be conducted. Moreover, electrical performance of the lithium-ion battery needs to be characterized throughout the whole orbit tests. This is important especially in a non-fully regulated power system such as X-SAT where all the power performances are directly linked to the battery voltage. Throughout the satellite engineering model design, many orbital tests under different operating scenarios have been conducted. In this study, the voltage and current of the solar array bus, battery output and main bus are logged in real-time for analysis. Based on the logged results, the energy balance of the satellite in all mission operations has been calculated. The solar energy utilization factor of the proposed power system has also been estimated. Moreover, the battery state of charge profiles has been characterized based on the proposed battery model. Besides functional and whole orbit tests, thermal cycling tests have also been conducted on the electronic boards. The experiment and simulation results will be presented and discussed in this chapter.

6.1 Whole orbit test

The VI based power system setup for the whole orbit test has been described as shown in Fig. 5.1. For the whole orbit tests, the solar array simulator is used to generate the X-SAT

solar panel output power based on the open circuit voltage V_{oc} , short circuit current I_{sc} , maximum power point voltage V_{mp} and maximum power point current I_{mp} . Electronic load simulates the operation load power based on mission operation profiles. Both of the instruments are controlled by a PC through GPIB bus. Important power system parameters including solar array bus voltage and current, battery output voltage and current, main bus voltage and current are collected by the data acquisition system.

Fig. 6.1 One orbit experiment results under IRIS load.

Fig. 6.1 shows the experimental results over one orbit period when the satellite operates with the IRIS camera payload, which has the highest peak power consumption. The figure shows the currents and voltages from the battery, the main bus as well as the solar array. For the battery current shown in the figure, it is negative when the battery is discharging and positive otherwise. Based on these measurements, the power generated

by the solar array, the charging/discharging power of lithium ion battery and the power consumption from the electronic load can be calculated. These powers are shown in Fig. 6.2. The data sampling rate for these tests is 1 sample per second.

Fig. 6.2 Power profiles of solar array, lithium ion battery and loads.

From Figs. 6.1 and 6.2, it may be observed that one orbit takes approximately 100 minutes. They can be classified in 3 stages:

1. Eclipse stage

As shown in Fig. 6.2, the solar array power is zero during the eclipse (0-22.5 minutes). Under this situation, all the load power is supplied by the rechargeable battery through a discharge diode. Moreover, the main bus regulator operates at 100% duty cycle. It may also be observed from Fig. 6.1 that the battery output voltage decreases

gradually as the depth of discharge increases. In this stage, the battery is connected to the main bus through a diode. Thus, the solar array bus (SABus) voltage follows the decreasing battery voltage and can be observed in Fig. 6.1. The equivalent system under this operation is shown in Fig. 6.3.

Fig. 6.3 Equivalent PCM block diagram in eclipse stage.

2. Battery charging stage

When the satellite comes out from the eclipse, the solar array temperature is at its lowest range of -50°C . Due to the characteristics of the solar cell as explained in Fig. 2.9, both the output voltage and output power of the solar array are at their highest as shown in Figs. 6.1 and 6.2. Consequently, the solar array is capable of supplying power to all the payloads as well as recharges the battery. Subsequently, the solar array voltage decreases as the temperature increases. As the battery voltage increases, the battery charging current decays to zero after the battery reaches its highest voltage of 29.2V. This causes the discharge diode to be reverse-biased. When the solar array bus voltage (VSABus) is greater than 30.5V, the buck converter will regulate the output voltage to 30.5V. If the VSABus is less than 30.5V, the buck converter will

simply works as a MOSFET switch. The equivalent system of the PCM is shown in Fig. 6.4.

Fig. 6.4 Equivalent PCM block diagram in battery charging stage.

3. Battery discharging stage

Fig. 6.5 Equivalent PCM block diagram in battery discharging stage.

At 37.5 minutes, the satellite is commanded to rotate and start the IRIS operation. This causes the solar array incident angle increasing to 67°. Consequently, the power generated by the solar array decreases to 80W as shown in Fig. 6.2, which is smaller than the power required by payloads. Hence, the battery starts to discharge as shown in Fig.

6.1. In the battery discharging stage, the battery charge regulator is disconnected while the main bus regulator is operating at 100% duty cycle. The equivalent system is shown in Fig. 6.5.

The operating principle of different stages greatly influences the power system efficiencies, the battery output characteristics as well as the satellite power balance. The different characteristics of these three stages will be studied and discussed in the following sections.

6.2 System efficiency analysis

The efficiency of a satellite power system includes the system power efficiency and the solar energy utilization factor. The power efficiency takes into account of the power loss due to the electronic components and the conversion efficiency of the voltage regulators. It should be designed to minimize the power loss and heat dissipation. The solar energy utilization factor defines the percentage of energy used by the satellite with respect to the maximum energy that the solar panel can generate. The solar energy utilization factor depends on the power system topology, the loading scenario as well as the battery charging efficiency. In the following discussions, the IRIS operating scenario is taken as an example for both the power efficiency and solar energy utilization factor analysis.

6.2.1 Power efficiency analysis

The solar array generates power and always functions as the input to the power supply system. For the battery, it consumes power when it is being charged. On the other hand, it

supplies power when it is being discharged. Thus, the system power efficiency can be calculated as:

$$\eta = \begin{cases} \frac{\int_0^1 P_{Load} dt}{\int_0^1 (P_{SA} + P_{Bat}) dt} & \text{when battery is being discharged} \\ \frac{\int_0^1 (P_{Load} + P_{Bat}) dt}{\int_0^1 P_{SA} dt} & \text{when battery is being charged} \end{cases} \quad (6.1)$$

where P_{Load} is the load power, P_{SA} is the power generated by solar array, and P_{Bat} is the charging/discharging power of the lithium-ion battery. Fig. 6.6 shows the power control module input/output power as well as the power efficiency computed based on Fig. 6.1 and (6.1) over one orbit period under IRIS load. As described in section 6.1, both the main bus and battery charge regulators work at 100% duty cycle at eclipse stage and battery discharging stage. Under this condition, the conversion efficiency of the regulators is close to 100%. Thus, the system power efficiency is at its highest. As shown in Fig. 6.6, the power efficiency during eclipse stage and battery discharging stage remains relatively constant at 94%. During the battery charging stage, the power efficiency decreases since both the main bus and battery buck converters are regulating. As can be observed from Fig. 6.1, the solar array output voltage is 46 Volts when the satellite just comes out of eclipse. Both the main bus and battery charge regulators are active. Thus, the power efficiency is at its lowest value of 58%. From Fig. 6.6, the average power efficiency of the power system over one orbit period is 88.03%.

Fig. 6.6 PCM input power, load power and system power efficiency.

6.2.2 Solar energy utilization factor analysis

Fig. 6.7 compares the maximum and the actual power generated by the solar array under IRIS scenario. As shown in the figure, the maximum power that the solar array can generate is 340 watts when the satellite just comes out from eclipse. However, the required load power is only 75 watts as shown in Fig. 3.2. The actual output power from the solar array is 230 watts. Thus, the battery charging power is approximately 155 watts at the beginning of sunlight period. As the battery SOC increases, the charging power gradually decreases while the load power remains constant. Consequently, the solar array output power becomes smaller. At 37.5 minutes, the satellite rotates and the incident angle increases to 60°. At the same time, the IRIS load starts to operate. To meet the load

power requirement, the solar array works at its maximum power point. From the figure, it is observed that, during 52.5 – 57.5 minutes, the solar array output power is only 179 watts even though the load power is at its maximum value of 207 watts. The reason is that the battery is discharging. Thus, the solar array operating voltage is locked onto the battery voltage. Consequently, the solar array output power is much lower than its maximum power. At 87.5 minutes, which is the end of the sunlight period, the battery is almost fully charged. Thus, the solar array power is approximately equals to the load power. All the solar array power is used to supply the load. By calculating the area under each power curve, we obtain the total energy generated by the solar array over one orbit period is 197.06 Wh. The maximum energy that the solar array can generate is 243.96 Wh. Thus, the solar energy utilization factor is 80.82% under IRIS operating scenario.

Fig. 6.7 Solar array output power under IRIS operating scenario and theoretical maximum power.

6.3 Power balance analysis

Throughout the mission of the satellite, the solar array should be capable of supplying all the load power as well as recharging the lithium-ion battery to maintain the energy balance. During the simulation for power balance analysis, hot temperature profile, which is shown in Chapter 3, is adopted for solar array parameters estimation. The solar array EOL degradation factor is also taken into account. Lowest DC-DC converter efficiency, which is 75% from the manufacturer data, is used to calculate the load power consumption. All these assumptions are for the purpose of studying the worst case scenario.

Operating scenarios		Average power consumptions		Peak power consumption (Watts)
		during		
		Sunlight (Watts)	Eclipse(Watts)	
LEOP		66.68	66.68	66.68
Normal Operation Phase	Idle	80.48	75.83	111.76
	DataDownload	89.45	77.95	191.20
	IRIS	98.60	78.95	207.99
	PPU	106.27	80.06	143.99
	DLR-GPS	81.82	77.17	89.76

Table 6.1 Power consumption for various operating scenarios.

The detailed power figures in all the operation scenarios are shown in Appendix D. In the appendix, the input voltage, normal operating current level and duty cycle of all the

satellite modules are also shown. Thus, the load power ratings can be calculated by multiplying the input voltage and current level. Consequently, the average load power can be obtained by multiplying the respective load operational duty cycle and operating power consumption. These module power consumption figures are then summed up to obtain the overall power consumption for each type of operation. The average load power consumption during sunlight and eclipse under different scenarios are summarized as shown in Table 6.1.

Power balance in LEOP phase is critical for the success of the first ground contact. The first ground contact ensures the satellite entering NOP and carrying out different mission tasks. During the LEOP phase, the satellite is tumbling and the deployable solar panels are not deployed. Thus, the power generated by the solar array is small and the charging current of the lithium-ion battery is low. As discussed in Chapter 2, a longer charging period is required for the battery to be fully charged at a low charging current rating. Thus, the power balance analysis is to verify that the power supply system is capable of fully recharging the battery during the 65 minutes sunlight period. From Table 6.1, it is observed that the average power consumption of PPU is the highest of all the operation scenarios in the satellite normal operations. It requires an average power of 106.27 watts during sunlight and 80.06 watts during eclipse. During the NOP phase, if the power system is able to balance energy during PPU scenario, it is able to achieve power balance in all the other orbit scenarios. Thus, LEOP and PPU operation scenario during NOP are selected for the power balance analysis.

