

Dynamic Modelling of CubeSat Project MOVE

Narayanan Krishnamurthy

Luleå University of Technology
Master Thesis, Continuation Courses
Space Science and Technology
Department of Space Science, Kiruna



Lehrstuhl für Raumfahrttechnik
Prof. Dr. rer. nat.
Ulrich Walter



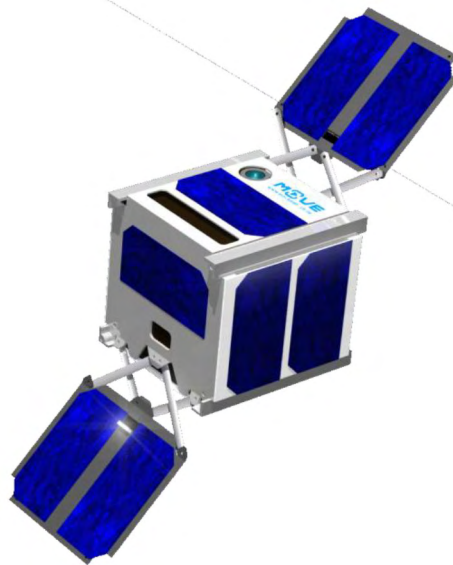
TECHNISCHE
UNIVERSITÄT
MÜNCHEN

Master Thesis

Dynamic Modelling of CubeSat Project MOVE

Author:

Narayanan Krishnamurthy



Supervisor:

Dipl.-Ing. Matthias Raif
Institute of Astronautics
Technical University of Munich



Erklärung

Ich erkläre, dass ich alle Einrichtungen, Anlagen, Geräte und Programme, die mir im Rahmen meiner Semester- oder Diplomarbeit von der TU München bzw. vom Lehrstuhl für Raumfahrttechnik zur Verfügung gestellt werden, entsprechend dem vorgesehenen Zweck, den gültigen Richtlinien, Benutzerordnungen oder Gebrauchsanleitungen und soweit nötig erst nach erfolgter Einweisung und mit aller Sorgfalt benutze. Insbesondere werde ich Programme ohne besondere Anweisung durch den Betreuer weder kopieren noch für andere als für meine Tätigkeit am Lehrstuhl vorgesehene Zwecke verwenden.

Mir als vertraulich genannte Informationen, Unterlagen und Erkenntnisse werde ich weder während noch nach meiner Tätigkeit am Lehrstuhl an Dritte weitergeben.

Ich erkläre mich außerdem damit einverstanden, dass meine Diplom- oder Semesterarbeit vom Lehrstuhl auf Anfrage fachlich interessierten Personen, auch über eine Bibliothek, zugänglich gemacht wird, und dass darin enthaltene Ergebnisse sowie dabei entstandene Entwicklungen und Programme vom Lehrstuhl für Raumfahrttechnik uneingeschränkt genutzt werden dürfen. (Rechte an evtl. entstehenden Programmen und Erfindungen müssen im Vorfeld geklärt werden.)

Ich erkläre außerdem, dass ich diese Arbeit ohne fremde Hilfe angefertigt und nur die in dem Literaturverzeichnis angeführten Quellen und Hilfsmittel benutzt habe.

Garching, den _____

Unterschrift

Name: Narayanan Krishnamurthy
Matrikelnummer:



Acknowledgement

This report is the result of the final thesis assignment that I have performed as a part of my Space Master Course study on Space Technology at Lulea University of Technology/ Paul Sabatier University of Toulouse III in cooperation with Technical University of Muenchen. This work has been done at the facilities of Lehrstule fur Raumfahrttechnik (LRT), Technical University of Muenchen, Germany. During the initial four months of my stay at LRT, I have performed the practical work of which a detailed description of the activities is provided in this thesis report. I would like to take this opportunity to thank my supervisor Dipl.-Ing. Matthias Raif.

I would like to thank Prof.Dr.rer.nat.Ulrich Walter, for providing me this opportunity and Dipl. - Ing. Manuel Czech for his support in helping me complete this thesis. I would also like to thank Prof. Dr.Victoria Barabash, Dr. Johnny Ejemalm and Dr.Christophe Peymirat for giving me constant Motivation and Encouragement as a Master thesis Examiner. Finally I'd like to thank my parents and friends for providing me necessary mental support and making this study possible. I would like to thank all the staff members who helped me while conducting the master thesis at LRT.

Munich, August 2008

Narayanan Krishnamurthy



Abstract

In the last few decades the number of small scale satellites has been in an increasing pace. The numbers of market players competing in the global scenario are highly motivated to develop satellites of micro and Pico scales, in order to reduce the time frame and project cost. However more complex task have to be performed and even more lifetime and reliability must be provided. The responsibility of an engineer starting from design to delivery (D to D) of a satellite has such an enormous complexity that it can only be solved through team work. The subsystems of a satellite consist of power (EPS), communication (COMM), On board data handling (OBDH), structure, mechanics, thermal, attitude and orbit control (AOCS), telemetry and tracking requires a multidisciplinary approach to achieve within the time span. At the Institute of Astronautics (LRT), Technical University of Munich (TUM), the Space System Concept Centre (S₂C₂) has been implemented to support the satellite design process, where the CubeSat MOVE is one of the projects under development process. Every work package is handled by different subsystem specialist. Each specialist has their own workstation. The workstations are linked such that all the users can work on the same project. MOVE is an educational project for realization of a CubeSat mission at the Institute of Astronautics. The first satellite “First-MOVE” will verify the satellite platform and new highly efficient solar cells from industry. It is a CubeSat in single-configuration (10x10x10 cm³) which features deployable solar arrays. For support of the development process, the satellite system will be modelled in a system engineering tool ((v)-Sys-ed). Also the system engineering tool can be used to dimension several design parameters like solar array size or battery size. In this thesis an already existing generic model of dimensioning the satellite is used for modelling the project MOVE, while at times the generic models needs to be either updated or changed according to the project requirements and verified with the literature references. The purpose is to keep track of the entire system parameters like mass, energy consumption, and mass memory size and so on. As these models and calculation are only based on a static state of the satellite system, it cannot simulate the dynamic behaviour of the satellite, hence it requires dynamic scenarios to be assumed and analysed. ESA’s SIMVIS tool has been used to validate the link and payload conditions through antenna and payload visibility studies along with visualization. The antenna connectivity to the ground station and the camera visibility with respect to earth are simulated using the SIMVIS tool. Further add to the validation, a similar setup is developed using the predominantly used mission analysis software called AGI Satellite Tool Kit (STK). The latter part of the thesis work is on the implementation of generic dynamic satellite models for the development of one another software called OpenSim Kit. Based on the different software approaches used, the ultimate aim is to compare the software usability and list the expectation of a spacecraft systems engineer.



Table of contents

1.1 Literature Survey	2
1.2 CubeSat	3
2.1 Environmental models	7
2.2 Dimensioning Models	8
2.3 Functional Architecture	8
2.3.1 Mass	9
2.3.2 Common Physical Properties	9
2.3.3 Power System	12
2.3.4 Communication System	13
3.1 Mission MOVE	17
3.2 MOVE Satellite	17
3.3 MOVE Orbit	18
3.4 Mission Phases	19
3.4.1 Launch and early operation Phase (LEOP)	20
3.4.2 Commissioning Phase	20
3.4.3 Experiment Phase	20
3.4.4 Disposal Phase	20
3.5 Satellite Modes	21
3.5.1 Initialization Mode	22
3.5.2 Safe Mode	22
3.5.3 Nominal Mode	22
3.6 Ground station	23
3.7 Launcher	24
4.1 Atmospheric model	26
4.2 Magnetic Field Model	26
4.3 Physical Constants	27
5.1 MOVE dimensioning model	28
5.2 Orbit Calculation	29
5.3 Electrical power system dimensioning	30
5.3.1 Battery dimensioning	30
5.3.2 Solar generator dimensioning	31
5.4 Data Storage Dimensioning	32
5.5 Communication dimensioning	32
5.5.1 Uplink dimensioning	33
5.5.2 Downlink Dimensioning	33
5.6 Attitude control system dimensioning	34



6.1 Electrical power system	36
6.1.1 EPS Module	36
6.1.2 Solar Cells	37
6.2 Communication System	39
6.2.1 VHF/UHF transceivers:	39
6.2.2 Dipole Antenna	39
6.2.3 Monopole antenna	41
6.3 Structure	41
6.4 Attitude Control System	42
6.5 OBDH	42
6.5.1 Micro Controller	42
6.6 Mechanics	43
6.7 Payload	43
6.7.1 Camera	43
6.7.2 Sun sensor	43
6.8 Conclusion	44
7.1 Introduction to SIMVIS	45
7.1.1 Designer	46
7.2 Simulation of MOVE Mission	46
7.2.1 Orbital Definition	46
7.2.2 Spacecraft Orientation in SIMVIS	48
7.2.3 Antenna and Camera Orientation	48
7.2.4 SIMVIS Designer Setting	49
7.2.5 SimSat Setup	50
7.2.6 Visualization Setup	52
7.3 Ground Station Visibility	52
7.4 Camera Visibility	53
7.5 STK Tool	53
7.5.1 Setting -Up STK	53
7.6 Results and Analysis	55
7.6.1 SIMVIS Results	55
7.6.2 STK Results	58
7.7 Conclusion	64
8.1 Mass budget	65
8.2 Power Budget	66
8.2.1 Scenario	66
8.3 Solar panel budgeting	70
8.3.1 Scenario	70
8.3.2 Solar Generator	76
8.4 Battery budgeting	78
8.4.1 Battery Cell Calculation	78



8.4.2	Battery Package	79
8.5	Link Budget	81
8.5.1	Ground Station antenna design verification	81
8.5.2	Scenario	82
8.5.3	Transceiver module	85
8.5.4	Uplink model	86
8.5.5	Downlink Model	87
8.6	Conclusion	87
9.1	Data budget	88
9.1.1	Camera design and data compression	90
9.1.2	Data Budget Analysis	92
9.2	Thermal Budget	94
9.3	Conclusion	98
10.1	Introduction	99
10.2	Reference Coordinates	99
10.2.1	Earth Centred Inertial Frame	100
10.2.2	Earth Centred Earth Fixed Coordinate system	100
10.2.3	Body Coordinate system	101
10.3	Orbit Propagator	101
10.4	Attitude Model	101
10.4.1	Direction Cosine Matrix	102
10.4.2	Euler Angle Rotation	102
10.4.3	Dynamic Equation of Motion	103
10.5	Disturbance Torques	104
10.5.1	Residual Magnetic field Torque	104
10.5.2	Aerodynamic Torque	106
10.6	Future Work with OpenSimKit	107
11.1	(v)Sys	108
11.2	SIMVIS	108
11.3	STK	109
11.4	OpenSimKit	110
12.1	Lessons Learnt	111



List of Figures

Fig. 2-1 General (v) - Sys layout	6
Fig. 2-2 (v)-Sys Illustration of MOVE Model	7
Fig. 2-3 Environmental model	7
Fig. 2-4 General Dimensional Model	8
Fig. 2-5 Power System generic layout	13
Fig. 2-6 BER vs E_b/N_o	15
Fig. 2-7 Communication Dimensioning	16
Fig. 2-8 Generic Communication Model for Satellite	16
Fig. 3-1 MOVE promotional background and official Logo	18
Fig. 3-2 Angle α	18
Fig. 3-3 Altitude vs inclination	19
Fig. 3-4 Initial Scenario design for MOVE Mission ((Römhild, 2007)	21
Fig. 3-5 Satellite Model illustration	23
Fig. 3-6 LRT Ground station, Muenchen	24
Fig. 3-7 Indian Launcher PSLV (BharatRakshak ISRO)	24
Fig. 3-8 PPODs	25
Fig. 5-1 MOVE Dimensional Model	28
Fig. 5-2 Orbital Elements(Wertz J. , 1999)	29
Fig. 6-1 Product architecture	35
Fig. 6-2 Product functional layout	35
Fig. 6-3: EPS Module Block Diagram Layout	37
Fig. 6-4 EADS/ASTRIUM Solar cell	38
Fig. 6-5 Solar-cells Layout	38
Fig. 6-6 Radiation patterns and its characteristics(Granite Island group, 2005)	40
Fig. 6-7 Illustration of Dipole antenna radiation	41
Fig. 6-8 MOVE Structural Model	42
Fig. 6-9 2 Axis Sun Sensor Layout	44
Fig. 7-1 Software Architecture of SIMVIS	45
Fig. 7-2 Mission Configuration sheet	47
Fig. 7-3 Dipole antenna Orientation	49
Fig. 7-4 Monopole Antenna Orientation	49
Fig. 7-5 Simsat Architecture	51
Fig. 7-6 Ground station setting MOVE configuration sheet	52
Fig. 7-7 SIMVIS designer	53
Fig. 7-8 STK Orbit generator	54
Fig. 7-9 SIMVIS Access study	55
Fig. 7-10 SIMVIS 30 day simulation	55
Fig. 7-11 Jan15th results	56



Fig. 7-12 Average access per day	56
Fig. 7-13 Ground Track STK 625 SSO	58
Fig. 7-14 3D view in STK	59
Fig. 7-15 Access study results in STK	59
Fig. 7-16 Access Result Verification	60
Fig. 7-17 Range plot from STK	60
Fig. 7-18 Range vs Elevation vs Time	61
Fig. 7-19 Sun Intensity vs Elevation vs time	61
Fig. 7-20 Monopole Beam Representation in STK over the Munich Ground station	62
Fig. 7-21 Monopole antenna in contact with the Munich Ground station	62
Fig. 7-22 Dipole receiver antenna beam in view with the Munich GS	63
Fig. 8-1 Mass Distribution (998.5 gms)	66
Fig. 8-2 Ground link Scenario	67
Fig. 8-3 Distribution of power consumption during Beacon/TX (2.5 W)	68
Fig. 8-4 Distribution of power consumption during Receive Mode (1 W)	68
Fig. 8-5 Distribution of power consumption during Receive (0.4W)	68
Fig. 8-6 Time distribution of satellite power modes (1440 min)	69
Fig. 8-7 Sun incident angle	71
Fig. 8-8 Satellite Orientation Scenario(Assmann, 2008)	72
Fig. 8-9 Zone A Orientation 90°	72
Fig. 8-10 Zone B 33° to 45°	73
Fig. 8-11 Satellite flip near the poles/ Sun incident angle 90°Zone C Orientation	73
Fig. 8-12 Comparison between case 1 and case 2	74
Fig. 8-13 Generated power profile in ½ revolution	75
Fig. 8-14 Case1 and (v) - Sys Results comparison	77
Fig. 8-15 Case2 and (v)-Sys results comparison	77
Fig. 8-16 General Threshold voltage characteristics with respect to time	78
Fig. 8-17: General Discharge/ charge characteristics	80
Fig. 8-18 Helical antenna Layout(Harder, 2007)	82
Fig. 9-1 Telemetry Data Budget (30.25 bytes)	90
Fig. 9-2: Focal length design layout	90
Fig. 9-3 Thermal Analysis	98
Fig. 10-1 Spacecraft model Architecture	99
Fig. 10-2: ECI Coordinate System	100
Fig. 10-3: ECEF coordinate system	101



List of Tables

Tab. 1-1 Satellite categorization with respect to mass	2
Tab. 3-1 Modes and Subsystem activity	23
Tab. 4-1 Physical Constants	27
Tab. 5-1 Orbital calculation	29
Tab. 5-2 Input parameters for dimensioning of battery	30
Tab. 5-3 Battery dimensioning calculation results	31
Tab. 5-4: Solar cell Input parameter	31
Tab. 5-5: Solar cell dimensioning results	32
Tab. 5-6: Uplink Dimensioning	33
Tab. 5-7: Downlink dimensioning	33
Tab. 7-1 With tumbling & Without Tumbling (Dipole)	57
Tab. 7-2 With tumbling & Without Tumbling (Monopole)	57
Tab. 7-3 Verified Results	63
Tab. 8-1 Mass budget	65
Tab. 8-2 Power Budget	67
Tab. 8-3 Scenario Vs Energy consumption	69
Tab. 8-4 Total energy consumption sunlight period Vs eclipse period	69
Tab. 8-5 Predicted Scenrio	70
Tab. 8-6 Case 1 with respect to satellite tumbling	71
Tab. 8-7 Case 2 Scenario with respect to the orbit/magnetic field	74
Tab. 8-8: Solar generator input parameter	76
Tab. 8-9: Solar generator output	76
Tab. 8-10: Comparison on different battery types	78
Tab. 8-11: Input parameters for Battery cell product VARTA	79
Tab. 8-12: Calculation results	79
Tab. 8-13: Battery package Input parameters	79
Tab. 8-14: battery package Results	80
Tab. 8-15: Helical Antenna Design Verification	81
Tab. 8-16: Downlink Budget	83
Tab. 8-17 Uplink Budget	84
Tab. 8-18:ISIS Transceiver Parameters	86
Tab. 8-19:Transceiver results	86
Tab. 8-20: Uplink model	86
Tab. 8-21: Downlink Model	87
Tab. 9-1: Input/ Output Budgets	88
Tab. 9-2: Telemetry packet budget	89
Tab. 9-3: Camera Data compression analysis	92
Tab. 9-4:Payload Data Budget	92



Tab. 9-5: Total Data generated per orbit	93
Tab. 9-7: Number of Pictures vs Time of transmission	93
Tab. 9-8: Memory size vs Total number of orbit data storage	94
Tab. 9-9 Thermal Budget input parameters	94
Tab. 9-10 Thermal Iteration table	97

Symbols

$\Delta x, \Delta y, \Delta z$ [m]	Translation along x,y,z
P_r [W]	Received power
S [W/m ²]	Flux density
A_r [m ²]	Effective Area
P_t [W]	Transmitter Power
B [Hz]	Bandwidth
T [K]	Noise Temperature
α [°]	Sun/Orbit plane Angle Alpha
$B(R, \lambda)$ [tesla m ³]	Earth's Magnetic Field
P [min]	Orbital Time
T_E [min]	Eclipse Time
T_D [min]	Daylight Duration
$\eta_{\max}$ [°]	Max Nadir Angle
$\lambda_{\max}$ [°]	Max Earth Central Angle
$\lambda_{\min}$ [°]	Min Earth Central Angle
$\eta_{\min}$ [°]	Min Nadir Angle
λ [m]	Wavelength
SC_{eff} [%]	Solar Cell Efficiency
I_d [%]	Inherent Degeneration
a [°]	Average sun angle
F [m]	Focal Length
R_s [m]	Spatial Resolution
R_{TX} [bps]	Transmission data rate
T_{equ} [°]	Equivalent Temperature
ϵ []	Emissivity
G_t [dB]	Transmitter Gain
G_r [dB]	Receiver Gain
L_p	Propagation Loss
f [Hz]	Frequency of communication
d [Km]	Distance between GS & SC
E_b/N_o	Energy bit per noise density
A_{drag} []	Atmospheric drag
N_{Aero} [Nm ²]	Aerodynamic Torque
ω [rad/s]	Angular Velocity
I [kgm ²]	Inertial tensor



Abbreviations

ACS	Attitude control System	NASA	National Aeronautics and Space Administration
AGI	Analytical Graphic Inc	OBC	Onboard Computer
API	Application Programming Interface	PL	Payload
ATJ	Advance Triple Junction	P-POD	Poly Pico satellite Orbital Deployer
BCR	Battery Charge Regulator	RAAN	Right Ascension of the Ascending Node
BOL	Beginning of Life	RF	Radio Frequency
BPSK	Binary Phase Shift keying	SA	Solar Array
CAD	Computer Aided Design	SIMSAT	Simulation Satellite toolkit
CMOS	Complementary metal Oxide Semiconductor	SIMVIS	Simulation and Visualization ESA's tool kit
COA	Centre of Area	SSO	Sun Synchronous orbit
COG	Centre of Gravity	STK	Satellite Tool Kit
CCD	Charge coupled Diodes	TBC	To Be Confirmed
COM	Communication System	UML	Unified modelling language
COTS	Commercial Off the Shelf	UoSAT	University of Surrey Series Satellite
EADS	European Aeronautic Defence and Space Company	VB	Visual Basics
ECEF	Earth Centred earth Fixed	VHF/UHF	Very High Frequency/ ultra High Frequency
EIRP	Effective Isotropic Radiated Power	VLSI	Very Large Scale Integrated Circuit
EOL	End of Life	WDT	Watch Dog Timer
EPS	Electrical Power System	XML	Extensible Mark-up Language
ESA	European Space Agency		
FSK	Frequency Shift Keying		
HDRM	Hold Down Return Mechanism		
ISIS	Innovative Solutions In Space		
J2	Perturbation		
LEOP	Launch And Early Operation Phase		
LPTM	Low Power Transmission		
LRT	Lehrstühle für Raumfahrt Technique		
MOVE	Munich Orbital Verification Experiment		
MPP	Maximum Peak Power		
MS	Microsoft		
MSIS	Mass Spectrometer Incoherent Scatter		



1 Introduction

Man has only had the ability to operate the spacecraft successfully since 1957. In little more than 30 years unmanned explorer spacecraft have flown past all the major bodies of the solar system. Many countries have the capability of putting spacecraft into orbits; satellites have now established a firm foothold as a part of the infrastructure of the society (Peter Fortescue, 2002). There is every expectation that they have much more to offer in the future. The CubeSat program's primary mission is to provide access to space for small payloads. CubeSats are cube shaped Pico-satellites with a nominal length of 10cm per side. The mass of the CubeSat weighs not more than 1kg. The centre of mass must be 2cm within the geometric centre. The structure of the CubeSat must be strong enough to survive the maximum loading and cumulative loading of test and launch.

MOVE is an educational project for realization of a CubeSat mission at the "Institute of Astronautics at TUM". The first satellite "First-MOVE" will verify the satellite platform and new highly efficient solar cells from industry. It is a CubeSat in single-configuration (10x10x10 cm³) which features deployable solar arrays. This design improves the performance of such small CubeSats in several ways. The MOVE project is on since summer 2006. Launch is planned for the spring of the year 2009. As a part of this thesis work, and to support the development process, the satellite system has been modelled in a system engineering tool (v)-Sys. The purpose has been to keep track of the product architecture and entire system parameters like mass, energy consumption, and mass memory size and so on. Also the system engineering tool has been used to dimension several design parameters like solar array size, battery size, link budgets and data budgets. The prediction of the behaviour is important to verify some of the system or mission requirements in an early stage of the system development.

At ESA a toolbox named SIMVIS is used to transform the static model into a dynamic simulation and visualize the simulation results. The overall architecture of SIMVIS includes a designer to define a simulation, and a Run-Time Environment SIMSAT to execute it. Data generated by the simulation can be visualized at run-time using External Visualization Applications that connect to an external simulator Application Programming Interface (API) OpenGL. The interface to the concurrent engineering process is through a Workbook, which is maintained using Microsoft Excel. For the mission MOVE, the SIMVIS is used to simulate the communication link access and the visibility of the camera experiment. The results of the simulation are evaluated using additional software AGI-STK Satellite Tool Kit.

As a part of the development process, the final part of the thesis work will include the development of generic dynamic model for space missions, since the Institute of Astronautics is developing dynamic modelling software as a part of their research catalogue. This work comprises the developed for the dynamic models of the orbit and attitude behaviour and power subsystem simulation. Based on the user friendliness, performance and comforts, the tools are compared in this proposal.

1.1 Literature Survey

Satellite projects have historically been large and expensive operations. Because of this, reducing cost and size of satellites have become issues in engineering. Space industries, however, has moved towards larger, more elaborative spacecraft, because this may reduce launch costs and increase the longevity of space investments. But the technology of small satellites is not new (Parasher, 2003). The space age began in 1957 with the launch of Sputnik1 weighing only 75 kg. The need for ever-larger more capable and more complex satellites lead to bigger satellites (Sputnik 2 already weighted about 400 kg), limited initially by launcher capability and later by finance (M.N.Sweeting, Uosat Microsatellite Missions, 1992).

The prohibitively high costs of these space projects have constrained actual access to space to a handful of nations and international agencies. Research institutions are trying to reach against this trend by developing small satellites. This is mainly possible because state-of-the-art Very Large Scale Integrated circuits (VLSI), microelectronics (MEMS) enable very sophisticated functions to be achieved without having the restriction of small mass, volume and power (M.N.Sweeting, Uosat Microsatellite Missions, 1992). There has been considerable confusion whether a particular satellite is 'large' or 'small' as the definition varies, but the following classification has become widely accepted:

Tab. 1-1 Satellite categorization with respect to mass

Class	Mass
large satellite	> 1000 kg
small satellite	500 - 100 kg
Mini-satellite	100 - 500 kg
Micro-satellite	10 - 100 kg
Nano-satellite	< 10 kg
Pico-satellite	< 2 kg

The ability to construct sophisticated yet inexpensive microsatellites must also be matched by a correspondingly inexpensive method of launch into orbit. These days, small satellites can be launched inexpensively as auxiliary payloads complementing, or sometimes providing an alternative to the high-cost traditional satellites (M.N.Sweeting, Uosat Microsatellite Missions, 1992).

At present, small satellites are being proposed for numerous commercial and military missions as a very cost-effective and quick-response adjunct to large satellite programs (University A. , 2007). There has been a common believe that with satellites weighing less than some 250 kg, nothing useful can be done. This may be true for missions requiring massive instruments or high-power communications payloads, but there remain many tightly focused or forte missions where micro/nano-satellites can be effectively used. These satellites can offer a very-quick turn around and inexpensive means of exploring well-focused, small-scale science objectives, like monitoring the space radiation environment, updating the international geomagnetic reference field or providing an early proof-of-concept prior to the development of large scale instrumentation. The University of Surrey Series Satellite



(UoSAT) missions demonstrated the possibility to launch a small satellite within 12 months from start of concept, costing less than 2 million US \$ (M.N.Sweeting, Cost Effective Microsatellite for Low Earth Orbit Communications, 1994). Until now eleven UoSAT microsatellites being part of this project have been launched into orbit, carrying a wide range of payloads covering topics like satellite communications, space science, Remote sensing and in-orbit technology demonstration of payloads (M.N.Sweeting, Uosat Microsatellite Missions, 1992). One other very important issue of these missions was to conduct in-orbit tests of modern VLSI devices in space radiation. In general, major areas of work for micro- and nano-satellites are:

- Specialized Communications
- Small-scale Space Science
- Remote Sensing
- Technology Demonstration
- Education & Training

Satellites generally must operate with limited power budgets and in a space radiation environment that is detrimental to the reliability of semiconductor micro-electronics (Cpt. Anthony, 2002). Microsatellites and nano-satellites mostly operate at Low Earth Orbits (LEO). The altitudes of these orbits lie between 600 km and 1000 km. Below 600 km the orbital lifetime before the satellite re-enters the Earth's atmosphere because of friction is too limited and unpredictable.

1.2 CubeSat

At the 2nd University Space System Symposium (USSS) held in Hawaii in November 1999, the idea of a CubeSat was proposed by Professor Robert Twiggs from Stanford University. A CubeSat is a so-called S3-SAT (Student, Space, Study Satellite), having a cubic shape measuring less than 10 cm and weighing less than 1 kg. In November 2004, about a handful of these educational satellites were already put into space. Being educational satellites, the main mission objective of CubeSat is to use them for education purposes, learning how to build and control satellites, to enhance the student engineering skills and to carry out scientific data gathering. But commercial space experiments can be conducted with these CubeSats as well. These days the CubeSat Initiative is a global congregation of universities and private companies from all over the world, trying to advance small satellite technology. CubeSat heavily rely on commercial-of-the-shelf (COTS) components, reducing cost of one satellite to 40.000 US\$ or even less, launching a CubeSat costs about the same amount of money. The satellites fit into the CubeSat deployer, the P-POD. This carrier device holds a number of CubeSats and releases them with spring-loaded mechanism into space. In order to launch a single CubeSat successfully in orbit together with other satellites, a number of requirements have to be fulfilled by each satellite, guaranteeing that no satellite presents any danger to neighbouring CubeSats, the deployer, or the launcher itself. General guidelines available define the following issues to be fulfilled by CubeSats (Twiggs., August 2003) (this list is not exhaustive but only a few major issues are covered):

- *All parts must remain attached to the CubeSat launch, ejection and operation. No additional space debris must be created.*
- *All satellites must be powered off during integration and launch to prevent any electrical or RF interference.*
- *CubeSats must use designated space materials approved by NASA.*



- *According to NASA standards the use of Aluminium 7075 or 6061-T6 is suggested for the main structure. If other materials are used, the thermal expansion must be similar to that of Aluminium 7075-T73 (the P-POD material).*
- *The outer surfaces of the CubeSats must be hard anodized in order to prevent wear between the sliding rails and the CubeSats*
- *There must be a time delay, on the order of several minutes to an hour, before all primary transmitters are activated. Low power beacon transmitter may be activated after deployment.*
- *Operators must provide proof of the appropriate license for frequency use.*

University of Aalborg, Denmark – AAUSAT II

This CubeSat project culminated in the launch of AAUcube on 28th April 2008. From the project website as on July 7th :(AAUSAT II, 2006)

All the project documents were available online with major reports written for several key subsystems. The project report on the communication system and the power system were of most relevance to this work. The Attitude Determination report contains brief discussions of modelling orbits, Sun position and Earth's magnetic field. The Attitude Control report contains some discussion of satellite modelling and control and disturbance torques. Both reports also include large amounts of discussion of hardware design.

University of Tokyo – Cute1.7

Cute-1.7 + APD II is the successor of the *CUTE-I nano-satellite* and *Cute-1.7 + APD* developed and built by the second generation of students of the *Tokyo Institute of Technology Matunaga Laboratory for Space System (LSS)*. The Avalanche Photo Diode sensor module, or APD, embarked on this 20cm x 15cm x 10cm nano-satellite was also developed by the *Tokyo Institute of Technology Kawai Laboratory*. (technology, S. Tokyo Institute)

This was another CubeSat that launched in 2008. It has operated successfully in orbit, taking photographs of Earth and space and transmitting these down to Earth. There is little documentation of the project readily available, none of which is of particular relevance to this project.

University of Toronto Institute for Aerospace – Space Flight Laboratory – CanX-2

This CubeSat was launched in 2008. Some documentation is available on the product architecture and its specification, but little is of relevance to this project. It is stated that during mission design, simulation of the mission was carried out using the program Satellite Toolkit (STK). This was used to examine passes over the ground station for a 25 day period to help design the communications system and the payload. (Space Flight Laboratory)

Technical University of DELFT, Netherland DELFI C³

On the 28th of April at 03:53 UTC the Delfi-C3 was successfully launched with a PSLV launch vehicle. At 06:45 UTC the first signal was received from a radio amateur in California. At 11:55 UTC the satellite signal was acquired and decoded at the TU Delft ground station, initial analysis showed the craft to be in excellent condition, all solar panels and antennas were deployed and the internal temperatures and voltages were within the ranges expected(DelfiC3). The mission website carries documents with respect to the subsystems and its specification which is of major interest. Especially with the antenna cluster and the transceiver used for the satellite mission.

2 Introduction to (v)-Sys

For many years' system modelling, process and organization architecture and concurrent engineering are among the main research areas of the Institute of Astronautics (LRT) at the Technical University of Munich. As part of research projects with aerospace companies and automotive industries a system and cost modelling tool called MuSSat (Modelling and Simulation of Satellite Systems) was developed to be used in concurrent design centres for satellite development. This tool allows developers to model a system concurrently and to simulate any implications of system changes. An interface to Excel allows the developer to model functions for sizing the system. Each component and subsystem of the satellite is represented by one model element and by one Excel file, respectively. MuSSat was developed in cooperation with EADS Astrium at Friedrichshafen/Germany in several concurrent engineering workshops, which usually lasted for one week. In course of these workshops up to ten students had to design a spacecraft for a given mission. With MuSSat the students first modelled a baseline and then investigated design alternatives.

The workshop experiences together with research work in the recent years led to a more elaborate software tool called (v)-Sys. The focus of the improvement was to generalize the modelling schema to model not only a physical description of a system or its costs, but an arbitrary user-specific abstraction of any system (e.g. functional architecture). The second improvement was to design a system object-oriented. Therefore (v)-Sys offers an object-oriented approach with diagrams similar to UML. The developer therefore is no longer limited to model a system in a specific domain (e.g. product tree or costs), but the functional architecture with functions related to requirements or components can be modelled. These modelling features were implemented along with a central database as done before with MuSSat. The database enables a version and access control for the different projects stored. Due to this central database, functionality for multi-user access and design, as required for concurrent engineering, could be implemented easily. To demonstrate the usability of (v)-Sys, it is currently used for this thesis work to model the system architecture of a CubeSat. The author started to model the product architecture by building a product tree.

Based on top level requirements and derived functional requirements for the MOVE mission, functional architectures for each subsystem had been developed in (v)-Sys. These functions have been assigned to various components, which will be implemented via (v)-Sys. The whole program is based on two types of relations, order relations and flow relations. Order relations can be translated with "consists of" or "is pieced together of" and give the horizontal order of the system resulting in a tree structure. Flow relations however describe the interaction of the different elements with each other. This may be forces, energies, and data and so on.