In the experiment results, the battery power is negative when it is discharging and is positive otherwise. As shown in the Figs. 6.8 and 6.9, the battery supplies all the load power in eclipse period.

6.3.1 Launch and early operation phase

During LEOP phase, the deployable solar panels are not deployed. For maximum power generation, the fixed panel and one deployable panel each faces the sun at 45° . The sunlight power factor is calculated as the sum of the power output of each panel divided by the power output of all panels at 0° solar incident angles. As introduced in Chapter 3, 4 strings of solar cells are fixed on the satellite structure. The other 14 strings are equally distributed on 2 deployable solar panels. Thus, during LEOP phase, the average sunlight power factor can be calculated as follows:

$$S_{avg} = \frac{P_{fixed}^{0^\circ} \cos 45^\circ + P_{deploy}^{0^\circ} \cos 45^\circ}{P_{total}^{0^\circ}} \quad (6.2)$$

where $P_{fixed}^{0^\circ}$ and $P_{deploy}^{0^\circ}$ are the power generated by the fixed panel and deployable panels under normal sunlight respectively.

Since all the solar cells are well matched, (6.2) can be simplified to:

$$S_{avg} = \frac{N_{fixed} \cos 45^\circ + N_{deploy} \cos 45^\circ}{N_{total}} \quad (6.3)$$

where N_{fixed} and N_{deploy} are the number of solar strings on fixed panel and deployable panels respectively. With N_{fixed} , N_{deploy} and N_{total} being 4, 7 and 18 respectively, we have,

$$S_{avg} = \frac{4 \cdot \cos 45^\circ + 7 \cdot \cos 45^\circ}{18} = 0.432 \quad (6.4)$$

During LEOP, only essential subsystems such as ADCS, OBC and PSS are activated to minimize the load power consumption. Thus, for the orbit scenario experiments, we assume a constant sunlight power factor of 0.432, and a constant load power of 66.68W throughout the LEOP phase. The experimental results are shown below in Fig. 6.8. By calculating the area under the power curves in the figure, the total energy discharged by the battery during eclipse period is 48.62Wh. Since the full capacity of the 7s9p battery is 340Wh, the theoretical battery DOD during LEOP is calculated to be,

$$\frac{48.62}{340} = 14.3\% \quad (6.5)$$

which is smaller than the required 25% DOD. The actual battery DOD during LEOP can be estimated based on the experimental results to be discussed in Chapter 6.4.3.

From Fig. 6.8, it is also observed that during the sunlight period, the solar panel supplies the load power as well as recharges the battery. The battery charging power decreases as the battery state of charge approaches 1. At the end of sunlight period, the battery achieves its highest output voltage of 29.4 Volts. Moreover, the charging power drops to approximately 15 watts. Thus, the battery charging current is 0.5 Amps, which is approximately of C/27 current rating.

Fig. 6.8 Solar array power, battery power and load power during LEOP phase.

6.3.2 PPU operation in normal operation phase

In PPU operating scenario, the satellite will keep sun-pointing for the whole orbit. Thus the sunlight power factor is 1. The load power profile is shown in Fig. 3.2. The power generated by the solar panel, charging/discharging power of the lithium-ion battery and the load power under PPU loading scenario over one orbit period are shown in Fig. 6.9.

Based on the power diagram, the total energy discharged by the battery during eclipse period is 48.557 Wh. The battery depth of discharge during PPU operation is

$$\frac{48.337}{340} = 14.22\% \quad (6.7)$$

Fig. 6.9 Solar array power, battery power and load power under PPU scenario.

From Fig. 6.9, we calculated that the total energy generated by the solar array during sunlight period is 229.0023 Wh, of which 70.533 Wh is to recharge the lithium-ion battery to fully charged state. All the load power is supplied by the rest of the power generated.

Since the battery can meet the maximum depth of discharge requirement, and the solar array is capable of fully recharging the battery during the sunlight periods of both LEOP and PPU operations, we conclude that the designed power system is able to meet the energy requirements of all the load scenarios throughout the mission life.

6.4 Battery module analysis

6.4.1 Battery trickle charge observation

Controlling the battery overcharge is extremely important for maintaining the battery temperature, which has a strong influence on the battery life. Thus, when the battery full charge is approached, the battery charge current is often tapered down to the trickle rate and maintains the battery SOC at around 95%. The charge rate at trickle charge stage is in the $C/100$ to $C/50$ range. However, before the battery reaches its maximum voltage, the charge rate is maintained constant between $C/4$ to $2C$ range to maximize the charging efficiency and shortens the charging time.

Fig. 6.10 Battery voltage and current over one orbit period.

Fig. 6.10 shows the battery output voltage and current under PPU scenario over one orbit period. During the eclipse, the discharge current rises (becomes more negative) as the battery voltage drops because of the constant power nature of the load. As the battery is placed in charge at the beginning of sunlight, the battery voltage rises under constant charging current, which equals 4.5 Amps (C/3). When the battery output voltage approaches its maximum voltage (29.2 Volts), the charge rate gradually tapering to 0.46 Amps (C/30) at the inception of the next eclipse.

6.4.2 Battery latch-up analysis

Fig. 6.11 Experiment result showing battery latch-up phenomenon.

The reason for battery latch-up phenomenon has been discussed in Chapter 3.3. This phenomenon is also shown clearly in Fig. 6.11, which is the experiment result under IRIS operating scenario over one orbit period. When the load power is at its maximum value of 207 watts at 47.5 minutes, the battery is discharging even though the satellite is in sunlight. The battery output voltage is lower than 28.7 Volts. Both the main bus and the solar array voltages are locked at the battery voltage, and differ by a diode drop voltage. When the load power reduces and the battery is being charged, the battery output voltage is greater than 28.7 Volts. At this period of time, the solar array simulator is working in the region where its output voltage is greater than the maximum power point voltage.

The advantage of this power supply system architecture over the conventional DET sun-regulated topology can also be observed from Fig. 6.11. When the satellite just comes out from eclipse, the solar panel output voltage is 46 Volts. In the proposed system, the main bus voltage is well regulated at 30.5 Volts and all the power generated is used to supply the load as well as recharge the battery. However, if such a high input voltage happens in the convention design, some of the shunt regulators are activated to regulate the main bus voltage within the designed range. Certain amount of solar energy is shunted into ground. Thus, the solar utilization factor decreases. Consequently, the battery is not being charged at the maximum power and the battery latch-up problem arises. Eliminating the shunt regulators in this design provides the advantage to overcome all these problems at cold temperature environment. This is critical for X-SAT to survive the LEOP stage due to the low solar panel temperature during that period.

6.4.3 Battery SOC estimation

By implementing the proposed battery model as described in Chapter 4 into the integrated test system, the battery SOC can be estimated in real-time as the test is running. The estimated battery SOC under IRIS and PPU load over five continuous orbits are shown in Figs. 6.12 and 6.13 respectively. By comparing the results with the manufacturer data, we observe that the estimated results have an average error of 1.2% and a maximum error of 2.7%. Thus, the estimated battery model is fairly accurate and it can be used in our system for predicting the battery status.

Fig. 6.12 Estimated battery SOC under IRIS operating scenario for 5 orbits period.

Fig. 6.13 Estimated battery SOC under PPU operating scenario for 5 orbits period.

From Fig. 6.12 – 6.13, it is observed that during the IRIS operation, when power balance is achieved, the lowest battery SOC is stabilized at 76.2% at the EOL of the satellite. The stabilized lowest battery SOC for PPU operation is at 79.6%. Though the PPU load has the highest average power consumption among all the load scenarios, it is observed that the lowest battery SOC is higher in PPU scenario as compared to IRIS scenario. The reason is that the battery is discharging during the peak load power period of IRIS scenario. Since the lowest battery SOC is 76.2% at EOL of the satellite, we conclude that the designed power system is capable of meeting the requirements of 75% lowest battery SOC throughout three years mission life.

Fig. 6.14 Battery SOC under orbit number 1-4 of LEOP stage.

Fig. 6.15 Battery SOC under orbit number 50-52 of LEOP stage.

Figs. 6.14 and 6.15 show the battery SOC under LEOP. Fig. 6.14 shows the results for the first 4 orbits of LEOP, while Fig. 6.25 shows the results for orbit number 50-52. From the figures, it is observed that for the first several orbits, the battery SOC drops gradually at the end of each orbit. However, after 49 orbits period, the power system balances the battery charging and discharging power. The lowest battery SOC during LEOP is 39.2%. The results give us the same conclusion as has been discussed in Chapter 6.3 that the designed power system is capable of balancing the satellite power during LEOP.

6.5 Summary of the whole orbit tests results

The X-SAT power system whole orbit test results are summarized in Table 6.2. From the table, it is observed that the average power efficiency of the power system is above 80% in all power scenarios. Moreover, the solar energy utilization factor is between 71% -85% depends on the operation scenario as well as the initial battery state of charge. Generally, higher load power and lower battery state of charge require more solar energy to be generated. Consequently, the solar energy utilization factor is higher. However, this is not true for IRIS scenario. During the peak load power period, the solar array output voltage is locked onto the battery voltage instead of the maximum power point voltage. Thus, the solar energy utilization factor decreases. From the table, it is also observed that the highest battery DOD during normal operation phase is 23.8% at EOL. The lowest battery SOC is 39.2% during the LEOP stage when the power balance is achieved. Thus, we conclude that the designed power system is capable of keeping power balance during LEOP stage until the successful first ground contact. Once the satellite enters NOP stage,

the battery can be fully charged and the lowest battery SOC is 76.2% throughout the satellite mission life.

Operation scenario	Power efficiency (%)	Maximum solar energy (Wh)	Solar energy generated (Wh)	Solar utilization factor (%)	Lowest battery SOC (%)
Idle	87.65	270.0886	193.0593	71.48	82.92
IRIS	88.03	243.9608	196.8538	80.69	76.20
Data download	87.73	270.0886	202.1823	74.88	80.90
PPU	82.02	270.0886	229.0023	84.79	79.6
LEOP	89.76	116.678	91.73	78.62	39.2

Table 6.2 Summary of the power supply system whole orbit tests results.