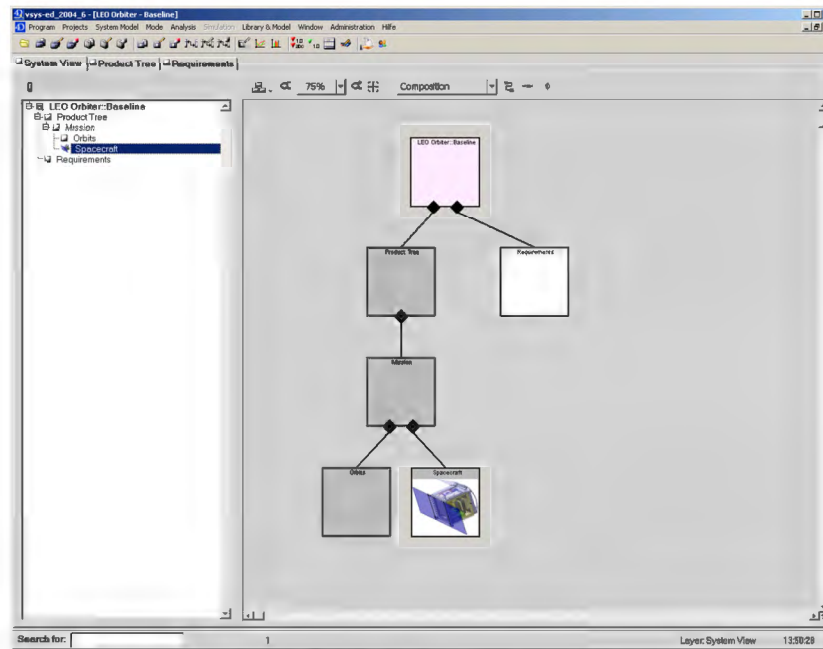


Fig. 2-1 General (v) - Sys layout

There are three different types of models we are working with:

- Dimensioning model
- Environment model
- Functional architecture

Dimensioning model contains all mathematical functions and calculations carried out for the elements in order to obtain specifying values for the elements.

Environment model consists of all kind of environmental information and values which will interact with our system.

In the functional architecture finally you can find the complete system with all its elements which contain components' attributes as well as inputs and outputs of all values needed to interact with the other elements. Through relations between elements system budgets can be calculated.

There have always been critical mission values which limit the performance of a satellite:

- power budget
- link budget
- mass budget

(v)-Sys provides a feature to automatically add up these critical values. If the property of concern is implemented with SUM as inheriting action (there are, of course, also other mathematical functions like MAX or MIN available) and the mass property has correctly been assigned to all system elements, you just have to look at the top level element and immediately get the total mass of the system.

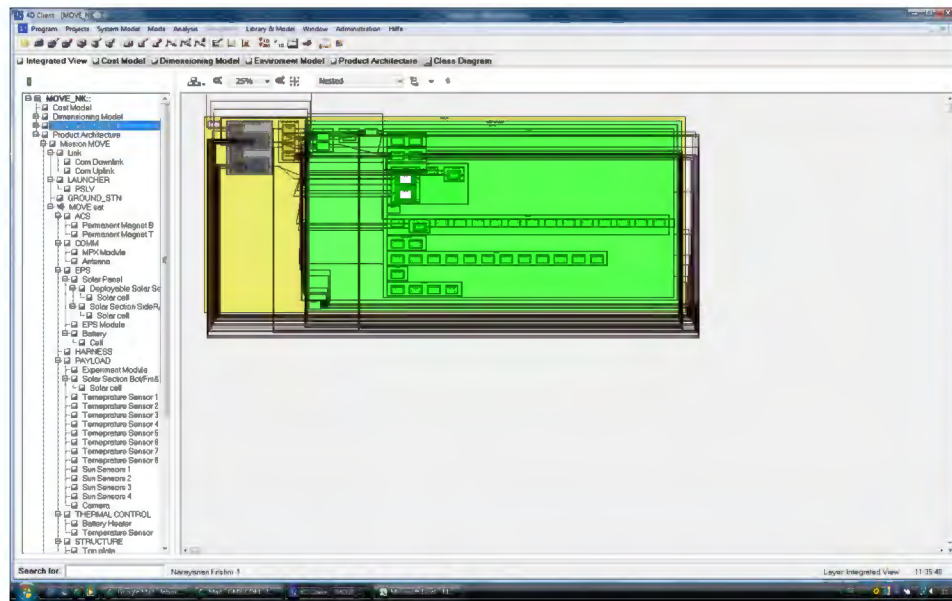


Fig. 2-2 (v)-Sys Illustration of MOVE Model

The generic satellite model consists of the cost mode, dimensioning model, environmental model, and product architecture. The topics of concern to the thesis work are the Dimensioning model, product architecture and Environmental model.

2.1 Environmental models

Environment model consists of all kind of environmental information like the atmospheric model, magnetic field model and physical constants.

This model is based upon the data of the Mass Spectrometer Incoherent Scatter MSIS atmosphere model, altitudes between the given values are linearly interpolated, and therefore they contain a certain level error. Based on the altitude chosen, the minimum and maximum atmospheric density values are derived from the tabulation (Wertz J. , 1999).

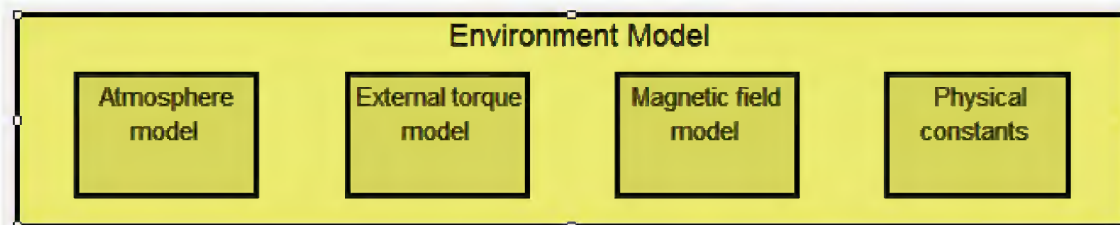


Fig. 2-3 Environmental model

The magnetic field model consists of the calculation of the maximum and minimum magnetic field strength in a satellite orbit. Within this calculation it is assumed, that the inclination of the orbit plane is equal to the magnetic latitude. (Tribble.C.Alan,2003)

The physical constants model consists of all the commonly used physical constants for the generic satellite model calculations, for example, Boltzmann constant, solar intensity, obliquity of the ecliptic, gravity constant of the Earth, Stephan Boltzmann constant etc.

2.2 Dimensioning Models

The dimensional model is the interface between the satellite mission requirement and the design. The dimensioning model gives the initial design values to decide on the product architecture. The dimensioning model sets the preliminary parameters to initiate the satellite design irrespective of the commercially available component data sheets.

To start with the design procedure for a satellite, the required subsystems are to know, for example, electrical power system, communication system, attitude control system etc. Even though the above terms comes out of the design books, each of the system needs a dimensioning to decide on the satellite application requirement.

A sample dimensioning model for a satellite is show below,

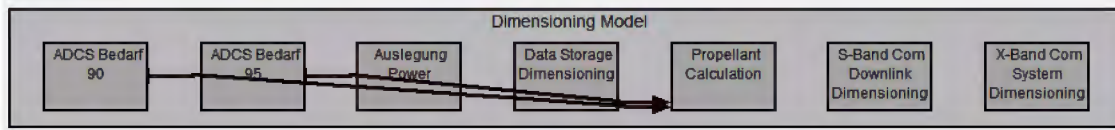


Fig. 2-4 General Dimensional Model

Since (v)-Sys is meant to be user friendly software; any dimensional model can be developed under it. As mentioned earlier, (v)-Sys is developed based an object oriented approach. Hence on creating any model, that can be defined under or referred to the parent class. Hence on doing this, the same set of calculation under each model can be reused for different mission. This makes the software compatible for any industrial system engineering application.

The calculations are designed in Excel sheets; this reduces the training or the learning period of the software. The operations used for the calculation in Excel is limited compared to other programming languages, even still MS Excel has the provision to import macros, which includes Visual Basic(VB) codes. Certain complex functions can be written in VB codes.

The dimensioning model gives the preliminary suggestion of the way in which the satellite is going to look like in terms of technical dimensions.

2.3 Functional Architecture

The functional architecture replicates the realised elements of the satellite. It carries all the critical elements of the subsystem. The functional architecture is defined in terms of three levels. The top level class is the system class, underneath is the subsystem class and the element class. Each class carries an individual functionality. Even though by vision the layout of the system is from the top level, the general way to model is from the element level.

After making the initial layout of the system architecture, the addition of the functionality will be from the element level. Since based on the mission requirement and the team's requirement, the elements are to be chosen. For example, when deciding on the electrical power system, the team may not be interested on the working of a voltage regulator or a resistor rather than the electrical distribution board. In case of requirement, that can also be done. As mentioned before the (v)-Sys is flexible to accommodate any kind of modelling.



On designing the system layout based on the order relation and the flow relation, a component database can be built for each and every critical element used. The components database is the specification data sheets from the commercially available components. (v)- Sys has the ability for carry all the required datasheet values from the component database in an Excel sheet and these values can be used for the design and design verification calculation.

Each element carries common physical properties like mass, moment of inertia, orientation, translation, centre of gravity and centre of mass etc. In the same way all the electrical and electronic system have power consumption at different phase and periodically generated data, sampling rates etc. These properties are self made by the systems engineer; the parameter set for the elements can be expanded or contracted based on the requirement. By defining the inputs and the parameters, mathematical functions can be written in form of calculation sheets in Excel and be used for the verification of the element with respect to the dimensioning models. Even though the dimensioning model and the function architecture do not have a direct relation, the values calculated out of these models can be compared analytically by the systems engineer.

When an element *object* is created from the element *class*, if on using an existing function the object is connected to the main class, whereas on making changes in the input, parameter or output definition, the objects gets disconnected from the main class. Even though this will not affect the modelling, instead the satellite model works independently. Depending on the user, the subsystem and elements can be inherited from the main class or reworked or redefined. If on reworking or redefining an object, that particular object works independently from the base class. This kind of architecture gives the user an inbuilt advantage to share among the project functions.

On the whole, the (v)-Sys is developed as a scientific calculator for any systems engineering application with built in functions and also function can be created or rewritten. There are few common functions existing in (v)-Sys that will be reused in OpenSimKit.

The common functions are the mass, inertial calculation with transformation matrices, rotational tensors, translational shift into global inertial tensor, rotation of local centre of gravity to global, and the total centre of gravity of the system. The solar torque calculation is also done in (v)-Sys. Even though the functionalities are not developed in the current work, it can be referred to the thesis work of (Hager, 2007).

2.3.1 Mass

In (v)-Sys, every component which makes a part of the product architecture has the property called mass. Since as explained earlier the (v)-Sys works on a hierarchical order. The mass parameter also sums up to get the total mass of the system in the same order.

$$M_s = \sum_{n=i} M_n \quad (2-1)$$

2.3.2 Common Physical Properties

Inertial Calculation

A Satellite consists of fewer amounts of components in a variety of shapes and structures. Each component of a satellite owns certain geometrical dimensions and has a specific mass

distribution. A rigid body can translate as well as rotate. In order to determine those movements it is important to know certain *physical properties* of a rigid body, which are its mass, its centre of gravity and its inertia properties, latter expressed in the moment of inertia matrix. During the design process the position of those components will shift causing the physical properties to change. Therefore in order to determine the position of a component one has to use *global coordinate system*, to do so on a satellite level. Every component has its own local coordinate system in which its geometry properties are expressed, e.g. the position of its centre of gravity. With knowledge of this orientation, certain *translations* and *rotations* can be performed. Furthermore every component of a satellite modelled in (v)-Sys is regarded as cuboids, and has height, length and depth. At the beginning of the design phase the origin of the global coordinate system is fixed, and afterwards the components are placed relative to this system. Often the axes of the global coordinate system are aligned with relevant orientations of the satellite. In order to calculate the physical properties of the satellite it is necessary to transform the information of every single mass element into the global coordinate system.

Based on the translation matrix and the rotation matrix, (v)-Sys converts the local coordinate system to Global coordinate system. These input elements are derived from the CATIA model. The inertial calculation and centre of gravity are calculated from each hardware element and summed up in the satellite model and given as I_{Global} and COG_{Global} .

While calculating the physical properties of an element, it is important to know the *mass distribution* of the satellite. Since everybody is inert, an object moving in free space encountering forces will respond to those forces in a certain way. One can get information about the habit of such an object, if one considers it as a spinning body and examines its *angular momentum*. Within the *moment-of-inertia matrix*, that information is summarized. The largest moment of inertia will occur about the *principal axis*. It is also important to know, were external forces will attack, and if they cause torques. To estimate whether a force creates a torque or not, the *centre of gravity* must be determined. The centre of gravity of the satellite depends on its mass distribution and therefore on the position, orientation and mass of its components. With knowledge of the geometrical size and orientation of a component, it is possible to calculate the local centre of gravity.

Translation

A translation is a movement in which all points follow the same track. Here the translation shifts the local coordinate system into the global coordinate system. One vector is needed, to point out the origin of the local coordinate system in the global coordinate system. With knowledge of this vector, a vector addition shifts any point from the local into the global coordinate system.

$$\begin{aligned} X_{global} &= X_{local} + \Delta_x \\ Y_{global} &= Y_{local} + \Delta_y \\ Z_{global} &= Z_{local} + \Delta_z \end{aligned} \quad (2-2)$$

Moment of Inertia Matrix

Just as the mass m of a body is a measure of the resistance to translational acceleration, the moment of inertia I is a measure of resistance to rotational acceleration of the body. If a mass is rotated one can express the moment of inertia as

$$I = \sum r_i^2 m_i \quad (2-3)$$

Where r_i is the radial distance from the inertia axis to the representative particle of mass m_i and where the summation is taken over all particles of the body.

The moments of inertia are always positive; products of inertia may be either positive or negative. The units of products of inertia are the same as those of moments of inertia, which are [kg*m²].

$$[I] = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix} \quad (2-4)$$

If the moment of inertia of a body is known about an axis passing through the center of mass, it may be determined about any parallel axes. This Transformation is called Parallel Axes Theorem (Wiesel, 1997).

$$I' = I + M \begin{bmatrix} \Delta y^2 + \Delta z^2 & -\Delta y \Delta x & -\Delta z \Delta x \\ -\Delta x \Delta y & \Delta x^2 + \Delta z^2 & -\Delta z \Delta y \\ -\Delta x \Delta z & -\Delta y \Delta z & \Delta x^2 + \Delta y^2 \end{bmatrix} \quad (2-5)$$

In order to get one global inertia tensor the information about the local inertia tensors need to be shifted into the global coordinate system. To use the Parallel Axes Theorem, the axes of the components local coordinate system need to be parallelized with the global coordinate system. Therefore it is necessary to perform a rotation of the local inertia tensor. This is done by

$$I_{ij,rot,global} = \sum_{k,m} A_{ik} A_{jm} I_{km,local} \quad (2-6)$$

Whereas $I_{km,local}$ is the inertia tensor of a component and A is the transformation Direction cosine matrix, consisting of 9 elements defining the orientation. The result $I_{ij,rot,global}$ is the inertia tensor of a component rotated into the global coordinate system. Now the axes are aligned, but the origin of local and global coordinate systems are not coincided yet. If the vector, giving the position of local coordinate system, in the global coordinate system is known, it is possible to use the Parallel Axes Theorem to calculate the final matrix.

$$I_{global} = I_{rot,global} + m \begin{bmatrix} \Delta y^2 + \Delta z^2 & -\Delta y \Delta x & -\Delta z \Delta x \\ -\Delta x \Delta y & \Delta x^2 + \Delta z^2 & -\Delta z \Delta y \\ -\Delta x \Delta z & -\Delta y \Delta z & \Delta x^2 + \Delta y^2 \end{bmatrix} \quad (2-7)$$

The sum over all individual moment of inertia matrices leads to the global moment of inertia matrix of the satellite. This is the function used in (v) - Sys for Inertial calculation, to transform the inertial elements from the local coordinates to the Global coordinates. This work is referred from one another student working on generic satellite models (Hager, 2007).

Centre of gravity

If a body is not seen as a dimensionless point, forces which act upon this body may not necessarily attack in the centre of gravity. It is rather producing a torque with the lever arm being the distance between centre of gravity and the point of application. In this paper the term “centre of gravity” is equally used to the term “centre of mass”. This assumption is only true, if gravity can be treated as a constant force across the rigid body (Wiesel, 1997).

Therefore it is important to know the position of the centre of gravity of the whole system. But since every component has its own centre of gravity as well as its own coordinate system there are transformations to be made. The information about the position of the centre of gravity of a component has to be transferred into the global coordinate system. This requires a rotation and a vector addition.

A vector addition finally gives the position of the local centre of gravity in the global coordinate system:

$$\begin{aligned} X_{cog,global} &= X_{cog,local,rot} + \Delta_x \\ Y_{cog,global} &= Y_{cog,local,rot} + \Delta_y \\ Z_{cog,global} &= Z_{cog,local,rot} + \Delta_z \end{aligned} \quad (2-8)$$

Whereas $\Delta\vec{s} = (\Delta x, \Delta y, \Delta z)$ is the offset of the local coordinate system in the global coordinate system. With this transformations each components centre of gravity can be shifted into the global reference frame. In order to get the unified centre of gravity of the entire system one has to calculate.

$$\begin{aligned} X &= \frac{\sum_{i=0}^n (m_i \cdot X_i)}{\sum_{i=0}^n m_i} \\ Y &= \frac{\sum_{i=0}^n (m_i \cdot Y_i)}{\sum_{i=0}^n m_i} \\ Z &= \frac{\sum_{i=0}^n (m_i \cdot Z_i)}{\sum_{i=0}^n m_i} \end{aligned} \quad (2-9)$$

While n is the number of components of the system x , y and z are the components of the vector giving the position of the total centre of gravity in the global coordinate system (Hager, 2007).

Solar Torque calculation

Unlike gravity gradient torque, magnetic torque and atmospheric drag, solar radiation pressure is not calculated in the class “Satellite” but in those components that are mainly exposed to solar radiation, that are solar arrays and panels and structural surfaces. This is comprehensible since every surface has a different reflectance factor ε and, ε has a large impact on the effect of solar radiation pressure. The vector of the components centre of area (COA) as well as its surface normal vector is given in local coordinates. Since it is the same for all satellite components, the vector giving the direction to the Sun is expressed in the global COS (Hager, 2007). The Sun angle of incident θ depends on the location of the component within the satellite and therefore is a local, component own, property. Moreover θ is a dynamic value and changes not only during one orbit but all along the whole flight of the satellite. The maximum torque will be created, if θ is zero.

2.3.3 Power System

The Power system dimensioning and the functional architecture were built on generic power system satellite model (Zeigler, 2008). The functionalities from this work have been referred in the current thesis work also. The power subsystem is divided into solar power generator, power storage and power distribution. The Power generator contains the dimensioning of the solar cell and the solar panel required for every mission. It has a main class power system,

under the main class contains the subsystem battery class, solar panel and distribution. Further the solar panel is divided into solar section and solar cells. The calculations are made in a hierarchical order. (Zeigler, 2008). Each class carries its functionality in design and verification of the power sub system.

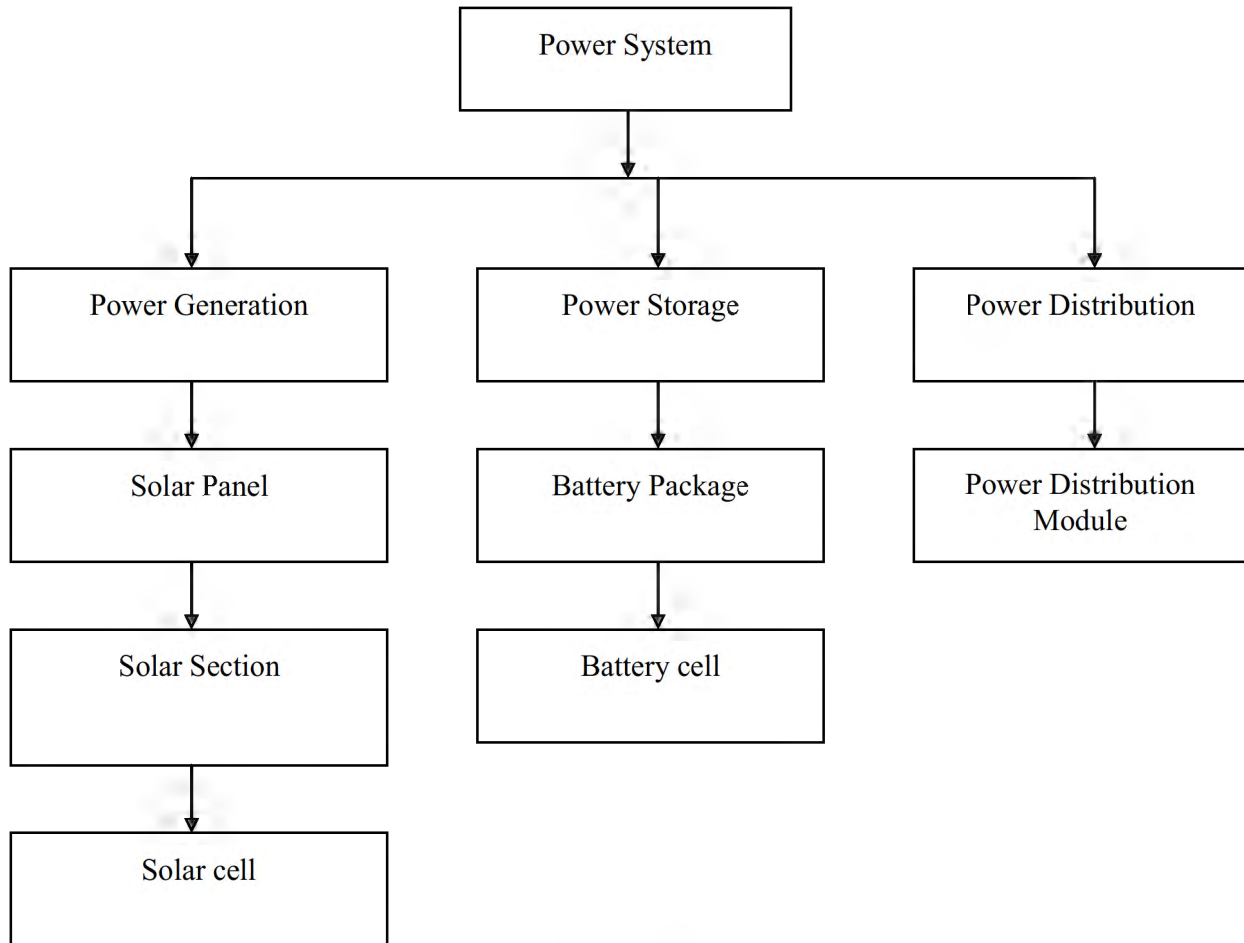


Fig. 2-5 Power System generic layout

2.3.4 Communication System

As a part of this thesis, a generic model for the communication dimensioning and functional architecture was built. The communication system dimensioning consists of uplink and downlink dimensioning. Based on the above values the functional architecture for the communication system is selected from the datasheet. The theory of the communication system modelling is explained in the following section.

Theory

In this section calculations regarding the satellite's link budget are made. These calculations are essential in order to guarantee reliable communication between the satellite and the ground station. Given the limitations of the CubeSat and a ground station, it is important to calculate if a reliable communication between the satellite and the Earth is possible in all circumstances. Before actual link budget calculations are made, the background about these

calculations will be introduced briefly in order to provide the formula necessary for derivation of the link budget.

The link budget is the foundation for designing any radio link, regardless if it is terrestrial or in space. It is possible to express to relationship between the transmitting power P_t and the power at the output terminal of the receiving antenna, based on the formula for the collecting power $P_r = S * A_r$, Transmitter gain G_t and receiver gain G_r .

$$P_r = \left(\frac{c}{4\pi df}\right)^2 * P_t * G_t * G_r \quad (2-10)$$

c is the velocity of light, f is the frequency of transmission and d is the maximum distance between ground station and the satellite. The quantity $\left(\frac{c}{4\pi df}\right)^2$ is also denoted L_p and is called the Path Attenuation or Path Loss, a dimensionless quantity. Using this Path Loss, the relationship between the received and transmitted power may now be simply expressed with:

$$\frac{P_r}{P_t} = \frac{G_t G_r}{L_p} \quad (2-11)$$

It is more convenient to express the link budget in decibel (dB), mathematical calculations become more easily (a multiplication transforms into an addition and a division into a subtraction).

$$P_r = P_t + G_t + G_r - L_p \quad (2-12)$$

It is possible to transform the path attenuation, into the dB-domain as well, resulting in a formula depending on the distance and frequency:

$$L_p = 32.34 + 20 \log \frac{d}{1 \text{ km}} + 20 \log \frac{f}{1 \text{ MHz}} \quad (2-13)$$

32.34dBW contains the constant $\frac{4\pi}{c}$ as well as the power of 10 due to the more convenient usage of kilometres instead of meters for the distance and Megahertz instead of Hertz for the carrier frequency. The final formula for the link budget cannot be given as

$$P_r = P_t + G_t + G_r - 32.34 - 20 \log \frac{d}{1 \text{ km}} + 20 \log \frac{f}{1 \text{ MHz}} \quad (2-14)$$

The Signal-to-Noise Ratio (SNR) is the ratio between the power of the information carrying signal and the power of the unwanted noise (in the same bandwidth).

$$\frac{E_b}{N_0} = P_r - 10 * \log \frac{B}{1 \text{ Hz}} - 10 * \log \frac{k}{1 \text{ J/K}} - 10 * \log \frac{T}{1 \text{ K}} \quad (2-15)$$

Absolute Temperature T , k is the Boltzmann constant and B is the Bandwidth.

$$\frac{E_b}{N_0} = EIRP + \frac{G}{T} + 196.15 - 20 * \log \frac{d}{1 \text{ km}} - 20 * \log \frac{f}{1 \text{ MHz}} - 10 * \log \frac{B}{1 \text{ Hz}} [\text{dB}] \quad (2-16)$$

$EIRP = P_t + G_t$ is called the “Equivalent Isotropic Radiated Power” or the power required by the transmitter output stage if the antenna radiated equally in all directions (isotropic). The advantage of using EIRP is that it is possible to trade antenna gain for transmitter output power for a given EIRP requirement.

$$\frac{G}{T} = G_r - 10 * \log \frac{T}{1K} [dB/K] \quad (2-17)$$

G/T allows the link designer to trade receiving antenna gain for system noise temperature with a given G/T requirement. Again there is an inconsistency of nomenclature: G/T is a symbol, not a calculation.

Other way to represent E_b/N_o is,

$$\frac{E_b}{N_o} = \ln \left(\frac{1}{2BER} \right) \quad (2-18)$$

Where, BER is the Bit error rate, which is assumed to be 10^{-5} or less in most of the space missions. Deciding the required E_b/N_o (energy bit to Noise spectrum) is based on a graph,

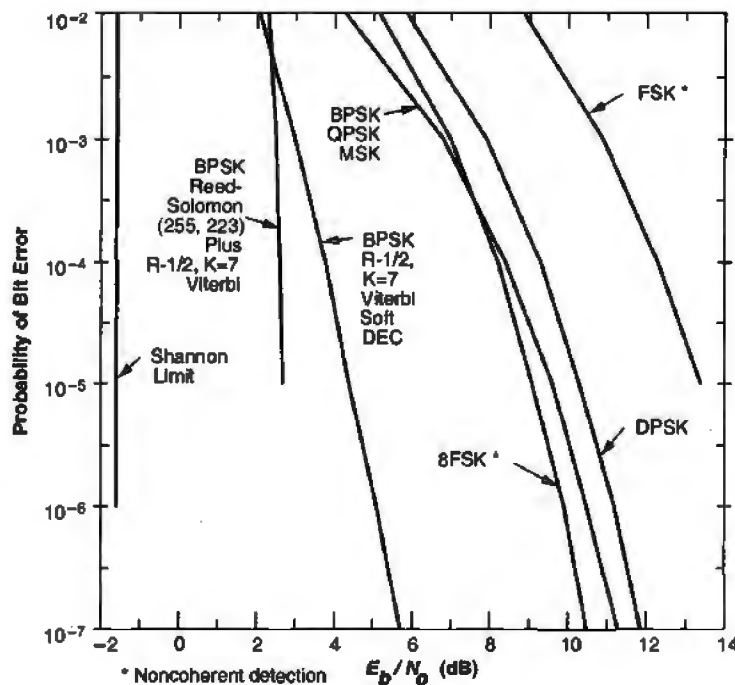


Fig. 2-6 BER vs E_b/N_o

Hence the required Carrier to noise ratio can be given as,

$$\left(\frac{C}{N} \right)_{req} = \frac{P}{B} \ln \left(\frac{1}{2BER} \right) \quad (2-19)$$

The dimensioning model in (v)-Sys consists of Uplink Dimensioning and Downlink Dimensioning. The link budget is the foundation for designing any radio link, regardless if it is terrestrial or in space. The inputs to the communication dimensioning model are the required data rate, type of modulation, antenna specification and the physical constants used for the link dimensioning.

The outcome of the dimension model are the specified minimum ratio of Energy per bit to noise ratio (E_b/N_o), Bandwidth B, Transmit data rate, Specified EIRP, Specified Minimum Transmitter power and Required Carrier to noise ratio (C/N).

The figure below gives a generic layout of the communication system dimensioning and its functional architecture. A systems engineering viewpoint has to be applied on the above theory to accommodate the same in (v) - Sys.

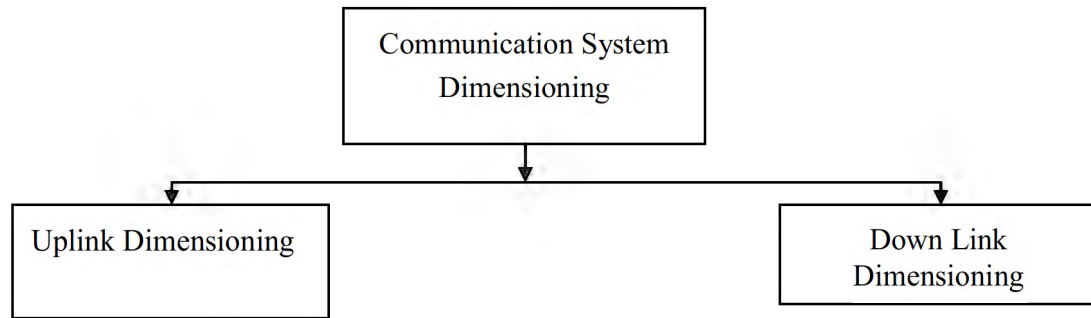


Fig. 2-7 Communication Dimensioning

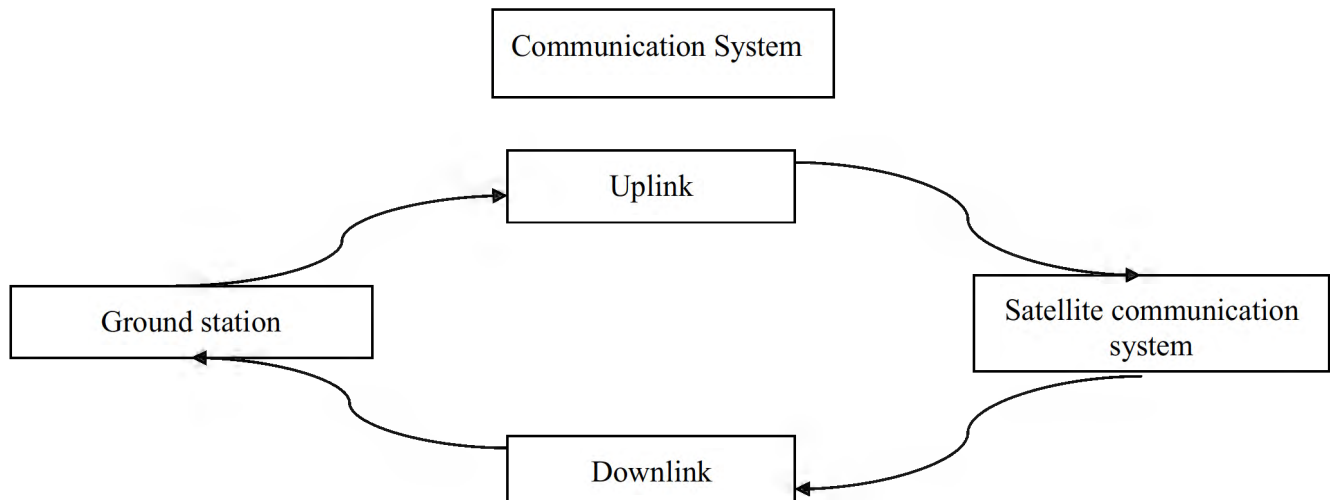


Fig. 2-8 Generic Communication Model for Satellite



3 Introduction to MOVE Mission

3.1 Mission MOVE

The reliability of innovative Space technologies cannot be realized without an on-orbit demonstration. The Institute of Astronautics has been developing a satellite platform for verification of micro-scale components using MOVE “Munich Orbital Verification Experiment” in cooperation with industrial partners. The MOVE verification platform has the CubeSat standards because of high degree of consistency and numerous commercial off the shelf components. The MOVE platform has the capability to provide demands for multiple payload and sufficient interfaces. The first version of the satellite platform has an on board camera for photographing the Earth. For the development of MOVE, LRT’s experience in system engineering has been utilized through in house development tools, such as (v)-Sys. As an educational project, the participating student’s learning has been maximized by system level design together with hardware integration under the supervision of the LRT employee.