6.6 Thermal cycle test

The objective of the thermal cycle test is to evaluate the performance of the power supply system electronic boards in extreme temperature environments. The test is conducted using a thermal chamber. The thermal chamber controls its internal temperature automatically based on programmed thermal cycle profiles. During the tests, thermocouples are used to sense the temperature of various electronic components. Subsequently, the temperature, voltage and current data are logged by the data acquisition system. The thermal cycle test includes operation scenario test and functionality test.

6.6.1 Operation scenario test

The operation scenario test is to confirm that the power system is able to support the dynamic and static loads during its mission operations at the extreme temperatures. The

objective of the test is to observe the battery performance, the main bus voltage variations, the payload operation during dynamic loading and temperature variations.

Fig. 6.16 Block diagram of the thermal cycle test set-up.

Fig. 6.16 shows the block diagram of operation scenario test set-up. As shown in the figure, the PCM and PDM are electrically separated during the tests for fault isolation purpose. The load of the PCM board is programmed using a HP E6050B electronic load. The power of the PDM and ADCS Interface Module Power (AIM-PWR) boards are supplied by DC power supplies with a constant voltage of 28 Volts throughout the tests.

LabVIEW based test programs for the PCM and PDM are independent so that any error occurs in one system would not affect the testing of the other. However, both the programs start simultaneously with equal time step to simulate more realistically. Both hot case solar array temperature profile as shown in Fig. 3.3 and the EOL degradation factor as summarized in Table 2.3 are included to simulate the worst case scenario.

Fig. 6.17 Programmed and actual thermal cycle profiles.

The thermal chamber is programmed to operate for three continuous thermal cycles with a temperature range of 10°C to 60°C, which is determined from the component working temperature range. The programmed as well as the actual thermal cycle's interval graphs are shown in Fig. 6.17. From the figure, it is observed that the thermal chamber takes approximately 25 minutes to stabilize when the temperature changes. Moreover, the

programmed temperature and the actual chamber temperature have a maximum error of 0.7°C after stabilization.

As shown in Fig. 6.17, each thermal cycle includes 4 steps during the tests. The first step is the constant low temperature period; second and fourth steps are temperature ramping periods while the third step is the constant high temperature period.

Fig. 6.18 Satellite load power profile under 'eclipse data download' scenario.

In the constant low temperature period, the system is commanded to operate at 10°C for 60 minutes. The first 15-20 minutes are meant for the chamber temperature to be stabilized. Subsequently, the eclipse period of 'eclipse data download' scenario is simulated, which is indicated by "A" in the figure. The 'eclipse data download' scenario

is the most power consuming operation during eclipse period of X-SAT. The load power profile is shown in Fig. 6.18. In the figure, the satellite is in eclipse for the first 35 minutes, which is represented in solid line; while the remaining 65 minutes are under sunlight.

The chamber is programmed to work under maximum ramping rate during steps 2 and 4. Both the ramping-up and ramping-down periods are approximately 12-15 minute. During the ramping period, the power supply system operates in standby mode, i.e., the solar array simulator did not supply any power, the load power drawn by the electronic load is zero, and all the subsystems connected to the PDM are in the OFF states.

As shown in Fig. 6.17, sunlight period of IRIS operation scenario (highest peak power consumption scenario) are conducted for the first 2 thermal cycles when the temperature stabilized at 60°C, which is indicated using “B”. “C” indicates the sunlight period of PPU operation scenario (highest average power consumption scenario). It is conducted during the constant high temperature period of the third thermal cycle. Both IRIS and PPU scenario tests take 65 minutes. The remaining 25 minutes are allocated to allow the temperature to be stabilized. The load power profile of the IRIS and PPU loading scenario are shown in Fig. 6.19.

Fig. 6.19 Load power profiles of IRIS and PPU operation scenario.

6.6.2 Functional test

The objective of the functional test is to check the functionality of individual power supply system module in both the low (10°C) and high (60°C) temperature environments. Results to be observed are the switching between main and redundant components (including battery charge regulators, main bus regulators and DC-DC converters), as well as PDM switching and over-current trip profile.

The power supply system set-up is the same as the operation scenario test with the thermal chamber profile changed. The thermal chamber now remains at 10°C all the way till all the tests completed. Subsequently, it ramps up to 60°C. All the functional tests are

then conducted again at high temperature. Switching functionality is observed from PSS-EGSE program, while the over current trip waveforms are observed and measured using oscilloscope.

6.6.3 Observations and discussions

6.6.3.1 PCM

The PCM voltage and current results for 3 cycles of operation scenario tests are shown in Fig. 6.20 and 6.21. The operations during each test steps are labeled in the figures. From the figures, it is observed that the lowest battery output voltage is 25 Volts during eclipse when the load power is at its maximum value of 191.2 watts. Thus, the lithium-ion battery is capable of supplying all the load power during eclipse at 10°C. Moreover, the battery output voltage reaches its full charge voltage at the end of every thermal cycle. Thus, the system is capable of balancing the power in the worst case scenario. It is also observed that the maximum solar array output voltage is 55 Volts when the satellite just enters sunlight at the second cycle. During this high solar array voltage stage, the battery is being charged and the main bus regulator regulates the main bus voltage to 30.5 Volts. Thus, the power system eliminates the battery latch-up problem in the cold temperature environment.

Preliminary test with an infra-red imaging camera has been conducted in normal laboratory environment before the thermal cycle tests. From the infra-red pictures, it is observed that the possible hot spots/areas on PCM are the driving transistor of MBR, driving transistor of BCR and the main data bus copper board. The driving transistors are

hot when they are regulating. The main bus copper board temperature increases when the current flow increases.

Temperature profiles of the transistors and copper board are logged and shown in Fig. 6.22. From the figure, it is observed that in the second thermal cycle, the solar array output voltage is as high as 55 Volts when the satellite just entering sunlight. Thus, the main bus regulator regulates the main bus voltage to 30.5 Volts. Consequently, the MBR transistor temperature increases to its highest temperature of 140°C.

At the end of the third cycle, the battery reaches its full charged voltage. Moreover, the solar array output voltage is approximately 35 Volts. Thus, both the battery charge regulator and main bus regulator are regulating. Consequently, the BCR transistor temperature increased to 96°C, while the MBR transistor temperature increased to 114°C. We observed that in the third thermal cycle, the MBR transistor temperature is lower as compared to the second cycle. The reason is that the solar array output voltage is in a lower range of 35 – 40 Volts.

The main data bus temperature increases during ‘eclipse data download’ scenario period. As the respective satellite modules being switched on, the current flowing through the main data bus gradually increase. Consequently, the heat generated by the copper bus increases due to I^2R loss. From the experiments, the highest temperature difference between the copper bus and the ambient air is approximately 10°C.

Fig. 6.20 PCM voltages during 3 cycles of operation scenario tests.

Fig. 6.21 PCM currents during 3 cycles of operation scenario tests.

Fig. 6.22 Temperature profile under 3 cycles of operation scenario tests.

The experimental PCM board temperature profile is summarized in Table 6.3.

Hot spots	MBR transistor	BCR transistor	Main bus copper board
Highest Temperature (respect to ambient air)	87 °C	83 °C	10 °C
High temperature period	When MBR is regulating	When BCR is regulating	Data bus current increases due to data download operation.
Remarks	Temperature depends on solar array output voltage. The higher the VSABus, the higher the temperature	Temperature depends on solar array output voltage. The higher the VSABus, the higher the temperature	Temperature increases as the current flowing through data bus increases. When the current is at its maximum value of 9.8 Amps, the copper board temperature is about 10°C above the ambient air.

Table 6.3 Summary of PCM board temperature profiles.

6.6.3.2 PDM

Throughout the thermal tests, the PDM1, PDM2 and AIM-PWR boards are powered using DC power supplies and connected to resistive dummy loads as shown in Fig. 6.16. They undergo 3 cycles of thermal tests simultaneously with the PCM board. All the subsystems are switched on and off automatically based on the operating scenarios. These are shown in Appendices A, B and C. In the tables, the green background color indicates the payload is on, red means off, while yellow color means the payloads is for redundancy. Redundancy payloads are always off unless the primary payloads are switched off. Throughout the tests, the payloads are switched in a time stamp of 50 seconds.

From the preliminary test with the infra-red imaging camera, the identified hot spots on the PDM boards are the DC-DC converters, switching MOSFETs and sensing resistors. Due to limited number of thermocouple channels, only the X-Band transmitter solid state (XBTSS) module on/off switching MOSFET and its sensing resistor are monitored, as they have the highest current flow. The characteristics of other on/off switching MOSFETs and sensing resistors are expected to be similar but with a lower temperature.

The temperature profiles of the PDM boards obtained from the thermal cycle tests are shown in Figs. 6.23 and 6.24. Fig. 6.23 shows all the DC-DC converters temperature, while Fig. 6.24 shows the MOSFET and resistors temperature. From the figures, it is observed that all the DC-DC converters are of similar characteristics. Their temperature rises when the respective payload is connected, and drops otherwise. From Fig. 6.23,

MSA 2805S and MTR 2805SF are the hottest with a temperature of 14-18°C higher than the ambient air temperature, while MTR 283R3SF temperature is the lowest due to the fact that its respective module, the solid state recorder (SSR) only operates for a short duration.

The yellow line in Fig. 6.24 is the OBC1 sensing resistor temperature; the blue and purple curves are XBTSS switching MOSFET and its sensing resistor temperature. OBC1 is always on and its sensing resistor temperature is about 8 -10 °C higher than the ambient air. When the XBTSS module is switched on, the temperature of its switching MOFET increases by 3°C and the resistor temperature increases by 5 -6 °C.

The thermal characteristics of the PDM boards' components are summarized in Table 6.4. From these results, we conclude that the temperature of PDM components increases only when their respective load is operating. Moreover, the temperature increases with respect to the ambient air is not significant. Thus, the performances of the PDM boards are expected to be stable when the environment temperature varies.

Hot spots/ components	DC-DC converters :	DC-DC converters:	Switching MOSFET for XBTSS	Sensing Resistor of XBTSS
	MHV 2805SF MSA 2805S	MSA 2812S MSA 2812D MTR 283R3SF MTR 2805SF MSA 2805S		
Highest Temperature (respect to ambient air)	6°C	≤8°C	3°C	6°C
Conditions	PDM is power on	Respective payload is on	Respective payload is on	Respective payload is on
Remarks		MSA 2805S and MTR 2805SF are the hottest; while MTR 283R3SF temperature is the lowest	All the other switching MOSFETs are of similar characteristics but at lower temperature	All the other sense resistors are of similar characteristics but at lower temperature

Table 6.4 Summary of PDM boards thermal profiles.

PDM DC-DC

Fig. 6.23 PDM DC-DC converters temperature profiles.

MOSFET & Resistor

Fig. 6.24 PDM switching MOSFETs and sensing resistors temperature profiles.

6.7 Conclusions

Both component level and sub-system level tests have been conducted for the power supply system. The sub-system level tests include the whole orbit tests and the thermal cycle tests. The developed integrated testing system simulates the orbit scenario successfully and provides useful information for analysis. The test results show that the designed power supply system is capable of meeting the power budget requirements of all mission operations throughout the three years' mission life. Moreover, the power supply system works at high power efficiency and solar energy utilization factor.

Chapter 7

Conclusions and recommendations

7.1 Conclusions

The design and performance evaluation of the power supply system for a micro-satellite have been presented in this thesis. The design process includes three steps: system topology study and design, integrated testing system development and system performance evaluation.

In this study, a new power supply system architecture has been proposed and developed. The proposed model method adopts the sun-regulated direct energy transfer topology with the shunt regulators being eliminated. Moreover, a main bus regulator is introduced to regulate the main bus voltage. As compared to the conventional DET topology, the battery latch-up problem during the cold solar array temperature environment has been eliminated. Thus, maximum solar energy is utilized to recharge the battery. The experimental results show that the main bus voltage is regulated within the required range. Moreover, the satellite power balance could be achieved throughout the mission life.

To evaluate the performance of the developed system, an integrated testing system has been developed using virtual instrument concept. Based on the instrument requirements, a solar cell model has been proposed and implemented. The proposed model estimates

the solar cell voltage and current parameters based on the temperature and incident angle factors. The satellite operation scenarios have been simulated successfully. The developed system has been proven that it is capable of providing valuable information of the power supply system.

A lithium-ion battery SOC estimation method has been proposed. This method first calculates the battery charge/discharge rate factors based on the internal resistances values. The battery SOC is then determined from the voltage, current and the rate factors in real-time. The proposed method supports continuous varying load power situation. It can also be integrated into a fully automated virtual instrument system, which can not be achieved by conventional methods. The estimated results have been validated with the manufacturer data and proven to be accurate.

The complete power supply system which includes the power control module, the power distribution module, solar array and lithium-ion battery has been integrated and tested for whole orbit tests under different loading scenarios. Functionality tests of the switching circuit, over current protection circuit and redundant circuits have been conducted to ensure the system reliability. Thermal cycling tests have also been conducted. The experimental results have shown that the power supply system is capable of meeting the satellite mission requirements.

7.2 Recommendations for future works

Presently all the satellite position data including the incident angle and environmental temperature are generated by the ADCS system at a time step of 1 second and stored in the database. The EGSS reads and extracts the necessary data from the database. By integrating the EGSS with the ADCS system, all the input data can be transferred directly at a much higher speed without being logged. Thus, the satellite orbit scenarios could be simulated in real-time. Moreover, the output of the solar array simulator could be more accurate and the performance analysis could be more realistic.

The level of the solar flux varies over different seasons of the year. In the present solar cell model, the flux effect on the solar cell parameters is not considered. It could be useful to study this effect and include it in the model. With the knowledge of the solar flux level on the proposed satellite launch date, the LEOP stage tests and analysis could provide more valuable information.

Both the temperature and aging effects have been neglected in the present battery model. To make the estimation results more precise, the influence of temperature and aging on battery internal parameters should be included. Moreover, the initial battery state of charge of the present model is assumed to be 1. This assumption is made based on the fact that the battery will be fully charged before the satellite is launched. However, this may not be true during the satellite design and testing phase. Thus, a method that determines the initial battery state of charge needs to be developed in the future.

Author's Publications

1. Low, K S and Zheng, C, "Virtual Instrument for a Micro-satellite Power Supply System", *IEEE Instrumentation and Measurement Conference (IMTC'07)*, 1-3 May 2007, Warsaw, Poland, Special session on "Space-Frontiers of measurement".
2. Chao Zheng, Kay-Soon Low, "A Virtual Instrument for the Testing and Performance Evaluation of a Micro-satellite Power Supply System", *IEEE Trans. Instrum. Meas.*, Accepted for special issue on IMTC 2007

References

- [1] Z. F. Wu, Z. You and J. Salvignol, "Tsinghua-1 Micro-satellite power system architecture and design," *Proceedings of the 3rd International Power Electronics and Motion Control Conference, Beijing, China, vol.3, pp. 1045–1049, 15-18 Aug. 2000*
- [2] A. Falke, M. Hartling, R. Laufer and H. P. Roeser, "Implementation of a simulation environment for software & hardware simulation of power management for a lunar small satellite mission," *55th International Astronautical Congress, Vancouver, Canada, 4-8 Oct. 2004*
- [3] Sweeting, M. N., "UoSAT microsatellite missions," *Electronics & Communication Engineering Journal, vol. 4, issue 3, pp.141-150, Jun. 1992*
- [4] Van der Zel, V.M.; Blewett, M.J.; Clark, C.S.; Hamill, D.C , "Three generations of DC power systems for experimental small satellites," *Applied Power Electronics Conference and Exposition, vol. 2, pp.664 – 670, 3-7 Mar. 1996*
- [5] Da Silva C. Ward and Sweeting, "Space at Surrey: Recent successes and future missions," *22nd International Symposium on Science and Technology, Morioka, Japan, May 2000*
- [6] C. S. Clark, A. H. Weinberg and A. Lopez-Mazarius, "A Versatile power system: Scalable from 50W LEO to 2KW GEO," *1st International Energy Conversion Engineering Conference, Portsmouth, Virginia , 17-21 Aug. 2003*
- [7] D. Olsson, " A power system for a micro-satellite," *Proceedings of the European Space Power Conference, pp. 351-356, 23-27 Aug. 1993*

- [8] D. O'Sullivan, "Satellite power system topologies," *ESA Journal*, vol. 13, no. 2, pp. 77-88, 1989
- [9] J. Z. Slowikowski and M. J. Blewett, "Power supplies, conditioning and distribution on UoSAT," *The Radio and Electronic Engineer*, vol. 52, no. 8/9, pp. 425-430, Aug./Sep. 1982
- [10] H. S. Wong and M. J. Blewett, "The UoSAT-2 satellite power system," *Journal of IERE*, vol. 57, no. 5 (Supplement), pp. S.116-S.122, Sep./Oct. 1987
- [11] M. R. Patel, "Satellite power systems," *CRC press*, 2005
- [12] T. Noguchi, S. Togashi and R. Nakamoto, "Short-current pulse-based maximum-power-point tracking method for multiple photovoltaic-and-converter module system", *IEEE Trans. on Industrial Electronics*, vol. 49, no. 1, pp. 217-223, Feb. 2002
- [13] M. A. S. Masoum, H. Dehbonei and E. F. Fuchs, "Theoretical and experimental analyses of photovoltaic systems with voltage- and current-based maximum power-point tracking", *IEEE Trans. on Energy Conversion*, vol. 17, no. 4, pp. 514-522, Dec. 2002
- [14] R. Leyva, C. Alonso, I. Queinnec, A. Cid-pastor, D. Lagrange and L. M. Salamero, "MPPT of photovoltaic systems using extremum-seeking control", *IEEE Trans. on Aerospace and Electronic Systems*, vol. 42, no. 1, pp. 249-258, Jan. 2006
- [15] E. Koutroulis, K. Kalaitzakis, and N. C. Voulgaris, "Development of a microcontroller-based, photovoltaic maximum power point tracking control system", *IEEE Trans. on Power Electronics*, vol. 16, no. 1, pp. 46-54, Jan. 2001

- [16] P. T. Huynh and B. H. Cho, "Design and analysis of a regulated peak-power tracking system," *IEEE Trans. on Aerospace and Electronic Systems*, vol. 35, no. 1, pp. 182-190, Jan. 1999
- [17] N. Mutoh, M. Ohno and T. Inoue, "A method for MPPT control while searching for parameters corresponding to weather conditions for PV generation systems," *IEEE Trans. on Industrial Electronics*, vol. 53, no. 4, pp. 1489-1494, Aug. 2006
- [18] Z. H. Jiang, S. Y. Liu, and R. A. Dougal, "Design and testing of satellite power systems using VTB", *IEEE Trans. on Aerospace and Electronic System*, vol. 39, no. 3, pp. 1295-1304, Jul. 2003
- [19] J. R. Lee, B. H. Cho, S. J. Kim, and F. C. Lee, "Modeling and simulation of satellite power systems," *IEEE Trans. on Aerospace and Electronic Systems*, vol. 24, no. 3, May 1998
- [20] A. Falke, M. Hartling, R. Laufer and Hans-Peter Roeser, "Implementation of a simulation environment for software & hardware simulation of power management for a lunar small satellite mission," *55th International Astronautical Congress, Vancouver, Canada*, pp. 1-11, 4-8 Oct. 2004
- [21] D. Loche, "POWER SYSTEMA: The next generator of power simulator," *Proc. Of the 6th ESPC, Porto, Portugal*, pp. 297-304, 2002
- [22] Caldara, S.; Nuccio, S.; Spataro, C, "A virtual instrument for measurement of flicker," *IEEE Trans. Instrum. Meas*, vol. 47, pp. 1155-1158, Oct. 1998.
- [23] Ramos, H.M.G.; Pereira, J.M.D.; Viegas, V.; Postolache, O.; Girao, P.M.B.S. "A virtual instrument to test smart transducer interface modules (STIMs)," *IEEE Trans. Instrum. Meas*, vol. 53, pp. 1232-1239, Aug. 2004.