The motivation for the overall MOVE CubeSat project development is primarily educational: educate students in space technologies and space system engineering. This motivation has several impacts:

- *The project shall involve undergraduate and postgraduate students and young engineers through its whole life cycle;*
- *The project cost shall be relatively small, related to a university type development;*
- *Compared to an industry type space project, decisions were taken to simplify the design or design for low-cost and thus might not comply with the usual standards.*
- *Keeping these aspects in mind, the mission and science objectives for the project are summarized in the following requirements.*

The project shall design, build, and test a satellite.

The success criterion is: deliver a fully tested satellite to the launch site. This objective assumes the development of both a ground and space system.

The project shall launch the satellite and communicate with it using the ground and space systems.

The success criterion is: establish a radio connection with the developed ground system and download telemetry.

3.2 MOVE Satellite

The MOVE’s minimal mission duration is at least four months after its launch. This period is divided into one month for commissioning and three months for Payload experiment. The satellite platform consists of an electrical power system, on board data handling, Communication system and provision for payload interface. The distinction between MOVE and other CubeSat has been the deployable solar panels. The satellite has a passive magnetic control. The MOVE satellite works on three different modes Initialization mode, safe and nominal mode, further explanation will be given in the coming chapters.



Fig. 3-1 MOVE promotional background and official Logo

3.3 MOVE Orbit

For the MOVE mission different orbits were chosen. The possibilities were sun synchronous Earth orbit where the former CubeSat missions were launched with an altitude of 600 to 850 km and a launch from the ISS. For viewing all parts of the Earth's Surface at close quarters it is necessary to adopt a low altitude polar orbit. (Peter Fortescue, 2002). The sun synchronization occurs when the orbit plane rotates in space at the same rate as the earth moves around the sun- at one revolution per year or roughly one degree per day.

Sun Synchronous orbit has an advantage for some Earth viewing missions, in that the Earth is always viewed at one of two times of day. An inclination in excess of 90 is needed to achieve synchronization through regression of the line of nodes without the use of fuel. Since for the Earth observation satellites in Low earth orbit the altitude are 550- 950Km then orbital time of 95-100 min (Wertz J. , 1999). The spacecraft at low earth orbit will experience eclipse, the frequency and the duration of which is determined by orbital inclination- for example, in low altitude equatorial orbit the satellite is eclipsed for about 40% of every orbit. Such concerns identify a suitable launch season for sunlit operation as outlined below. In Sun-synchronous orbits (SSO), the nodal regression caused by the J₂ (non-spherical Earth) is matched with the angular rotation of the Earth around the Sun. Thus the plane of a Sun synchronous orbit keeps a constant angle alpha with the Earth-Sun vector. As show in the figure these orbits are almost polar and therefore cover all latitudes.

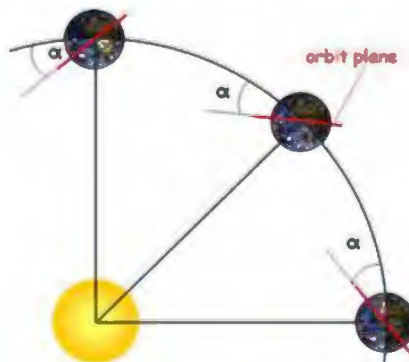


Fig. 3-2 Angle α

SSO orbits are ideal for Earth observation missions since the satellite crosses the Equator always at the same local time. Further they simplify the satellite design since eclipse durations are almost constant.

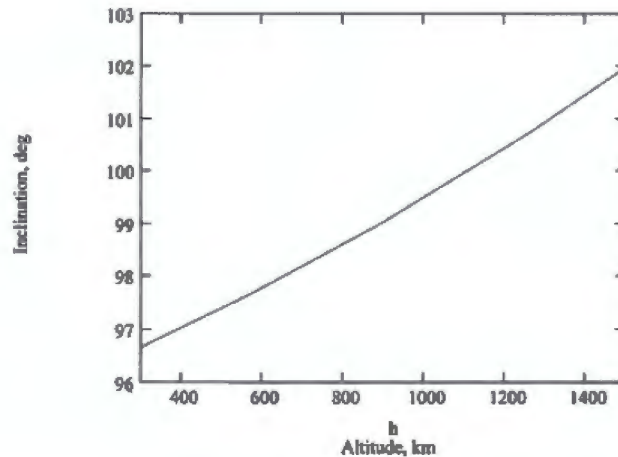


Fig. 3-3 Altitude vs inclination

The eclipse duration is an important parameter for the design of the space system. An analysis was performed that calculates the minimum, mean and maximum eclipse duration as a function of the Sun /orbit plane angle α and the altitude (using STK). Shows that:

(Assmann, 2008)

- For $\alpha=0$, the orbit plane coincides with the terminator, the satellite is in constant light, and there are no eclipses.
- While eclipse duration varies between 19 min and 35 min for orbits with $20 \text{ deg} < \alpha < 80 \text{ deg}$ at 600-650 km altitudes.

After the initial analysis the chosen orbit was the SSO with respect to the overall performance and former experience with CubeSat. The sun synchronous orbit maintains the orbital plane nearly a fixed angle relative to the sun throughout the earth orbit. The orbital inclination being used for modelling was 97.871° . The orbital analysis was made using the STK software as a part of a different thesis work. On comparison based on the mission lifetime, cost of launch, ground link timing, quality of communication and experience of previous mission shows the SSO 625 km 18:00 o'clock LT orbit has a better recital with respect to the mission statement. The thermal analysis shows that the 18:00 o'clock orbit is the easiest to control with passive methods of thermal control (Assmann, 2008).

The solar cycle is expected to peak in 2011. Thus in 2008, the solar environment can be expected to be relatively high. As the solar activity influences the Earth's atmospheric density profile at high altitude significantly, disturbance forces due to atmospheric drag will cause the satellite to lose altitude.

3.4 Mission Phases

The following paragraph summarizes the various phases after final satellite acceptance until the disposal of the satellite.



3.4.1 Launch and early operation Phase (LEOP)

In the pre-launch phase the final launch site tests are performed and the satellite is prepared for launch. Activities include the charging of batteries and the check-out of the satellites subsystems. This action will be performed while the satellite is already integrated into its launch pod. LEOP phase will start after satellite separation from its launch container. It will include the following:

- 15 minutes transmission and antenna deployment dead time.
- Switch-on and antenna deployment.
- Initial satellite acquisition (RF Beacon).
- Validation of the correct operation environment on-board the satellite.
- Validation of the space-ground data link (RF Transceiver).

This phase will end once the validation steps above have been performed. This phase will be terminated within four days after ejection from the launch container.

3.4.2 Commissioning Phase

During satellite commissioning the on-board systems of the MOVE satellite will be tested and their operational performances confirmed. The results will be used to correct and calibrate the on ground satellite models. Commissioning will end after the validation of all systems and will be terminated within 14-28 days after ejection from the launch container.

During nominal period MOVE will be fully operational and shall perform the distinct payload experiment. This phase will be over 3 months after the end of the commissioning phase.

3.4.3 Experiment Phase

In this phase MOVE's primary payload experiment is being conducted. It is important to define this separate stage for the payload operation, since it is very power consuming and only active for a short period of time. Every two orbits one test cycle is carried out for the batteries. Should the remaining electrical power be lower, the payload operation is postponed until sufficient energy is available. (Römhild, 2007).

Initialization routine for PL:

- OBC receives internal or external (COM) command
- Remaining battery power is checked positive
- OBC initiates payload experiment

Shutdown routine for PL:

- Experiment returns finished signal and shuts down independently
- OBC forces payload shutdown after timeout
- Forced shutdown if system health levels are below given limits

3.4.4 Disposal Phase

If the satellite is still operational after this phase, the mission will be extended. No active disposal is foreseen. Science operations will be performed as long as possible and subsystem degradation will be monitored up to a mission critical failure, or up to 1 year, whichever comes first.

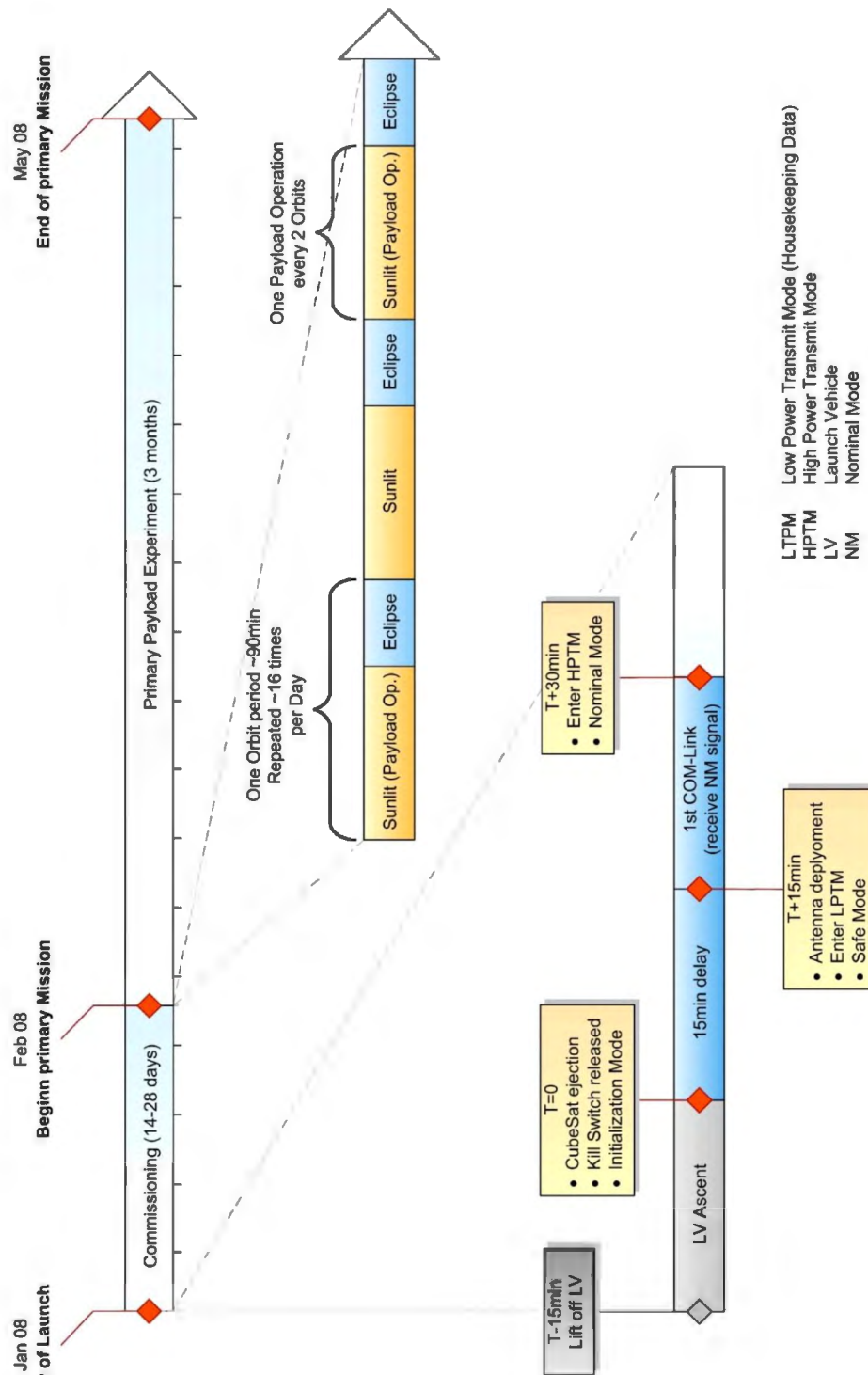


Fig. 3-4 Initial Scenario design for MOVE Mission ((Römhild, 2007)

3.5 Satellite Modes

The following selection is the modes of operation used for the MOVE mission. The major modes that have been identified are:



3.5.1 Initialization Mode

During the initialization mode the important tasks like antenna deployment is activated. The subsystems that will be in the power up mode are the Electrical power system and the onboard data handling system. The Electrical power system is activated via a deployment switch or when returning from deep discharge protection or during a hard reboot.

During this mode, the watch dog system is being activated periodically by the on board data handling. When the boot sequence succeeds the OBC initially performs a memory check which will determine the memory hardware or the software stored in them is corrupted or not. Once the burn wire used for the antenna deployment burns, immediately the state of the system changes to safe mode. If not, the OBC waits for a certain period of time and the reinitiates the burn operation (Römhild, 2007). A minimum of 10 retries are made before the system goes into safe mode.

Once on checking the burn operation as positive the OBC switches to safe mode.

3.5.2 Safe Mode

The safe mode is intended to ensure the survival of the CubeSat at all time and the state can be changed when needed (nominal mode). During the safe mode all the non essential activity are shut down and check or monitors the health condition of the satellite. The subsystems in operation during safe mode are the EPS, OBC and the communication module (COM). The temperatures of the system are monitored periodically and the batteries are put to charging and conditioning (Römhild, 2007). The COM starts transmitting beacon or (Low power Transmission) of the housekeeping data periodically.

On need, the OBC can perform software updates which are transmitted from the ground station. The MOVE satellite stays in safe mode until a confirmation command is send to change the state to nominal mode. On the other hand, when the battery charge is high enough and the temperature are within limits, and then the mode can very well change to nominal mode without any initiation from the ground.

If, the confirmation is not given then the system state shifts to initialization mode. The operation shall prevent the mission failure in case the antenna is not deployed.

3.5.3 Nominal Mode

The main operational mode MOVE is expected to spend most of its time in. In this state the satellite will perform the major activities like primary payload experiment, downlink communication with the ground station as well as take and transmit pictures. The subsystem in operation will be EPS, OBC, COM, and Payload. (Römhild, 2007). In this mode the CubeSats activities can be classified into different phases (The “Standby Phase” and the “Experiment Phase” are described in the following two chapters). It is vital to define several subsystem activation limits within the “Nominal Mode”, since the survival of MOVE has to be ensured at all time. Therefore from the remaining power, the amount for one eclipse phase battery heating period has to be reserved.

When a picture has to be taken (provided that MOVE is not in “Standby Phase”) the camera is activated by the OBC for the duration of the process. OBC switches to “Safe Mode” when

temperature levels are critical or the battery power left is just enough to heat the satellite during one eclipse phase.

Tab. 3-1 Modes and Subsystem activity

Modes	EPS	COM TX	COMM RX	BEACON	OBDH	PL
Initialization Mode	Active	Inactive	Inactive	Inactive	Active	Inactive
Safe Mode	Active	Inactive	Active	Active	Active	Inactive
Sleep Mode	Active	Inactive	Inactive	Active	Active	Inactive
Nominal Mode	Active	Active	Active	Active	Active	Active

Active	
Inactive	

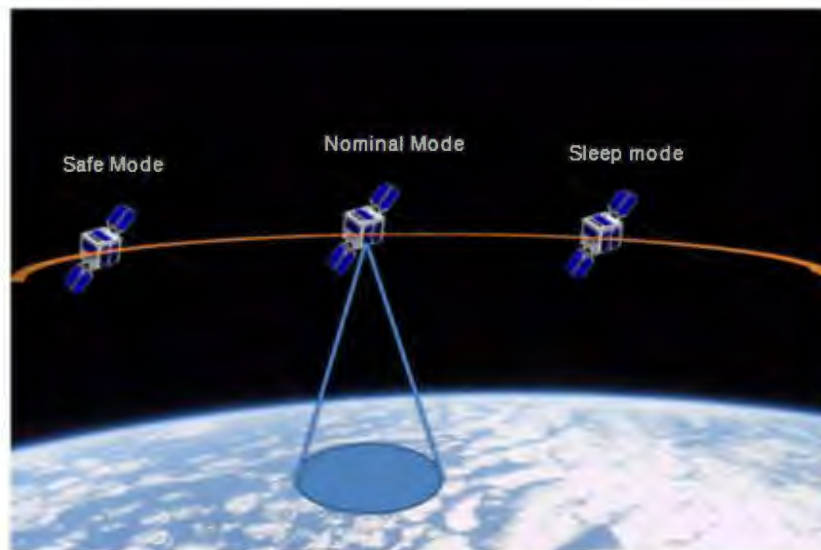


Fig. 3-5 Satellite Model illustration

3.6 Ground station

The Ground station being used for the MOVE satellite operation had two choices, the ground station from Berlin and Ground station from Garching Munich. As the results (Assmann, 2008) from STK (Satellite Tool Kit) software shows that on contrast based on different parameters, the outputs are almost similar to the requirement with both the ground stations. On comparison with cost and future developments the LRT Garching ground station was chosen. As LRT have plans to upgrade the Ground station to support the MOVE mission and future missions like Bayern SAT. The latitude and the longitude were referred from the Google map as 49.0108N and 11.081E respectively



Fig. 3-6 LRT Ground station, Muenchen

3.7 Launcher

The MOVE satellite has been planned to be launched using an Indian Launcher PSLV (Polar Satellite launch vehicle). On the cost and the reliability analysis PSLV was chosen. The polar launch vehicle is a versatile heavy launcher. The launcher can reach an orbital altitude of 850Km. The MOVE satellite will be flying at the altitude of 625 Km and inclination of 97.871° .