- [24] Wang, C.; Gao, R.X, "A virtual instrumentation system for integrated bearing condition monitoring," *IEEE Trans. Instrum. Meas*, vol. 49 , pp. 325–332, Apr. 2000
- [25] C. P. Young, W. L. Juang, and M. J. Devaney, "Real-time intranet-controlled virtual instrument multiple-circuit power monitoring," *IEEE Trans. Instrum. Meas.* vol. 49, no. 3, pp. 579-584, Jun. 2000
- [26] D. R. Zrudsky and J. M. Pichler, "Virtual instrument for instantaneous power measurement," *IEEE Trans. on Instrumentation and Measurement*, vol. 41, no. 4, pp. 528-534, Aug. 1992
- [27] G. Bucci, A. Germano, and C. Landi, "Virtual instrument based on real-time measurement apparatus," *Proc. IEEE IMTC'94*, pp. 357-360, May 1994
- [28] L. Cristaldi and A. Ferrero, "A virtual instrument for the determination of the electronic power quality," *Proc. IEEE IMTC'94*, pp. 1309-1312, May 1994
- [29] A. Ferrero and V. Piuri, " A simulation tool for virtual laboratory experiments in a WWW environment," *IEEE Trans. on Instruments and Measurement*, vol. 47, pp. 741-746, Jun. 1998
- [30] M. C. Smart, B. V. Ratnakumar, L. D. Whitcanack, K. B. Chin and S. Surampudi, "Lithium-Ion Batteries for Aerospace," *IEEE AESS systems magazine*, pp. 18-25, Jan. 2004
- [31] N. Miyanaga, T. Inoue, H. Yoshida and K. Komada, "Large scale lithium-ion battery cells for space use," *IEICE/IEEE INTELEC'03*, Oct. 19-23, 2003
- [32] C. Lurie, "Evaluation of lithium ion cells for space applications," *Proceedings of the 32nd Intersociety Energy Conversion Engineering Conference ,Honolulu, Hawaii*, pp. 58-63, 1997

- [33] C. M. Reid, M. A. Manzo, T. B. Miller, M. C. Smart and R. V. Bugga, "Performance and comparison of lithium-ion batteries under low-earth-orbit mission profiles," *4th International Energy Conversion Engineering Conference and Exhibit, San Diego, California, 26-29 Jun. 2006*
- [34] T. F. Fuller, M. Doyle, and J. Newman, "Simulation and optimization of the dual lithium ion insertion cell," *J. Electrochem. Soc.*, vol. 141, no. 1, pp. 1-10, 1994
- [35] M. Doyle and J. Newman, "Comparison of modeling predictions with experimental data from plastic lithium ion cells," *J. Electrochem. Soc.*, vol. 143, no. 6, pp. 1890-1903, 1996
- [36] L. Song and J. W. Evans, "Electrochemical-thermal model of lithium polymer batteries," *J. Electrochem. Soc.*, vol. 147, no. 6, pp. 2086-2095, 2000
- [37] W. B. Gu and C. Y. Wang, "Thermal-electrochemical modeling of battery systems," *J. Electrochem. Soc.*, vol. 147, no. 8, pp. 2910-2922, 2000
- [38] R. Peng and M. Pedram, "An analytical model for predicting the remaining battery capacity of lithium-ion batteries", *IEEE Trans. on Very Large Scale Integration Systems*, vol. 14, no. 5, pp. 441-451, May 2006
- [39] L. Gao, S. Liu and R. A. Dougal, "Dynamic lithium-ion battery model for system simulation", *IEEE Trans. on Components and Packaging Technologies*, vol. 25, no. 3, pp. 495-505, Sep. 2002
- [40] B. Schweighofer, K. M. Raab and G. Brasseur, "Modeling of high power automotive batteries by the use of an automated test system", *IEEE Trans. on Instrumentation and Measurement*, vol. 52, no. 4, pp. 1087-1091, Aug. 2003

- [41] S. Gold, "A PSPICE macro model for lithium-ion batteries," *Battery Conf. Appl. Adv., California*, pp. 215-222, 1997
- [42] A. Capel and P. Perol, "Comparative performance evaluation between the S4R and the S3R regulated bus topologies," *Rec. IEEE Power Electronics Specialists Conference, Vancouver, Canada*, pp. 1963-1969, 2001
- [43] G. B. Joung and K. S. Ma, "Improved model of a partial shunt regulator for satellite," *Proc. of Power Electronics and Drive Systems, Singapore*, vol. 2, pp. 896-900, May 1997
- [44] P. R. K. Chetty, W. M. Polka and R. D. Middlebrook, "Microprocessor – Controlled digital shunt regulator," *IEEE Trans. on Aerospace and Electronics Systems*, vol. AES-16, no. 2, pp. 191-201, Mar. 1980
- [45] A. Garrigos, J. A. Carrasco, J. M. Blanes and E. Sanchis, "Modeling the sequential switching shunt series regulator," *IEEE Power Electronics Letters*, vol. 3, no. 1, pp. 7-13, Mar. 2005
- [46] A. Garrigos, J. A. Carrasco, J. Rubiato, E. Avila, J. M. Blanes and E. Sanchis, "System model of the sequential switching shunt series regulator for satellite regulated high power busses," *35th Annual IEEE Power electronics Specialists Conference, Aachen, Germany*, pp. 2645-2650, 2004
- [47] S. Megahed and W. Ebner, "Lithium-ion battery for electronic applications," *J. Power Sources*, vol. 54, pp. 155-162, 1995
- [48] B. Kennedy, D. Patterson and S. Camilleri, "Use of lithium-ion batteries in electric vehicles," *J. Power Sources*, vol. 90, pp. 156-162, 2000

- [49] G. Bruce, P. Mardikian and L. Marcoux, "50 to 100 Ah lithium-ion cells for aircraft and satellite applications," *J. Power Sources*, vol. 65, pp. 149-153, 1997
- [50] M. Chen and G. A. Rincon-Mora, "Accurate electrical battery model capable of predicting runtime and I-V performance", *IEEE Trans. on Energy Conversion*, vol. 21, no. 2, pp. 504-511, Jun. 2006
- [51] D. Panigrahi, C. Chiasserini, S. Dey, R. Rao, A. Rahunathan and K. Lahiri, "Battery life estimation for mobile embedded systems," *Proc. IEEE International Conference on Computer-Aided Design, Bangalore, India*, pp. 488-193, Jan. 2001
- [52] B. A. Johnson and R. E. White, "Characterization of commercial available lithium-ion batteries," *J. Power Sources*, vol. 70, no. 1, pp. 48-54, Jan. 1998
- [53] D. E. Carlson, "Recent advances in photovoltaics," *Proceedings of the 30th Intersociety Energy Conversion Engineering Conference, ASME*, pp. 621-626, 1995
- [54] N. H. Karam et al. "Development and characterization of high efficiency GaInP/GaAs/Ge dual and triple junction solar cells," *IEEE Trans. on Electron Devices*, pp.2116-2125, Jul. 1999
- [55] S. Guha et al. "Low cost and light weight amorphous silicon alloy solar array for space applications," *Proceedings of the 34th Intersociety Energy Conversion Engineering Conference, Vancouver, Canada, Paper no. 2553*, Aug. 1999
- [56] C. Algora and V. Diaz, "Influence of series resistance on guidelines for manufacturing of concentrator p-on-n GaAs solar cells," *Progress in PV Research and Applications, John Wiley and Sons*, vol. 8, pp. 211-225, 2000

- [57] K. K. Laug and M. R. Holmes, "Paraboloidal thin-film concentrators and their use for power applications," *Proceedings of the 34th Intersociety Energy Conversion Engineering Conference, Vancouver, Canada, Paper no. 2552, Aug. 1999*
- [58] R. Anil Kumar, M. S. Suresh and J. Nagaraju, "Measurement of AC parameters of Gallium Arsenide (GaAs/Ge) solar cell by impedance spectroscopy," *IEEE Trans. on Electron Devices, vol. 48, no. 9, pp. 2111-2179, Sep. 1999*
- [59] Y. Choi and J. Wang, "Design of solar cell array preventing electrostatic discharge for satellite use," *Proceedings of the 32nd Intersociety Energy Conversion Engineering Conference, Honolulu, Hawaii, pp.416-421, 27 Jul. – 1 Aug. 1997*
- [60] M. C. Glass, "Improved solar array power point model with SPICE realization," *Proceedings of the 31st Intersociety Energy Conversion Engineering Conference, Washington D.C., pp.286-291, 11-16 Aug. 1996*
- [61] M. Santoni, "An instrument that isn't really [Laboratory virtual instrument engineering workbench]," *IEEE Spectrum, vol. 27, Issue 8, pp. 36-39, Aug. 1990*
- [62] J. Kang, Y. Gan and Q. Quan, "The method of developing virtual instrument platform", *Proceedings of International Workshop on Autonomous Decentralized Systems, Chengdu, China, pp. 64-67, 21-23 Sep. 2000*
- [63] B. K. Mcquiston, "Virtual instruments for use in test systems development", *Proceedings of IEEE Systems Readiness Technology Conference, Anaheim, California, pp. 639-642, 20-23 Sep. 1993*
- [64] D. Sakoda and J. A. Horning, "Overview of the NPS satellite architecture and technology demonstration satellite, NPSAT1," *16th Annual AIAA/USU Conference on Small Satellites, Logan, Utah, Aug. 2002*

- [65] D. Sakoda, M. Nobel and S. Paluszczek, "Preliminary design of PANSAT electric power and communication subsystems," *4th Annual AIAA/USU Conference on Small Satellites*, Logan, Utah, Aug. 1990
- [66] R. M. Sullivan, "The small Astronomy Satellite (SAS) power system", *4th Annual AIAA/USU Conference on Small Satellites*, Logan, Utah, Aug. 1990
- [67] W. Krummann, "Next generation GOES satellite power subsystem," *IEEE Aerospace and Electronic System Magazine*, vol. 16, pp. 23-28, Apr. 2001
- [68] W. Krummann, "The power subsystem for the next generation GOES satellite," *35th Intersociety Energy Conversion Engineering Conference and Exhibit*, Las Vegas, Nevada, vol. 1, pp. 207-213, 24-28 Jul. 2000
- [69] V. Glemba, L. Lazko and O. Lazko, "Power subsystem design for the Lviv Institute Space Apparatus (LISA) nanosat class satellite," *Proceedings of the International Conference on Modern Problems of Radio Engineering*, Lviv-Slavsko, Ukraine, pp. 593-594, Feb. 2004
- [70] A. G. Ritchie, "Recent developments and likely advances in lithium rechargeable batteries," *Journal of Power Sources*, vol. 136, pp. 285-289, 2004
- [71] S. Pang, J. Farrell, J. Du and M. Barth, "Battery state-of-charge estimation," *Proceedings of American Control Conference*, vol. 2, pp. 1644-1649, Jun. 2001
- [72] J. L. Stone, "Photovoltaics: unlimited electrical energy from the sun," *Physics Today*, vol. 46, Issue 9, pp. 22-29, Sep. 1993
- [73] S. J. Kim, J. R. Lee and B. H. Cho, "Large-signal analysis of spacecraft power systems", *Proceedings of the 24th Intersociety Energy Conversion Engineering Conference*, vol. 6, pp. 2873-2880, Aug. 1989