Fig. 3-7 Indian Launcher PSLV (BharatRakshak ISRO)

Mechanical interface between the satellite and the launcher is achieved through the P-POD or TPOD launcher.



Fig. 3-8 PPODs



4 Environmental model for MOVE

4.1 Atmospheric model

The atmospheric model calculates the mean atmospheric density at the given altitude. This model is based upon the data of the MSIS atmosphere, altitudes between the given values are linear interpolated, and therefore they contain a certain error.

$$f(x) = f_o + \frac{f_1 + f_0}{x_1 + x_0} (x - x_o) \quad (4-1)$$

The above equation is a linear interpolation equation used to find the average atmospheric density for the give altitude. Since (v)-Sys is used for static calculations the dynamic variation in atmospheric density is not considered.

The values of the atmospheric model are referred from (Wertz J. , 1999), the values are restricted to low earth orbits, particularly for the MOVE mission. The orbital altitude restricted between 100 to 1500 Km from the earth surface and the earth's radius is presumed to be 6378 km.

For the MOVE orbit, the orbital radius is calculated to be 7003 km. For an altitude of 625 km, the atmospheric density is calculated to be $1.0957 \cdot 10^{-14} \text{ kg/m}^3$.

4.2 Magnetic Field Model

Electric components of a satellite, like computation units or more important payloads e.g. scientific instruments induce magnetic fields. These interact with the magnetic field of the Earth and create disturbing torques.

The Earth's Magnetic field is roughly a dipolar; the equation for the local magnetic field intensity is given by, ((Wertz J. , 1999)

$$B(R, \lambda) = (1 + \sin^2 \lambda)^{1/2} B_o / R^3 \quad (4-2)$$

Where B is the local Magnetic field intensity, λ is the magnetic latitude, R is the radial distance measured in Earth radii and B_o is the magnetic field at the equator at the Earth's surface which is a constant $7.96 \cdot 10^{-5} \text{ tesla m}^3$.

The magnetic field of the Earth is an important shield to fend off these particles. Without this shield, it would make life on Earth impossible. The dynamics of the magnetic field of the Earth was not taken into account within this work.

To calculate the strength of the Earth's magnetic field in a certain position in a satellite orbit one can use this simplified equation. Since (v)-Sys is for static calculations the, field variations are also assumed to be static.



$$B_{min} = \sqrt{((1 + 3.\sin^2(i)).\frac{B_0}{R_{apo}^3}} \quad (4-3)$$

Within this calculation it is assumed, that the inclination of the orbit plane is equal to the magnetic latitude. The below equation calculates the minimum magnetic field strength in a satellite orbit.

The B_{max} is calculated using the below formulae, with the same assumption.

$$B_{max} = \sqrt{((1 + 3.\sin^2(i)).\frac{B_0}{R_{per}^3}} \quad (4-4)$$

For the MOVE satellite at an altitude of 625kms, the maximum earth magnetic field strength is calculated to be $4.59 \cdot 10^{-5}$ Tesla.

4.3 Physical Constants

The below table is the most predominantly used constants for the satellite design and verification calculations.

The values are taken from various standard references. In (v)-Sys, a physical constant object is created in the environmental model. These values can be referenced or used from any of the dimensional models or the functional architecture mathematical calculations. Commonly used constants are used stored under the physical constants object; this can also be updated when required. For the MOVE modelling these are the frequently used constants.

Tab. 4-1 Physical Constants

Parameter		
Property	Value	Dimension
Mean Solar Intensity at Earth Distance	1,3670E+03	[W/m ²]
Boltzmann Constant	1,3807E-23	[J/K]
Vacuum Speed of Light	2,9979E+08	[m/s]
Equatorial Radius of the Earth	6,3781E+06	[m]
Obliquity of the ecliptic	23,4390	[°]
Gravity constant of the Earth	3,9860E+14	[m ³ /s ²]
Standard Acceleration of Free Fall on Earth	9,8067E+00	[m / s ²]
Stefan Boltzmann Constant	5,6704E-08	[W/m ² K ⁴]

5 Model MOVE in (v)-Sys

The first step is to learn the systems engineering software (v)-Sys using the available resources and study the background of the MOVE Mission, and its hardware as well as literature concerning the project work. The next step is to understand the structure and the functioning of (v)-Sys, as well as MS Excel. It is important to know, which calculations and mathematical operations are possible within MS Excel and also the required functionalities to be updated before starting the work. A basic task is to identify the product architecture and its requirements for the MOVE mission and its functionalities. The relevant formulas and parameters concerning the link budgeting and the communication systems are to be worked out. MS Excel Sheets are generated, containing the formulas used to size the necessary components.

Finally the different configurations are integrated into the structure of (v)-Sys and the calculation sheets are implemented. The hardware components datasheets are compared and merged into databases. Common parameters are implemented into (v)-Sys and the databases are integrated into the (v)-Sys structure. The mass and its inertial components are to be computed using the data sheets and CATIA Model to verify its physical dimensions. The referencing between the subsystems is to be made. Based on the outcomes of the dimensional model and the product architecture the satellite configurations are to be verified and analyzed.

5.1 MOVE dimensioning model

The MOVE dimensioning model consists of the link dimensioning, electrical power system dimensioning, data storage dimensioning and attitude control dimensioning. Based on the given requirements the subsystems are decided. A generic CubeSat model was referred with the help of the background literature and the MOVE design documents to figure out the critical subsystems to be dimensioned. Even though the above discussed subsystems are to be dimensioned, the support systems like the launchers, ground station etc, are to be modelled under the Mission *object*. The only non realizable object in the MOVE modelling is the orbit calculation. Most of subsystem and system dimensioning requires the parameter calculated from the orbit definition. The following chapter will discuss about the orbital calculations in detail.

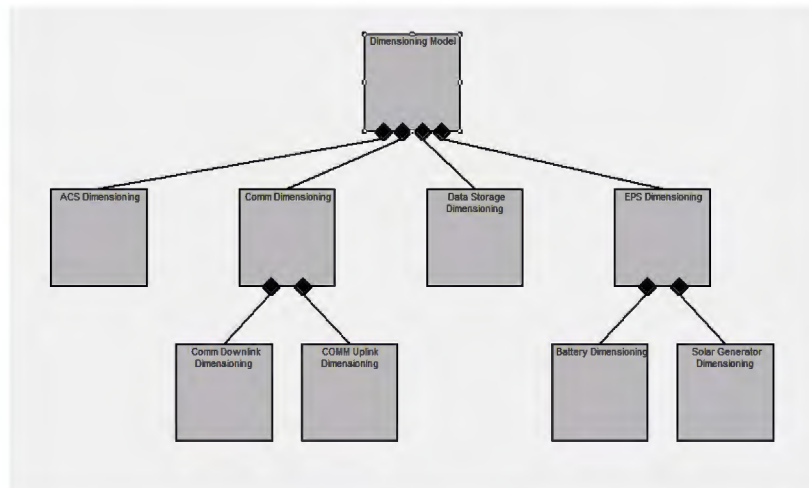


Fig. 5-1 MOVE Dimensional Model

5.2 Orbit Calculation

In (v) - Sys simulation, the orbit conditions are defined using an excel sheet. The inputs for the orbital definition are latitude and longitude of the ground station, equatorial radius of the earth, minimum elevation angle, mission duration and other physical constants required for the calculation of the orbital parameters. The parameter section has the facility to choose the orbit with different inclination and altitude. If the essential parameter doesn't match with the requirements, then it can be manually defined. Since we use a circular orbit the parameters required for calculation are the altitude and inclination.

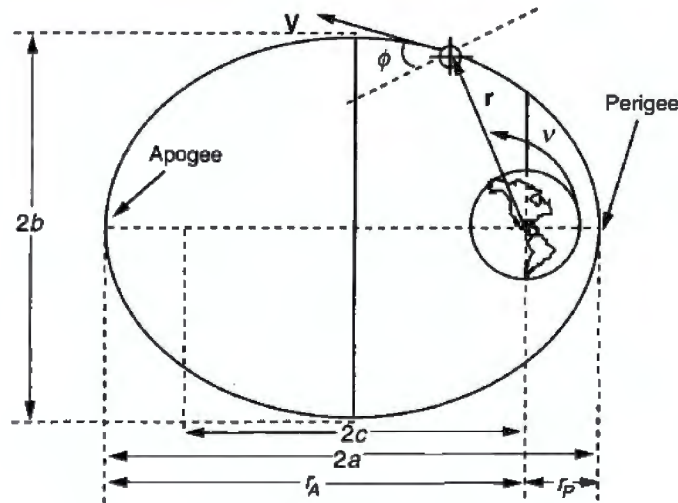


Fig. 5-2 Orbital Elements (Wertz J. , 1999)

The (v) - Sys orbital analysis gives the following results, as defined in the previous sub section. The orbital height was chosen to be 625km with 97.871° inclination. Since it is a Circular orbit, the eccentricity remains 0. The total mission time was required to be four months with one month for commissioning and three months for the payload experimentation. Based on the above inputs and parameter, the orbital calculations are performed in the excel sheets in (v)-Sys. The table below shows the results and the achieved values.

Tab. 5-1 Orbital calculation

Calculation of	Results
Angular radius of the spherical Earth	65.618°
Orbit period P	97.204 min
Eclipse duration T_E	35.431 min
Daylight Duration T_D	61.776 min
radius of perigee r_P	7003.14 km
Max Nadir Angle $\eta_{\max}$	63.75°
Max Earth Central Angle $\lambda_{\max}$	16.24°
Max Distance $D_{\max}$	2388.65 km
Min Earth Central Angle $\lambda_{\min}$	-5.86°
Min Nadir Angle $\eta_{\min}$	-44.71°
Min Distance $D_{\min}$	625 km
Time in View	616.5225 s

The maximum time in view, when crossing over the ground station with the antenna elevation angle of 5° gives approximately 10 minutes of access. But on increasing the elevation angle to 10° decreases the access time to 8 minutes approximately. The elevation angle varies dynamically with each orbit; hence an average of 7.5 min can be expected. The real time elevation value may vary between 5° - 40° . Since the mission uses the sun synchronous orbit, the maximum eclipse time remains to be 35.4 min. This will be the worst case possible. The eclipse time and the daylight duration will induce the design parameters of the power system and communication system. The maximum distance $D_{\max}$ decides the horizon for communication. Generally the communication between the ground station and the satellite are to be neglected while passing at this range. The results of the orbital calculation determine most of the subsystem Design and verification parameters.

5.3 Electrical power system dimensioning

The electrical power system consists of three subsystems, power generator, power storage and power distribution unit (Wertz J. , 1999). The power distribution is not of great importance to modelling, since the distribution unit will be chosen based on the dimensional outcome of the other two subsystems. Moreover the teams design plan is to have a reliable COTS subsystem block for this purpose. Hence the outcome of this EPS dimensioning model will be the storage and generator dimensioning.

5.3.1 Battery dimensioning

The battery dimensioning model requires the user to have vital information about the commercially available units to initiate the dimensioning process. Else the specification data sets can be referred from the product architecture which is not introduced in this report till now.

As per the initial design assumptions and information the bus voltage for the whole system is assumed to be 7V, hence the MOVE satellite dimensioning is expected to have a 7V DC power supply. The total electric capacity EOL cannot be assumed or predicted, so the value can be used from the data sheets of the battery to validate and begin the dimensioning process. The table below shows the inputs and the parameters used for dimensioning the battery package.

Tab. 5-2 Input parameters for dimensioning of battery

Input	Values
Depth of discharge allowed	30%
Main Bus Voltage	7 V
Mean Power need Eclipse	1.28 W
Mission Duration	4 Months
Nominal Voltage	3.3 V
Total Electric Capacity EOL	5.77 Ah
Parameter	
Efficiency Electrical Path Battery 2 Loads	90%

The chiefly used battery package for CubeSat mission are Lithium ion batteries, hence the value of total electric capacity at End of life and nominal cell voltage are taken from the data sheets of the respective battery type. The mean power need during the eclipse is estimated roughly for a Pico satellite. These values are not accurate, while these values are used to trigger the iterative dimensioning process.

Tab. 5-3 Battery dimensioning calculation results

Calculation of	Results
Specified Minimum Usable Battery Energy	6683.88 J
Specified usable Cell Energy	20564 J
Number of Battery cells in Series	2.1
Specified Minimum Usable Capacity	0.12 Ah
Specified Minimum Total Capacity	0.4 Ah
Number of Battery cells parallel	0.06
Minimum Number of Battery cells	0.2

The calculated values using the parameters shows that the number of cells connected in series should be of 2 and the number of battery cells connected in parallel shows to be a indefinite value in terms of dimensioning, thus it is rounded to be at least 1 and the minimum number of battery cells should be 0.3 rounded to be 1. The specified minimum usable battery energy is 6684 J.

5.3.2 Solar generator dimensioning

The Solar generator dimensioning model, to initiate a dimensioning model the power consumption during the Eclipse period and the Sunlight period are required, the values are referred to the orbital dimensioning object. Each electronic components used in the satellite consumes power. These values are taken from the Object MOVE Satellite, where the power consumption of the MOVE satellite subsystems is summed up. The values are to be assumed, while the initial values are not known. The Degradation and Temperature corrected Power at MPP EOL is referred from the solar cell datasheets.

Tab. 5-4: Solar cell Input parameter

Input	Values
Mean Power Need Eclipse	1.28 W
Mean Power Need Sunlight	2.41 W
Minimum Sunlight Period	3724,42 s
Maximum Eclipse Period	2124,27 s
Power at MPP EOL_T_corrected	1.0887 W
Main Bus Voltage	7 V
Voltage at MPP EOL_T_corrected	2.2667 V
Maximum Current_Solar Array Section	0.4523 A
Power at MPP_Solar Array Section	3.8385 W
Parameter	
Efficiency Electrical Path SA 2 Battery 2 Loads	60%
Efficiency Electrical Path SA 2 Loads	80%

The other parameters shown in the above table are from the requirement definition and general assumption done during solar panel design.

Tab. 5-5: Solar cell dimensioning results

Calculation of	Results
Specified min. solar generator power supply P_{SA}	4.23 W
Specified min. number of solar cells	3.88
Minimum number of solar cells in series in a solar array section	3.23
Maximum number of parallel solar cells in a solar array section	1.007
Number of solar array sections	1.10

The specified minimum solar generator power supply should be 4.23 W and the minimum number of solar cells used should be 4 cells with 3-4 cells connected in series. The number of solar array section required is 1 approximately.

5.4 Data Storage Dimensioning

The data storage dimensioning requires the different data rates from different subsystems. A detailed calculation and analysis are made in the chapter 9. The data rate is calculated to be 1134bps and the required minimum volatile mass memory will be 5MB.

5.5 Communication dimensioning

The communication requirements were driven in demands set by the mission requirement. With the payload the satellite has to collect and transfer user data as well as to transmit housekeeping telemetry data. The link shall comply with the propagation conditions of a sun synchronous low earth orbit. The satellite has to store the data when the satellite is not visible to the ground station and transmit when the satellite is in the reception area of the ground station. The payload data transmission from the satellite to the ground station is required once per day. The satellite will use a frequency between 435 MHz to 438MHz for the uplink and a frequency between 130MHz to 169 MHz for the downlink. As the satellite has restricted size and weight requirement, a sophisticated modulation are used. The modulation scheme shall be *Frequency shift Keying* (FSK) and *Binary Phase Shift Keying* (BPSK) respectively. Given the tiny size and diminutive weight of the satellite a simple design of the antenna is chosen. A dipole (Half wave length antenna) antenna is appropriate for the mission uplink and a monopole antenna for the mission downlink data transfers. The length has to be sized in order to match the given frequency, which will be used. As the satellite will spin around the X-axis, the antenna will be in inadequate position for transmission every now and then. To consider the loss of connection, the link shall be designed to provide a 3dB margin. The COM Module has to survive harsh environment like Vacuum, radiation, temperature between 5° to ± 15 °C.

The communication budgeting is based on the dimensional model and a product verification model. The dimensioning model is of 2 sections, an uplink dimensioning and a downlink dimensioning.

5.5.1 Uplink dimensioning

The Uplink dimensioning model calculates the required Bandwidth; transmit data rate, Specified EIRP and the carrier to Noise ratio. The bandwidth is calculated based on the modulation technique used. The energy per bit to noise density was interpolated based on the graph discussed under the Link budgeting.

Tab. 5-6: Uplink Dimensioning

S.No	Calculation of	Results
1	Specified minimum ratio of energy per bit to noise density	8.5171
2	Specified Data rate to transmit	0.3 Kbps
3	Bandwidth	0.6 Kbps
4	Specified Effective Isotropic Radiated Power (EIRP) = $P_{EIR,spec}$	0.8 W
5	Specified min. transmitter power $P_{Tx,spec}$	0.67 W
6	Specified Carrier to Noise Ratio(C/N)	8.51

Based on the possible modulation techniques, FSK modulation was chosen for Uplink dimensioning. The ground station antenna parameters were fixed because the mission will be using the Garching LRT Ground Station. The nominal data-rates were assumed based on the rough estimates and the requirements made on the tele-command data and payload data rates.