Appendix A PDM payloads operation in eclipse period of *eclipse data download* scenario

Start Time	End Time (s)	ΔT	RW_X	RW_Y	RW_Z	RW_W	MTMP	MTMS	STS	ADCS	TTCTX	TTCTX2	XBT RF	XBTSSIRIS_SHIRIS_PSM	PPU	SSR	OBC1OBC2GPS_AGPS_D	IIC	PPUC	TTCC1	TTCC2	XBTC1XBTC2	PDMC1	PDMC2	SS
0	50	50	1	1	1	1	2	2	1	1	0	0	0	0	0	1	1	0	0	0	1	0	0	1	1
50	100	50	1	1	1	1	2	2	1	1	0	0	0	0	0	1	1	0	0	0	1	0	0	1	1
100	150	50	1	1	1	1	2	2	1	1	0	0	0	0	0	1	1	0	0	0	1	0	0	1	1
150	200	50	1	1	1	1	2	2	1	1	0	0	0	0	0	1	1	0	0	0	1	0	0	1	1
200	250	50	1	1	1	1	2	2	1	1	1	0	0	0	0	1	1	0	0	0	1	0	0	1	1
250	300	50	1	1	1	1	2	2	1	1	1	0	0	0	0	1	1	0	0	0	1	0	0	1	1
300	350	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
350	400	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
400	450	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
450	500	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
500	550	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
550	600	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
600	650	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
650	700	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
700	750	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
750	800	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
800	850	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
850	900	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
900	950	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
950	1000	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
1000	1050	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
1050	1100	50	1	1	1	1	2	2	1	1	1	0	1	0	0	1	1	0	0	0	1	1	1	1	1
1100	1150	50	1	1	1	1	2	2	1	1	1	0	0	0	0	1	1	0	0	0	1	0	0	1	1
1150	1200	50	1	1	1	1	2	2	1	1	1	0	0	0	0	1	1	0	0	0	1	0	0	1	1
1200	1250	50	1	1	1	1	2	2	1	1	0	0	0	0	0	1	1	0	0	0	1	0	0	1	1

1250	1300	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1300	1350	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1350	1400	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1400	1450	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1450	1500	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1500	1550	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1550	1600	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1600	1650	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1650	1700	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1700	1750	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1750	1800	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1800	1850	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1850	1900	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1900	1950	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
1950	2000	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1
2000	2050	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	1	1	0	0	1	1	1	1

Appendix B PDM payloads operation in sunlight period of *IRIS* scenario

Start Time	End Time	ΔT (s)	RW_X	RW_Y	RW_Z	RW_W	MTMP	MTMS	STS	ADCS	TTCTX1	TTCTX2	XBT_RF	XBT_SS	IRIS_SH	IRIS_PS	PPU_M	SSR	OBC_1	OBC_2	GPS_A	GPS_D	IIC	PPU_C	TTCC1	TTCC2	XBT_C1	XBT_C2	PDM_C1	PDM_C2	PDM_S
0	50	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
50	100	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
100	150	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
150	200	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
200	250	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
250	300	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
300	350	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
350	400	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
400	450	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
450	500	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
500	550	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
550	600	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
600	650	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
650	700	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
700	750	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	1	1	1	1	1
750	800	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
800	850	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
850	900	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
900	950	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
950	1000	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
1000	1050	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
1050	1100	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
1100	1150	50	1	1	1	1	2	2	1	1	0	0	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0
1150	1200	50	1	1	1	1	2	2	1	1	1	1	0	0	0	0	0	1	1	1	1	0	0	0	2	2	0	0	0	0	0

1200	1250	50	1	1	1	1	2	2	1	1	1	1	0	0	0	0	1	1	1	1	0	0	2	2	0	0	1	1	1
1250	1300	50	1	1	1	1	2	2	1	1	1	1	1	0	0	1	0	1	1	1	0	1	2	2	0	0	1	1	1
1300	1350	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1350	1400	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1400	1450	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1450	1500	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1500	1550	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1550	1600	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1600	1650	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1650	1700	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1700	1750	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1750	1800	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1800	1850	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1850	1900	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1900	1950	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
1950	2000	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
2000	2050	50	1	1	1	1	2	2	1	1	1	1	1	1	0	0	1	0	1	1	0	1	2	2	0	0	1	1	1
2050	2100	50	1	1	1	1	2	2	1	1	1	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2100	2150	50	1	1	1	1	2	2	1	1	1	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2150	2200	50	1	1	1	1	2	2	1	1	1	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2200	2250	50	1	1	1	1	2	2	1	1	1	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2250	2300	50	1	1	1	1	2	2	1	1	1	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2300	2350	50	1	1	1	1	2	2	1	1	1	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2350	2400	50	1	1	1	1	2	2	1	1	1	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2400	2450	50	1	1	1	1	2	2	0	1	0	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2450	2500	50	1	1	1	1	2	2	0	1	0	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2500	2550	50	1	1	1	1	2	2	0	1	0	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2550	2600	50	1	1	1	1	2	2	0	1	0	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2600	2650	50	1	1	1	1	2	2	0	1	0	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1
2650	2700	50	1	1	1	1	2	2	0	1	0	1	1	0	0	0	0	1	1	1	0	1	2	2	0	0	1	1	1

2700	2750	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	0	0	0	2	2	0	0	1	1	1
2750	2800	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
2800	2850	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
2850	2900	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
2900	2950	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
2950	3000	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3000	3050	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3050	3100	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3100	3150	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3150	3200	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3200	3250	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3250	3300	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3300	3350	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3350	3400	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3400	3450	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3450	3500	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3500	3550	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3550	3600	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3600	3650	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3650	3700	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3700	3750	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3750	3800	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3800	3850	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3850	3900	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1
3900	3950	50	1	1	1	1	2	2	0	1	0	0	0	0	1	1	1	1	1	1	2	2	0	0	1	1	1

Appendix C PDM payloads operation in sunlight period of PPU scenario

Start Time	End Time (s)	RW	RW_Y	RW_Z	RW_W	MTMP	MTMS	STS	ADCS	TTCTX1	TTCTX2	XBT RF	XBTSS	IRIS	SHIRIS	PSM	PPU	SSR	OBC1	OBC2	GPS	AGPS_D	IIC	PPUC	TTCC	TTCC2	XBTC1	XBTC2	PDMC1	PDMC2	SS
0	50	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
50	100	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
100	150	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
150	200	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
200	250	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
250	300	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
300	350	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
350	400	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
400	450	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
450	500	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
500	550	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
550	600	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
600	650	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
650	700	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
700	750	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
750	800	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
800	850	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
850	900	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
900	950	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
950	1000	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
1000	1050	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
1050	1100	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1
1100	1150	1	1	1	1	2	2	0	1	0	0	0	0	0	0	0	1	1	1	1	1	0	0	1	2	2	0	0	1	1	1

[illegible]

[illegible]

Appendix D Power consumptions and operation duty cycles of satellite modules

Subsystem/ Payload	Module/ Component	Power Supply [Volts]	Eff [%/100]	Normal Current [A]	Normal Power (No Eff.) [Watts]	Actual Power (with Eff.) [Watts]	Normal - Idle				IRIS				PPU				Data Download			
							Sunlight		Eclipse		Sunlight		Eclipse		Sunlight		Eclipse		Sunlight		Eclipse	
							Duty	Power	Duty	Power	Duty	Power	Duty	Power	Duty	Power	Duty	Power	Duty	Power	Duty	Power
ADCS	Accord GPS	5	0.75	0.86	4.30	5.73	100%	5.73	100%	5.733	100%	5.73	100%	5.733	100%	5.73	100%	5.733	100%	5.733	100%	5.733
ADCS	FSSnZnX+FSStZpX	12	0.75	0.0333	0.40	0.53	100%	0.53	100%	0.533	100%	0.53	100%	0.533	100%	0.53	100%	0.533	100%	0.533	100%	0.533
ADCS	FSSpZ	12	0.75	0.0167	0.20	0.27	100%	0.27	100%	0.267	100%	0.27	100%	0.267	100%	0.27	100%	0.267	100%	0.267	100%	0.267
ADCS	FSSpX	12	0.75	0.0167	0.20	0.27	100%	0.27	100%	0.267	100%	0.27	100%	0.267	100%	0.27	100%	0.267	100%	0.267	100%	0.267
ADCS	ADCS Elect+CS15	5	0.75	0.5	2.50	3.33	100%	3.33	100%	3.333	100%	3.33	100%	3.333	100%	3.33	100%	3.333	100%	3.333	100%	3.333
ADCS	ADCS Elect	12	0.75	0.0416	0.50	0.67	100%	0.67	100%	0.666	100%	0.67	100%	0.666	100%	0.67	100%	0.666	100%	0.666	100%	0.666
ADCS	MTQX	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16
ADCS	MTQY	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16
ADCS	MTQZ	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16	10%	0.16
ADCS	MTM1	28	0.98	0.0332	0.93	0.95	100%	0.95	100%	0.949	100%	0.95	100%	0.949	100%	0.95	100%	0.949	100%	0.949	100%	0.949
ADCS	MTM2	28	0.98	0.0332	0.93	0.95	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0
ADCS	RWX	28	0.98	0.2739	7.67	7.83	50%	3.91	50%	3.913	66%	5.16	50%	3.913	50%	3.91	50%	3.913	50%	3.913	50%	3.913
ADCS	RWY	28	0.98	0.2739	7.67	7.83	50%	3.91	50%	3.913	66%	5.16	50%	3.913	50%	3.91	50%	3.913	50%	3.913	50%	3.913
ADCS	RWZ	28	0.98	0.2739	7.67	7.83	50%	3.91	50%	3.913	66%	5.16	50%	3.913	50%	3.91	50%	3.913	50%	3.913	50%	3.913
ADCS	RWW	28	0.98	0.2739	7.67	7.83	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0
ADCS	STS	28	0.98	0.1536	4.30	4.39	0%	0	0%	0	37%	1.62	0%	0	0%	0	0%	0	37%	1.624	0%	0
P_{ADCS}							23.97	23.97	23.97	23.97	29.35	23.97	23.97	23.97	23.97	23.97	23.97	23.97	25.59	23.97	23.97	23.97
OBDH	OBC1	5	0.75	1	5.00	6.67	100%	6.67	100%	6.667	100%	6.67	100%	6.667	100%	6.67	100%	6.667	100%	6.667	100%	6.667
OBDH	OBC2	5	0.75	1	5.00	6.67	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0
OBDH	OIM1	5	0.75	0.7	3.50	4.67	100%	4.67	100%	4.667	100%	4.67	100%	4.667	100%	4.67	100%	4.667	100%	4.667	100%	4.667
OBDH	OIM2	5	0.75	0.7	3.50	4.67	100%	4.67	100%	4.667	100%	4.67	100%	4.667	100%	4.67	100%	4.667	100%	4.667	100%	4.667
OBDH	OIM3	5	0.75	0.26	1.30	1.73	100%	1.73	100%	1.733	100%	1.73	100%	1.733	100%	1.73	100%	1.733	100%	1.733	100%	1.733
OBDH	SSR	3.3	0.75	0.8	2.64	3.52	100%	3.52	100%	3.52	100%	3.52	100%	3.52	100%	3.52	100%	3.52	100%	3.52	100%	3.52
OBDH	SSR	3.3	0.75	1.92	6.34	8.45	50%	4.22	50%	4.224	75%	6.34	75%	6.336	100%	8.45	100%	8.448	75%	6.336	75%	6.336
Результат							25.48	25.48	25.48	25.48	27.59	27.59	27.59	27.59	29.70	29.70	29.70	29.70	27.59	27.59	27.59	27.59

PSS	PCM		28	0.98	0.107	3.00	3.06	100%	3.06	100%	3.057	9.06	100%	3.06	100%	3.057	100%	3.057	100%	3.057
PSS	PDM		5	0.75	0.7	3.50	4.67	100%	4.67	100%	4.667	9.06	100%	4.67	100%	4.667	100%	4.667	100%	4.667
PSS	PDM CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.67	100%	0.667	9.06	100%	0.67	100%	0.667	100%	0.667	100%	0.667
PSS	PDM CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.67	100%	0.667	9.06	100%	0.67	100%	0.667	100%	0.667	100%	0.667
P_{PSS}																				
TTC	CCSDSI		5	0.75	0.7	3.50	4.67	100%	4.67	100%	4.667	9.06	100%	4.67	100%	4.667	100%	4.667	100%	4.667
TTC	TRX1		5	0.75	0.5	2.50	3.33	100%	3.33	100%	3.333	9.06	100%	3.33	100%	3.333	100%	3.333	100%	3.333
TTC	TTX1		5	0.75	0.8	4.00	5.33	23%	1.2	0%	0	23%	1.2	0%	0	23%	1.2	0%	0	0
TTC	TTX1DRV		10	0.75	0.25	2.50	3.33	23%	0.75	0%	0	23%	0.75	0%	0	23%	0.75	0%	0	0
TTC	TTX1PWR		10	0.75	0.9	9.00	12.00	23%	2.7	0%	0	23%	2.7	0%	0	23%	2.7	0%	0	0
TTC	CCSDS2		5	0.75	0.7	3.50	4.67	100%	4.67	100%	4.667	9.06	100%	4.67	100%	4.667	100%	4.667	100%	4.667
TTC	TRX2		5	0.75	0.5	2.50	3.33	100%	3.33	100%	3.333	9.06	100%	3.33	100%	3.333	100%	3.333	100%	3.333
TTC	TTX2		5	0.75	0.8	4.00	5.33	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0
TTC	TTX2 DRV		10	0.75	0.25	2.50	3.33	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0
TTC	TTX2 PWR		10	0.75	0.9	9.00	12.00	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0
TTC	TTC CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.67	100%	0.667	9.06	100%	0.67	100%	0.667	100%	0.667	100%	0.667
TTC	TTC CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.67	100%	0.667	9.06	100%	0.67	100%	0.667	100%	0.667	100%	0.667
P_{Prc}																				
XBT	MM		5	0.75	1.35	6.75	9.00	0%	0	0%	0	21.98	17.33	0	0%	0	21.98	17.33	0	17.33
XBT	MM		-5	0.75	0.5	2.50	3.33	0%	0	0%	0	1.8	0	0%	0	0	20%	1.8	0%	0
XBT	MM		12	0.75	0.7	8.40	11.20	0%	0	0%	0	0.67	0	0%	0	0	20%	0.667	0%	0
XBT	AM		-15	0.75	0.13	1.95	2.60	0%	0	0%	0	2.24	0	0%	0	0	20%	2.24	0%	0
XBT	SSPA		10	0.75	5	50.00	66.67	0%	0	0%	0	0.52	0	0%	0	0	20%	0.52	0%	0
XBT	SSPA		5	0.75	0.02	0.10	0.13	0%	0	0%	0	5	0	0%	0	0	7.50%	5	0%	0
XBT	XBT CAN I/F		5	0.75	0.1	0.50	0.67	0%	0	0%	0	0.01	0	0%	0	0	7.50%	0.01	0%	0
XBT	XBT CAN I/F		5	0.75	0.1	0.50	0.67	0%	0	0%	0	0.67	0	0%	0	0	100%	0	0%	0
XBT	XBT CAN I/F		5	0.75	0.1	0.50	0.67	0%	0	0%	0	0.67	0	0%	0	0	100%	0	0%	0
P_{SMT}																				
P_{Bus}																				
DLR GPS	DLR GPS		5	0.75	0.2	1.00	1.33	0%	0	0%	0	0.00	0	0%	0	0	89.4468229	0	0%	0
IRIS	IIC Electronics		5	0.75	0.22	1.10	1.47	0%	0	0%	0	0.29	0	0%	0	0	0	0	0%	0
IRIS	IRIS PCU		28	0.98	0.75	21.00	21.43	0%	0	0%	0	4.29	0	0%	0	0	0	0	0%	0
IRIS	IRIS EOU		28	0.98	0.143	4.00	4.09	0%	0	0%	0	0.82	0	0%	0	0	0	0	0%	0

TCS	IRIS Heaters	28	0.75	0.536	15.01	20.01	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0
PPU	PPU CAN I/F	5	0.75	0.2	1.00	1.33	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0
PPU	PPU	3.3	0.75	6.06	20.00	26.66	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0	0%	0
P _{AVG LOAD}							0.00	0.00	0.00	5.40	0.00	0.00	0.00	21.56	0.00	0.00	0.00	0.00	0.00	0.00
P _{TOTAL}							80.48	75.83	98.60	77.95	106.27	80.06	89.45	77.95						

Appendix D (Cont'd) Power consumptions and operation duty cycles of satellite modules

Subsystem/ Payload	Module/ Component	Power Supply [Volts]	Eff. [%/100]	Normal Current [A]	Normal Power (No Eff.) [Watts]	Actual Power (with Eff.) [Watts]	DLR GPS				LEOP			
							Sunlight		Eclipse		Sunlight		Eclipse	
							Duty	Power	Duty	Power	Duty	Power	Duty	Power
ADCS	Accord GPS	5	0.75	0.86	4.30	5.73	100%	5.733	100%	5.733	0%	0	0%	0
ADCS	FSSnZnX+FSStZpX	12	0.75	0.0333	0.40	0.53	100%	0.533	100%	0.533	100%	0.533	100%	0.533
ADCS	FSSpZ	12	0.75	0.0167	0.20	0.27	100%	0.267	100%	0.267	100%	0.267	100%	0.267
ADCS	FSSpX	12	0.75	0.0167	0.20	0.27	100%	0.267	100%	0.267	100%	0.267	100%	0.267
ADCS	ADCS Elect+CS15	5	0.75	0.5	2.50	3.33	100%	3.333	100%	3.333	100%	3.333	100%	3.333
ADCS	ADCS Elect	12	0.75	0.0416	0.50	0.67	100%	0.666	100%	0.666	100%	0.666	100%	0.666
ADCS	MTQX	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	100%	1.6	100%	1.6
ADCS	MTQY	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	100%	1.6	100%	1.6
ADCS	MTQZ	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	100%	1.6	100%	1.6
ADCS	MTM1	28	0.98	0.0332	0.93	0.95	100%	0.949	100%	0.949	100%	0.949	100%	0.949
ADCS	MTM2	28	0.98	0.0332	0.93	0.95	0%	0	0%	0	0%	0	0%	0
ADCS	RWX	28	0.98	0.2739	7.67	7.83	50%	3.913	50%	3.913	50%	3.913	50%	3.913
ADCS	RWY	28	0.98	0.2739	7.67	7.83	50%	3.913	50%	3.913	50%	3.913	50%	3.913
ADCS	RWZ	28	0.98	0.2739	7.67	7.83	50%	3.913	50%	3.913	50%	3.913	50%	3.913
ADCS	RWW	28	0.98	0.2739	7.67	7.83	0%	0	0%	0	0%	0	0%	0
ADCS	STS	28	0.98	0.1536	4.30	4.39	0%	0	0%	0	0%	0	0%	0
P _{ADCS}							23.97		23.97		22.55		22.55	
OBDH	OBC1	5	0.75	1	5.00	6.67	100%	6.667	100%	6.667	100%	6.667	100%	6.667
OBDH	OBC2	5	0.75	1	5.00	6.67	0%	0	0%	0	0%	0	0%	0
OBDH	OIM1	5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
OBDH	OIM2	5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
OBDH	OIM3	5	0.75	0.26	1.30	1.73	100%	1.733	100%	1.733	100%	1.733	100%	1.733
OBDH	SSR	3.3	0.75	0.8	2.64	3.52	100%	3.52	100%	3.52	0%	0	0%	0
OBDH	SSR	3.3	0.75	1.92	6.34	8.45	50%	4.224	50%	4.224	0%	0	0%	0
P _{OBDH}							25.48		25.48		17.73		17.73	