The calculation results show that the specified effective Isotropic Radiation is 0.8W, while the specified minimum transmitter power is 7.33W. The specified carrier to Noise ratio is calculated to be 7.33dB.

5.5.2 Downlink Dimensioning

Tab. 5-7: Downlink dimensioning

S.No	Calculation of	Results
1	Specified minimum ratio of energy per bit to noise density	10.8197783
2	Specified Data rate to transmit	1.2Kbps
3	Bandwidth	1.44 Kbps
4	Specified Effective Isotropic Radiated Power (EIRP) = $P_{EIR,spec}$	0.017 W
5	Specified min. transmitter power $P_{Tx,spec}$	0.17 W
6	Specified Carrier to Noise Ratio(C/N)	6.29

The downlink dimensioning is also done on the same method with different modulation technique and different telemetry data rates. The modulation technique was chosen in the same way as the uplink. BPSK modulation was decided to be used. The calculation results are as shown above. The required $P_{EIR,spec}$ is calculated to be 0.017W and the specified minimum transmitter power to be 0.17W.

5.6 Attitude control system dimensioning

The implementation of the magnetic control mechanism is similar to that of the external torque created through Earth's magnetic field. There is only a difference concerning which value is known, and which is being calculated.

The orbit radius R_{apo} [km], taken from the class "Orbit, the "Magnetic Field Model" calculates the minimum Earth's magnetic field strength B_{min} in tesla. Therefore the class "Passive Magnetic Control Dimensioning" needs to be connected with B_{min} of class "Magnetic Field Model". The external torques from the object "MOVE Satellite" is compared and the largest value is used within the calculation as T_{max} in order to calculate the necessary dipole moment D_{mag} . In order to estimate the impact on satellite attitude the torque created by interaction of the dipole moment of the satellite D_{mag} with the Earth's magnetic field can be calculated with

$$T_{mag,max} = D_{mag} \cdot B_{max} \quad (5-1)$$

Because it is always the aim in this work to get the biggest impact of a certain effect, the maximum torque $T_{mag,max}$ is calculated. This consideration requires the maximum Earth magnetic field strength B_{max} in the satellite orbit.

The initial values to dimension the Magnets were, the force due to air drag is assumed to be 10^{-9} , the solar pressure is assumed to be 10^{-10} and the gravity gradient was assumed to be 10^{-9} . Based on the above values the dipole moment required by the permanent magnet is calculated to be 0.023 Am^2 .

6 MOVE product architecture

The product architecture is a part of the (v)-Sys modelling, where the product specification are matched and modified according to the dimensional requirement. The product architecture carries the different systems and subsystems, which are modelled based on the outputs from the dimensional Models. The following gives the detailed explanation about the Subsystems of MOVE satellite.

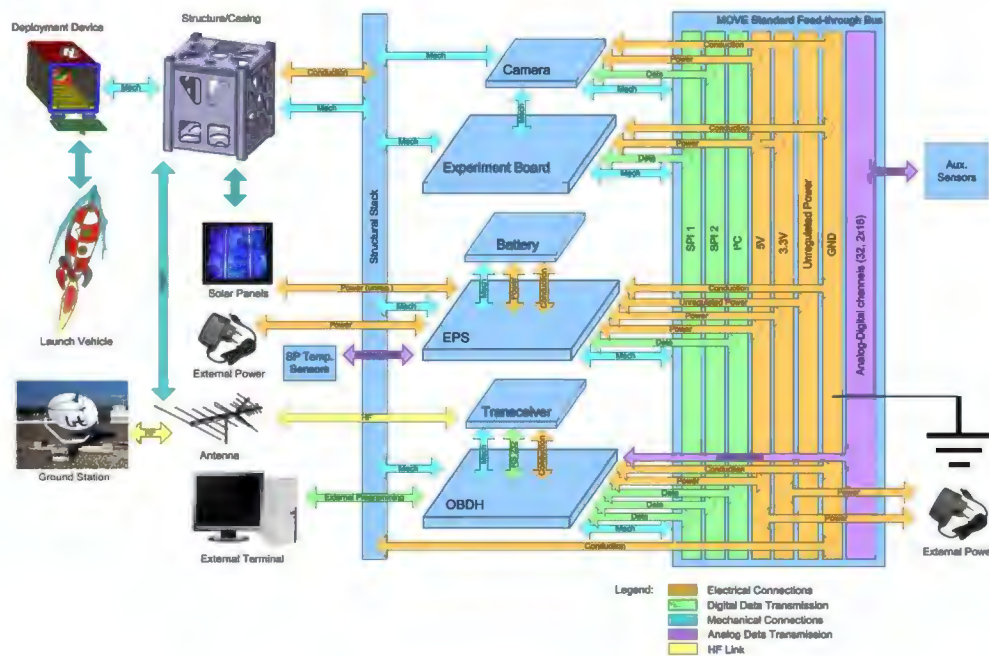


Fig. 6-1 Product architecture

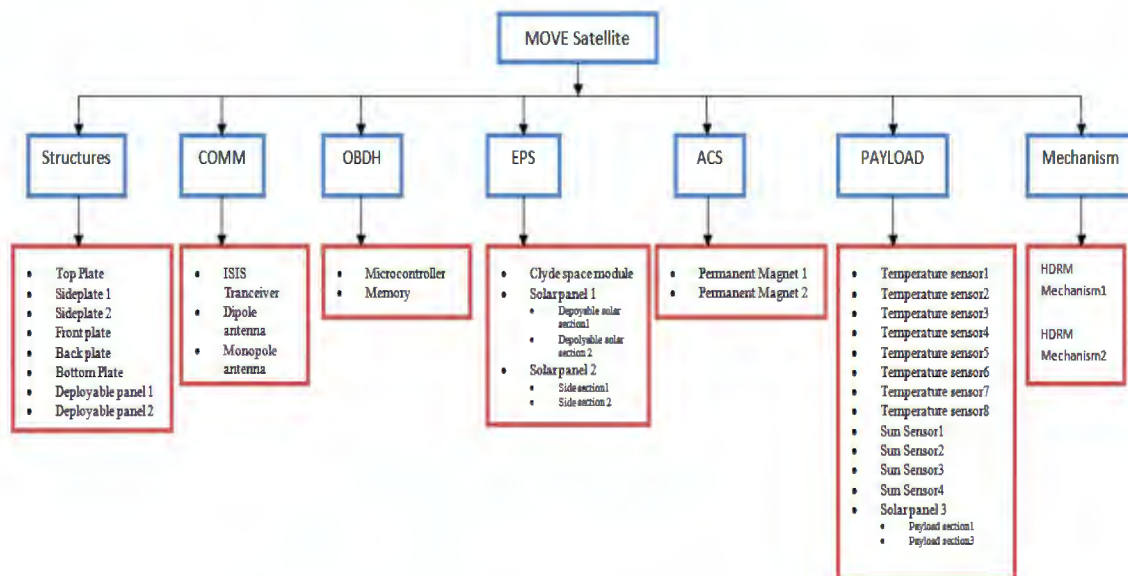


Fig. 6-2 Product functional layout



6.1 Electrical power system

The electrical power system (EPS) provides stores, distributes and control spacecraft electrical power. The most important sizing requirements are the demands for average and peak electrical power and the orbital profile. The power loads for mission operation at beginning and end of life are required. The usual procedure to assume the peak power requirement is multiplying the average power by 2 or 3 to obtain peak power requirement for payload, thermal and EPS. Fortunately all the system does not require the peak power at the same time during the mission.

The main task of the EPS is to supply the satellite with electric power during the whole mission. The requirements from the CubeSat mission MOVE where store and generate power, acquire and control housekeeping data, regulate and distribute power, protect components and payload. Also the system has to provide mechanical interface to the satellite structure, electrical interface to the subsystem and payload. Data interface to OBDH (On board Data Handling). The EPS must have the capacity of assuring enough power for the EPS module, COMM and OBDH, maximal additional power for experiments.

To verify these requirements, (v) - Sys software has been used. The (v) - Sys gives dimensional requirement of the entire satellite, which has to be satisfied by the subsystem product chosen from the database. The initial power budgets states the power required to survive was around 2.5 W and a minimum requirement of 0.8 W. The solar cells developed by Astrium has been planned to be used in the CubeSat. The Solar power generated from the solar panel with 4 Astrium cell configurations at 0 ° is estimated to be 4.1 W with a solar cell area of 301.6cm².

6.1.1 EPS Module

Since the MOVE satellite is a basic version of the CubeSat, a commercial off the shelf (COTS) electrical power system is used. After going through various modules for CubeSat application, Clyde space EPS was chosen. Clyde Space is one among the manufactures, which gives solution for Small satellite projects. The requirements of the mission have matched with the product from Clyde space specification.

In the product architecture, the total power consumed during the sunlight period and eclipse period are the inputs to the calculation. The Clyde Space EPS satisfies a power requirement from 1 W to 20 W of orbit average power. The EPS is specially built for nano/pico satellites. It provides a bus voltage of 3.3V to 5 V and a battery bus voltage up-to 10V. The EPS board is of a modular design. The Package consists of two lithium ion batteries with 2 cell configuration, with inbuilt Kapton heater. It has an inbuilt bus over current and battery under voltage protection.

The EPS module consists of the Battery charge regulators, Battery bus and an option of 5V regulator and 3.3V Regulator. The Charge generated from the solar sections is regulated using the Battery charge regulator and interfaced with the battery package through the Battery bus. The microcontroller monitors the function of BCR's and controls the regulator switches. The Clyde space EPS module has the facility to power the controller externally using a USB power.

Block Diagram

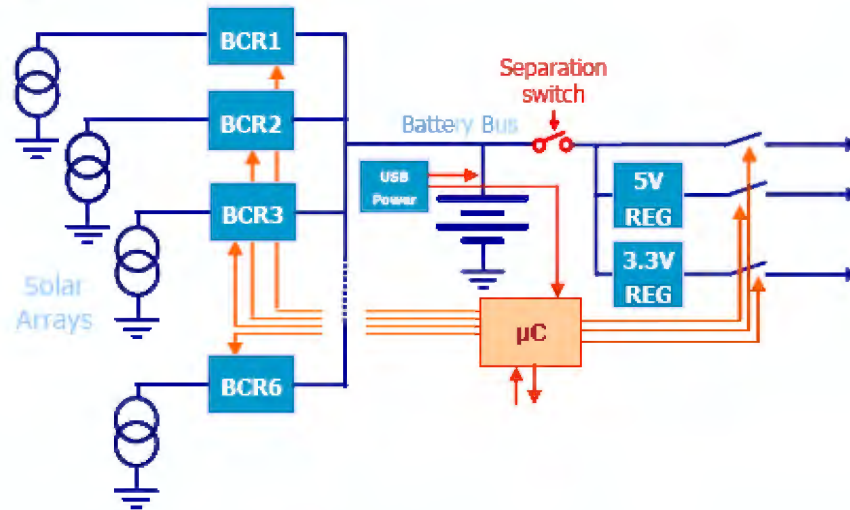


Fig. 6-3: EPS Module Block Diagram Layout

The CubeSat EPS peak power tracker actively monitors the characteristics of the solar array and sets BCR input voltage to the maximum power point of the array. The efficiency of each BCR power stage at full load is greater than 90%. The BCR will remain with its de-rated limits for an input voltage of 20V and an output voltage of 10V. There are 3 BCRs in total each capable of delivering up to 3W.

6.1.2 Solar Cells

The MOVE satellite uses the Solar cells from Astrium. EADS-Astrium solar cell consists of the base material of GaAs/GaInP/Ge on Ge substrate. The physical dimension of each cell of each cell is $40 \times 80 \text{ mm} \pm 0.1 \text{ mm}$ and an area of 30.18 cm^2 . The efficiency of the cell has been tested as 28%. The output voltage from each cell is estimated to be 2300mV and the current value to be 485 mA.

In (v)-Sys, the above parameters of the solar cell are chosen from the Excel datasheet. The orientation and the position of the solar cells and its electrical connectivity are to be optimized to get the best results. The figure below gives an idea about the power generation layout. The side section in red consists of 2 pairs of solar cells connected in parallel with the Battery charge Regulator (BCR1). The section in yellow have similar layout like the red section connected in parallel with the BCR2. The green sections are the Deployable panel of which 4 cells are connected in series on both the sides in parallel with the BCR3. The layout has been designed in such a way that, the power generators can generate the minimum required power when the satellite faces the sun in any possible direction. The BCRs can take an input voltage up-to 20V and gives an output maximum of 10 V.

In the MOVE satellite 16 solar cells are used, of which 12 cells are used for power generation and the remaining 4 are used as payload, which will be tested. The general solar panel consists of solar sections. The solar sections are composed of solar cells. In (v)-Sys, the solar panels have been modelled in a similar fashion.



Fig. 6-4 EADS/ASTRIUM Solar cell

Solar Panel Layout MOVE

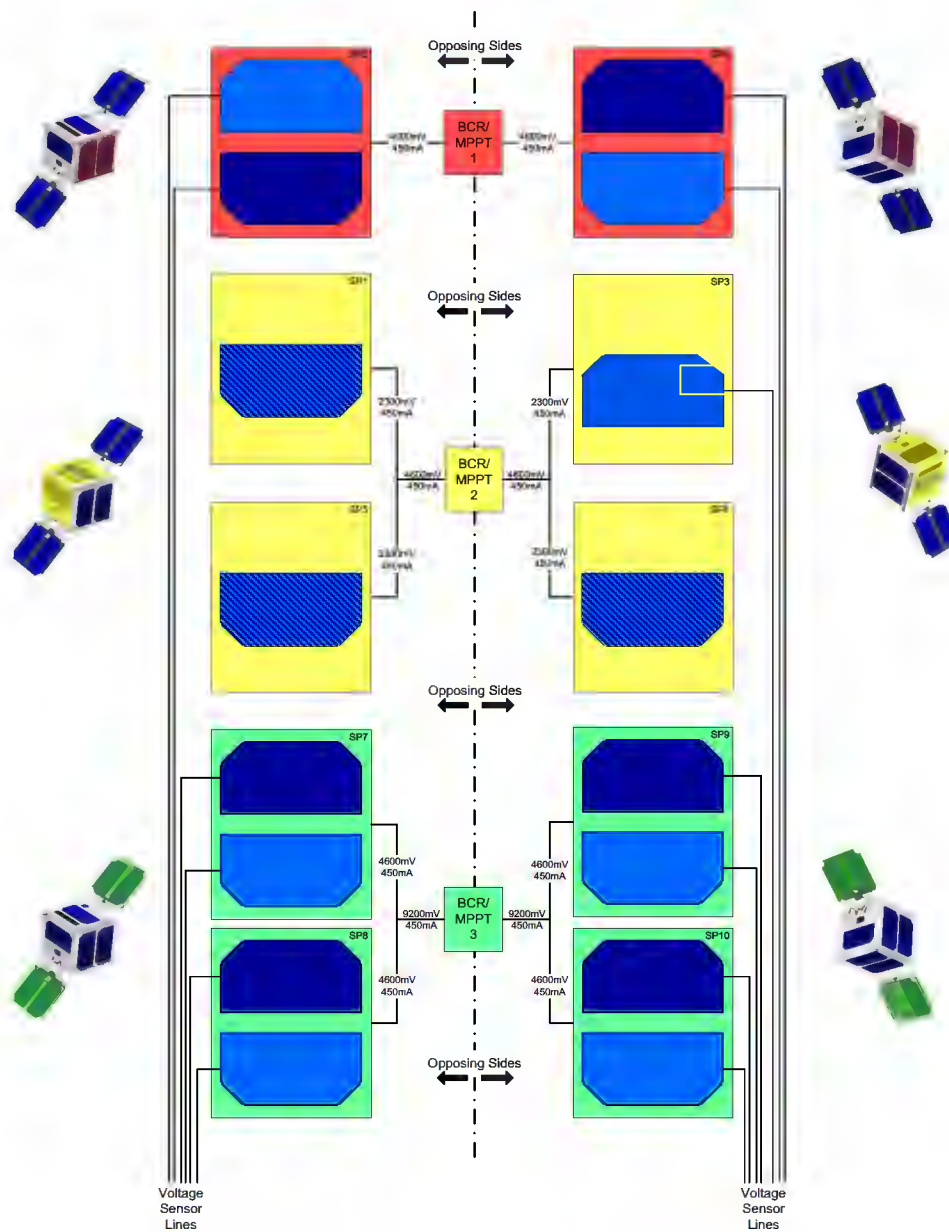


Fig. 6-5 Solar-cells Layout



6.2 Communication System

The communication system provides the interface between the spacecraft and ground station. Payload mission data and spacecraft housekeeping data pass from the spacecraft through this subsystem to operators and users at the operation centre.