Appendix D (Cont'd) Power consumptions and operation duty cycles of satellite modules

Subsystem/ Payload	Module/ Component	Power Supply [Volts]	Eff. [%/100]	Normal Current [A]	Normal Power (No Eff.) [Watts]	Actual Power (with Eff.) [Watts]	DLR GPS				LEOP			
							Sunlight		Eclipse		Sunlight		Eclipse	
							Duty	Power	Duty	Power	Duty	Power	Duty	Power
ADCS	Accord GPS	5	0.75	0.86	4.30	5.73	100%	5.733	100%	5.733	0%	0	0%	0
ADCS	FSSnZnX+FSStZpX	12	0.75	0.0333	0.40	0.53	100%	0.533	100%	0.533	100%	0.533	100%	0.533
ADCS	FSSpZ	12	0.75	0.0167	0.20	0.27	100%	0.267	100%	0.267	100%	0.267	100%	0.267
ADCS	FSSpX	12	0.75	0.0167	0.20	0.27	100%	0.267	100%	0.267	100%	0.267	100%	0.267
ADCS	ADCS Elect+CS15	5	0.75	0.5	2.50	3.33	100%	3.333	100%	3.333	100%	3.333	100%	3.333
ADCS	ADCS Elect	12	0.75	0.0416	0.50	0.67	100%	0.666	100%	0.666	100%	0.666	100%	0.666
ADCS	MTQX	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	100%	1.6	100%	1.6
ADCS	MTQY	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	100%	1.6	100%	1.6
ADCS	MTQZ	12	0.75	0.1	1.20	1.60	10%	0.16	10%	0.16	100%	1.6	100%	1.6
ADCS	MTM1	28	0.98	0.0332	0.93	0.95	100%	0.949	100%	0.949	100%	0.949	100%	0.949
ADCS	MTM2	28	0.98	0.0332	0.93	0.95	0%	0	0%	0	0%	0	0%	0
ADCS	RWX	28	0.98	0.2739	7.67	7.83	50%	3.913	50%	3.913	50%	3.913	50%	3.913
ADCS	RWY	28	0.98	0.2739	7.67	7.83	50%	3.913	50%	3.913	50%	3.913	50%	3.913
ADCS	RWZ	28	0.98	0.2739	7.67	7.83	50%	3.913	50%	3.913	50%	3.913	50%	3.913
ADCS	RWW	28	0.98	0.2739	7.67	7.83	0%	0	0%	0	0%	0	0%	0
ADCS	STS	28	0.98	0.1536	4.30	4.39	0%	0	0%	0	0%	0	0%	0
P_{ARCS}							23.97		23.97		22.55		22.55	
OBDH	OBC1	5	0.75	1	5.00	6.67	100%	6.667	100%	6.667	100%	6.667	100%	6.667
OBDH	OBC2	5	0.75	1	5.00	6.67	0%	0	0%	0	0%	0	0%	0
OBDH	OIM1	5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
OBDH	OIM2	5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
OBDH	OIM3	5	0.75	0.26	1.30	1.73	100%	1.733	100%	1.733	100%	1.733	100%	1.733
OBDH	SSR	3.3	0.75	0.8	2.64	3.52	100%	3.52	100%	3.52	0%	0	0%	0
OBDH	SSR	3.3	0.75	1.92	6.34	8.45	50%	4.224	50%	4.224	0%	0	0%	0
P_{OBH}							25.48		25.48		17.73		17.73	

PSS	PCM		28	0.98	0.107	3.00	3.06	100%	3.057	100%	3.057	100%	3.057	100%	3.057
PSS	PDM		5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
PSS	PDM CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
PSS	PDM CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
P _{PSS}								9.06	9.06		9.06		9.06		9.06
TTC	CCSDSI		5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
TTC	TRXI		5	0.75	0.5	2.50	3.33	100%	3.333	100%	3.333	100%	3.333	100%	3.333
TTC	TTXI		5	0.75	0.8	4.00	5.33	23%	1.2	0%	0	0%	0	0%	0
TTC	TTXIDRV		10	0.75	0.25	2.50	3.33	23%	0.75	0%	0	0%	0	0%	0
TTC	TTXIPWR		10	0.75	0.9	9.00	12.00	23%	2.7	0%	0	0%	0	0%	0
TTC	CCSDS2		5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
TTC	TRX2		5	0.75	0.5	2.50	3.33	100%	3.333	100%	3.333	100%	3.333	100%	3.333
TTC	TTX2		5	0.75	0.8	4.00	5.33	0%	0	0%	0	0%	0	0%	0
TTC	TTX2 DRV		10	0.75	0.25	2.50	3.33	0%	0	0%	0	0%	0	0%	0
TTC	TTX2 PWR		10	0.75	0.9	9.00	12.00	0%	0	0%	0	0%	0	0%	0
TTC	TTC CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
TTC	TTC CAN I/F		5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
P _{TTC}								21.98	21.98		17.33		17.33		17.33
XBT	MM		5	0.75	1.35	6.75	9.00	0%	0	0%	0	0%	0	0%	0
XBT	MM		-5	0.75	0.5	2.50	3.33	0%	0	0%	0	0%	0	0%	0
XBT	MM		12	0.75	0.7	8.40	11.20	0%	0	0%	0	0%	0	0%	0
XBT	AM		-15	0.75	0.13	1.95	2.60	0%	0	0%	0	0%	0	0%	0
XBT	SSPA		10	0.75	5	50.00	66.67	0%	0	0%	0	0%	0	0%	0
XBT	SSPA		5	0.75	0.02	0.10	0.13	0%	0	0%	0	0%	0	0%	0
XBT	XBT CAN I/F		5	0.75	0.1	0.50	0.67	0%	0	0%	0	0%	0	0%	0
XBT	XBT CAN I/F		5	0.75	0.1	0.50	0.67	0%	0	0%	0	0%	0	0%	0
P _{XBT}								0.00	0.00		0.00		0.00		0.00
P _{IRIS}								80.48438476	80.48438476		75.8343848		66.6770514		66.6770514
DLR GPS	DLR GPS		5	0.75	0.2	1.00	1.33	100%	1.333	100%	1.333	100%	0	0%	0
IRIS	IIC Electronics		5	0.75	0.22	1.10	1.47	0%	0	0%	0	0%	0	0%	0
IRIS	IRIS PCU		28	0.98	0.75	21.00	21.43	0%	0	0%	0	0%	0	0%	0
IRIS	IRIS EOU		28	0.98	0.143	4.00	4.09	0%	0	0%	0	0%	0	0%	0

PSS	PCM	28	0.98	0.107	3.00	3.06	100%	3.057	100%	3.057	100%	3.057	100%	3.057
PSS	PDM	5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
PSS	PDM CAN I/F	5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
PSS	PDM CAN I/F	5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
P _{PSS}														
TTC	CCSDSI	5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
TTC	TRX1	5	0.75	0.5	2.50	3.33	100%	3.333	100%	3.333	100%	3.333	100%	3.333
TTC	TTX1	5	0.75	0.8	4.00	5.33	23%	1.2	0%	0	0%	0	0%	0
TTC	TTX1 DRV	10	0.75	0.25	2.50	3.33	23%	0.75	0%	0	0%	0	0%	0
TTC	TTX1 PWR	10	0.75	0.9	9.00	12.00	23%	2.7	0%	0	0%	0	0%	0
TTC	CCSDS2	5	0.75	0.7	3.50	4.67	100%	4.667	100%	4.667	100%	4.667	100%	4.667
TTC	TRX2	5	0.75	0.5	2.50	3.33	100%	3.333	100%	3.333	100%	3.333	100%	3.333
TTC	TTX2	5	0.75	0.8	4.00	5.33	0%	0	0%	0	0%	0	0%	0
TTC	TTX2 DRV	10	0.75	0.25	2.50	3.33	0%	0	0%	0	0%	0	0%	0
TTC	TTX2 PWR	10	0.75	0.9	9.00	12.00	0%	0	0%	0	0%	0	0%	0
TTC	TTC CAN I/F	5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
TTC	TTC CAN I/F	5	0.75	0.1	0.50	0.67	100%	0.667	100%	0.667	100%	0.667	100%	0.667
P _{TTC}														
XBT	MM	5	0.75	1.35	6.75	9.00	0%	0	0%	0	0%	0	0%	0
XBT	MM	-5	0.75	0.5	2.50	3.33	0%	0	0%	0	0%	0	0%	0
XBT	MM	12	0.75	0.7	8.40	11.20	0%	0	0%	0	0%	0	0%	0
XBT	AM	-15	0.75	0.13	1.95	2.60	0%	0	0%	0	0%	0	0%	0
XBT	SSPA	10	0.75	5	50.00	66.67	0%	0	0%	0	0%	0	0%	0
XBT	SSPA	5	0.75	0.02	0.10	0.13	0%	0	0%	0	0%	0	0%	0
XBT	XBT CAN I/F	5	0.75	0.1	0.50	0.67	0%	0	0%	0	0%	0	0%	0
XBT	XBT CAN I/F	5	0.75	0.1	0.50	0.67	0%	0	0%	0	0%	0	0%	0
P _{XBT}														
P _{Bus}														
DLR GPS	DLR GPS	5	0.75	0.2	1.00	1.33	100%	1.333	100%	1.333	100%	1.333	100%	1.333
IRIS	IIC Electronics	5	0.75	0.22	1.10	1.47	0%	0	0%	0	0%	0	0%	0
IRIS	IRIS PCU	28	0.98	0.75	21.00	21.43	0%	0	0%	0	0%	0	0%	0
IRIS	IRIS EOU	28	0.98	0.143	4.00	4.09	0%	0	0%	0	0%	0	0%	0

TCS	IRIS Heaters	28	0.75	0.536	15.01	20.01	0%	0	0%	0	0%	0	0%	0
PPU	PPU CAN/UF	5	0.75	0.2	1.00	1.33	0%	0	0%	0	0%	0	0%	0
PPU	PPU	3.3	0.75	6.06	20.00	26.66	0%	0	0%	0	0%	0	0%	0
P _{PAVLOAD}							1.33		1.33	0.00		0.00		0.00
P _{TOTAL}							81.82		77.17	66.68		66.68		66.68