In order to satisfy the link requirements, a commercial off the shelf module is chosen. The required data rates on a first assumption, from the payload/Telemetry have been 1200bps and the uplink telemetry data rates were calculated to be 300bps. After looking through various options, the Innovative Solutions in Space (ISIS) VHF/UHF nano-satellite transceiver module matched with the requirements. As on the basis of mass and size constraints the antennas were chosen as a monopole antenna for transmission and a dipole antenna for the reception.

6.2.1 VHF/UHF transceivers:

The module is capable of delivering high performance wireless serial communication in a variety of network topologies. When the module is properly configured and installed, long range communication at very high speeds can be achieved. The frequency selectivity with the receiver is much flexible to satisfy the uplink requirements. The transceiver consists of an efficient downlink modulation scheme, which uses BPSK modulation for Transmission and FSK for receiving the data. The transceiver consists of a flexible UHF receiver. The dimensions of the com board give a compatible form factor for CubeSat application.

The (v)-Sys model calculates bandwidth, equivalent isotropic radiated power, received power, received energy per bit, noise spectral density, bit error rate and actual carrier to noise ratio.

6.2.2 Dipole Antenna

The antenna is a key component of the satellite communication system. A satellite without a functioning antenna can be considered a dead satellite.

An isotropic radio transmitter radiates its power P_t equally in all directions (isotropically). At a given distance d from the transmitter, the transmitted power is distributed equally on the surface of a sphere with a radius d and an area $4\pi d^2$. The flux density S in W/m² of an isotropically radiating antenna at a given distance d can be calculated using the following formula:

$$S = \frac{P_t}{4\pi d^2} \quad (6-1)$$

A real isotropic radio transmitter does not exist, and any physical antenna will have some directivity, therefore the emitted power will be concentrated into a certain direction. Because of this, less than the full surface area of this sphere ($4 * \pi$ steradian) will be “illuminated”. The ratio between the full $4 * \pi$ steradian spherical coverage and the actually illuminated solid angle is called the Directivity and assumes that power is evenly distributed over and zero outside this area.

More important is the so-called antenna gain (G), which is the ratio of the flux density in a specific direction at a distance d and the flux density from the same transmitter using a hypothetical isotropic antenna.

At the receiving antenna, a so-called “effective area” is assumed, in order to calculate the amount of electromagnetic energy received. Using this Effective Area A_r , it is possible to calculate the total collected power $P_r = S * A_r$.

A receiving antenna has an antenna gain like a transmitting antenna, which is related to the effective area:

$$A_r = \frac{\lambda^2 G_r}{4\pi} \quad (6-2)$$

In this formula λ refers to the wavelength of the transmitted signal and G_r is the antenna gain of the receiving antenna. In addition to this formula it can be theoretically proven that the transmitting and receiving antenna gain is the same for the same antenna at the same frequency.

The antenna gain depends on the antenna design; while parabolic antennas feature antenna gains of 60 dB and more, the Yagi-type antenna has an antenna gain of some 20 dB (depending on the number of directing - elements in the design) and a dipole antenna has an antenna gain of 2 dB.

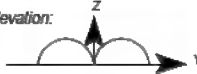
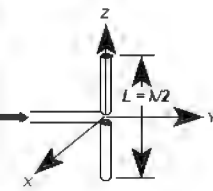
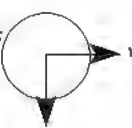
Antenna Type	Radiation Pattern	Characteristics
MONOPOLE 	Elevation:  Azimuth: 	Polarization: Linear Vertical as shown Typical Half-Power Beamwidth 45 deg x 360 deg Typical Gain: 2-6 dB at best Bandwidth: 10% or 1:1.1 Frequency Limit Lower: None Upper: None Remarks: Polarization changes to horizontal if rotated to horizontal
$\lambda/2$ DIPOLE 	Elevation:  Azimuth: 	Polarization: Linear Vertical as shown Typical Half-Power Beamwidth 60 deg x 360 deg Typical Gain: 2 dB Bandwidth: 10% or 1:1.1 Frequency Limit Lower: None Upper: 8 GHz (practical limit) Remarks: Pattern and lobing changes significantly with Lf. Used as a gain reference < 2 GHz

Fig. 6-6 Radiation patterns and its characteristics (Granite Island group, 2005)

As a very common and simple antenna the dipole antenna has proven to be reliable and sufficient within many working areas. The dipole antenna consists of one or more thin conductors. For a very small satellite it is reasonable to choose the $\lambda/2$ dipole. An antenna having a physical length that is one-half wavelength of the applied frequency is called a Hertz antenna or a half-wave dipole antenna. Hertz antennas are not found at frequencies below 2MHz because of the physical size needed of the antenna to represent a half-wave.

The (v) - Sys model for antenna gives the details of the antenna length, antenna gain. The antenna length is calculated in the Excel sheet, where the formula is given as

$$\lambda = \frac{2c}{f} \quad (6-3)$$

Where c is the velocity of light in vacuum and f is the frequency of communication. The gain for the dipole antenna is to be assumed since the directivity of the antenna is on all directions isotropic antenna.

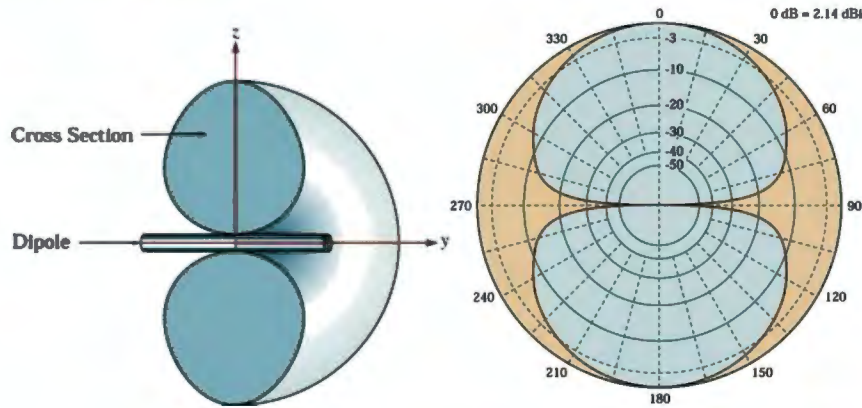


Fig. 6-7 Illustration of Dipole antenna radiation

As a safe margin the gain of the antenna can be assumed as -10dB to 2 dB. Radiation pattern is an indication of radiated field strength around the antenna. Power radiated from a $\lambda/2$ dipole occurs at right angles to the antenna with no power emitting from the ends of the antenna. Optimum signal strength occurs at right angles or 180° from opposite the antenna

6.2.3 Monopole antenna

A monopole antenna is a type of radio antenna formed by replacing one half of a dipole antenna with a ground plane at right-angles to the remaining half. If the ground plane is large enough, the monopole behaves exactly like a dipole, as if its reflection in the ground plane formed the missing half of the dipole. The behaviour of the monopole antenna is similar to that of the dipole antenna. A similar Donut shaped radiation is emitted. Since the mission requires to have a dual frequency for transmission and reception. Hence a monopole antenna is added along with a dipole antenna to have a comfortable transmission.

6.3 Structure

The main structure of MOVE is a cube with the function to provide support to all the subsystems of the spacecraft. Furthermore it is an interface to the launch vehicle. MOVE has to satisfy the requirement of the CubeSat standard. The main external structure consists of 6 plates and 2 deployable panels. The top and bottom plates are the ones facing the CubeSats inside the deployer. The origin of the coordinate system is located at the centre of the bottom plate. The X-axis is the flight direction; the satellite will rotate along this direction. The positive Y axis points towards the front plate and the Z axis along the Top side wall. The internal layout, the idea is to stack the subsystems boards on top of each other with required offset. They are mounted on four rods that are fixed on top plate and the bottom plate. To select the required offset between the subsystem plates spacers are used. The structural modelling and analysis are made using CATIA and ANSYS. The (v)-Sys uses the parameters from this software as an input to calculate the translation matrix, inertial elements, Global centre of gravity and for calculating the solar torques generated by solar pressure. As a launch requirement the centre of gravity should not shift more than 2cm from the centre of origin.

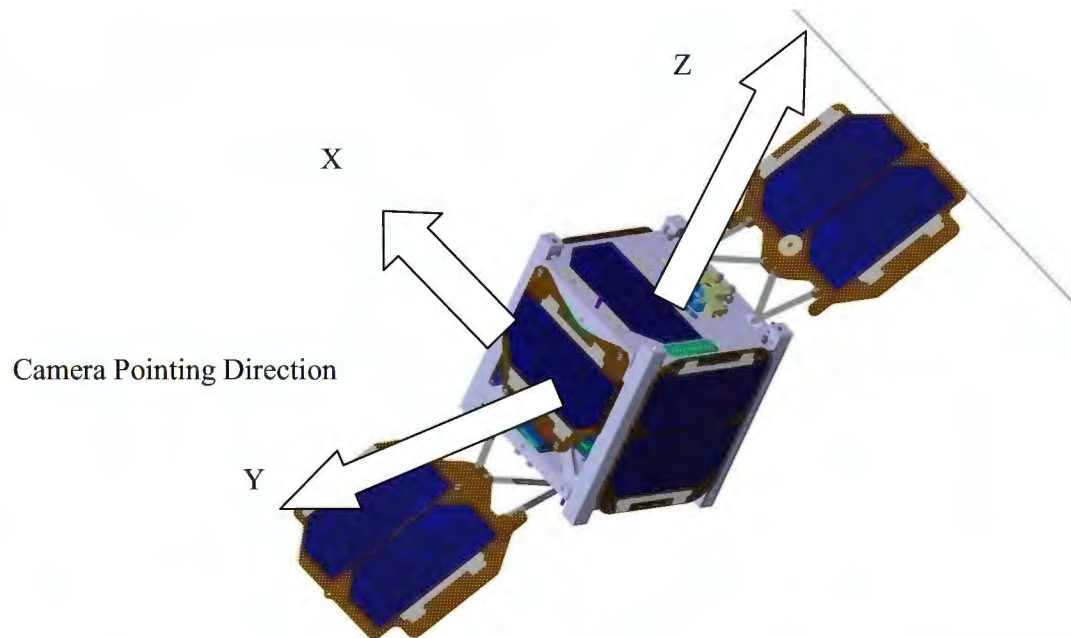


Fig. 6-8 MOVE Structural Model

6.4 Attitude Control System

Passive Controls response are simple and once brought into position they can be kept there with a small effort. But they are also restricted in their possible movements and often lance one axis unconstrained and therefore uncontrolled. Former satellites often used passive control mechanism. Passive magnetic control responds to external influences.

The permanent magnets are used to keep the X axis in orientation with the earth's magnetic field lines. As these lines are vertical at the poles and MOVE's orbit will be near Earth Polar, the satellite will perform a complete flip twice during every orbit, effectively reversing its flight direction each time. The most common magnet used for this purpose seems to be made of ALNICO 5, which is thermally stable and the same type will be used with a dipole moment of 0.023 Am^2 . The permanent magnets used in the mission should be to the dimensioned value, and then it remains safe from tumbling.

6.5 OBDH

The command and data handling system, C&DH, performs two major functions. It receives, validates, decodes, and distributes commands to other spacecraft systems and gathers, processes, and formats spacecraft housekeeping and mission data for downlink or use by an onboard computer. This equipment often includes additional functions, such as spacecraft timekeeping, computer health monitoring (watchdog), and security interfaces.

6.5.1 Micro Controller

The AT91SAM9260 is based on the integration of an ARM926EJ-S processor with fast ROM and RAM memories and a wide range of peripherals. The AT91SAM9260 embeds an Ethernet MAC, one USB Device Port, and a USB Host controller. It also integrates several standard peripherals, such as the USART, SPI, TWI, Timer Counters, Synchronous Serial Controller, ADC and multimedia card interface. The power requirements at different modes



and the data rates are important factors that are to be considered from the microcontroller module. The maximum power consumption of the microcontroller is approximately 330mW.

The important features of the AT91SAM9260 are as follows:

- High-performance 32-bit RISC Microcontroller with Thumb extensions,
- 32K Bytes ROM, 8K Bytes SRAM, USB 2.0 Device Port,
- USB 2.0 Host Single Port, Ethernet MAC 10/100 Base T,
- External Bus Interface, Bus Matrix, System Controller,
- One 4-channel 10-bit AD Converter, Three 32-bit PIO Controllers,
- Twenty two Peripheral DMA Channels (PDC),
- Four Universal Synchronous/Asynchronous Receiver Transmitters (USART),
- Two Three-channel 16-bit Timer/Counters (TC), Two-wire Interface (TWI),

The other subsystems also work on independent controllers, which come along with the subsystem package.

6.6 Mechanics

HDRM hold down mechanism is similar to a Pyro-cutter mechanism. While the MOVE satellite cannot be carried with the deployable panels open in the Pico-satellite dispensers, a mechanism is required to open the Deployable panel after ejection from the dispenser.

6.7 Payload

6.7.1 Camera

The satellite carries a 1.3 megapixels camera as a payload. The camera stores picture data in a 10bit format; it can be interpolated to higher colour schemes in ground. The camera can take pictures with different resolution. The resolution, most commonly used are 800 X 600, 1280X1024 etc. The 800 X 600 consumes 0.57 MB with 10 bit format, whereas for a resolution of 1052 X 864 consumes 1.56 MB of memory. Using a baud rate of 1200 bits/sec, it takes at-least 40 min to transmit 1 picture. Since the orbital time period is 98 min of which it can communicate with the ground station for 12 min per access. By using image compression methods the file size can be reduced to 70 KB. A detailed explanation on the camera design and the data storage requirements and visualization are discussed in the coming section.

6.7.2 Sun sensor

As a part of the payload the 2 axis sun sensor from Technical University of Denmark is used. Keeping in mind about the attitude determination system for CubeSat, this MEMS (Micro Electro Mechanical System) device is used. MEMS devices possess a number of features advantageous for space applications; e.g. low weight, high mechanical accuracy and robustness. The sensor is a slit type sun sensor with triangular photodiodes. It is build up by two major parts namely the photo diodes and a lid (Jan H. Hales). The photo diodes are fabricated on a SOI type silicon wafer. The lid covering the diode has a well-defined optical slit made with a thin film gold layer. The sensor measures 7x8 mm².

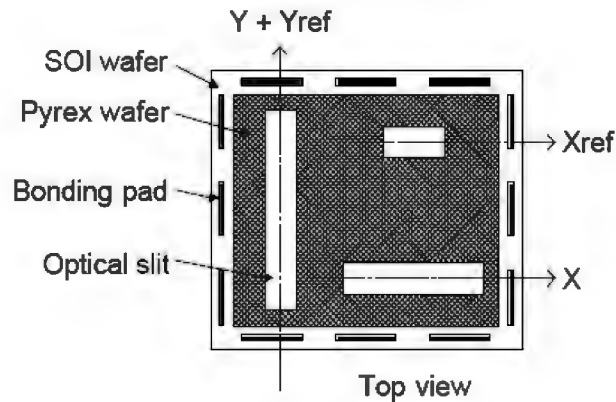


Fig. 6-9 2 Axis Sun Sensor Layout

6.8 Conclusion

The chapter 1 to 6 gives a detailed explanation about CubeSats, (v) - Sys and the mission MOVE. In this chapter, a thorough explanation on various subsystems like the electrical power, structure, communication system, on board data handling and passive attitude control and payload are given, since these subsystems will be verified in the following chapters. This chapter also gives the idea of how the product architecture looks like in the (v)-Sys modelling. The general parameters of the subsystem and its elements are also discussed in this chapter. The (v) – Sys software is capable of conducting static analysis only, hence to drive out a effective analysis, the following chapter will introduce the usage of the dynamic simulation software for the visualization application and based on the results a dynamic scenario will be established to estimate the satellites behaviour with respect to power, link and data budgets and verify the mission aspects in (v) - Sys model.

7 Modelling and Simulation in SIMVIS

Nearly all spacecraft missions involve sensing or interaction with the world around them, so a spacecraft needs to know or control its orientation. In general spacecrafts have payload like camera, sensors etc, and communication device like antennas pointing towards a particular direction.

The payload of MOVE consists of a camera and sun sensor and two antennas for communication a dipole and a monopole antenna. In this chapter, a visualization study will be conducted with the existing functionalities of SIMVIS and STK.

7.1 Introduction to SIMVIS

SIMVIS is a Simulation and Visualization tool developed by VEGA IT GmbH for ESTEC Operations. The overall architecture of SIMVIS includes a Designer to define a simulation and a Run time Environment to execute it. Data generated by the simulation can be visualized at run time using External Visualization application that connects to an External Simulation API. The interface to the concurrent Engineering process is through a Workbook, which maintained using Microsoft Excel. Using a Data processor, data from the Run Time Environment can as well be fed back into the workbook.

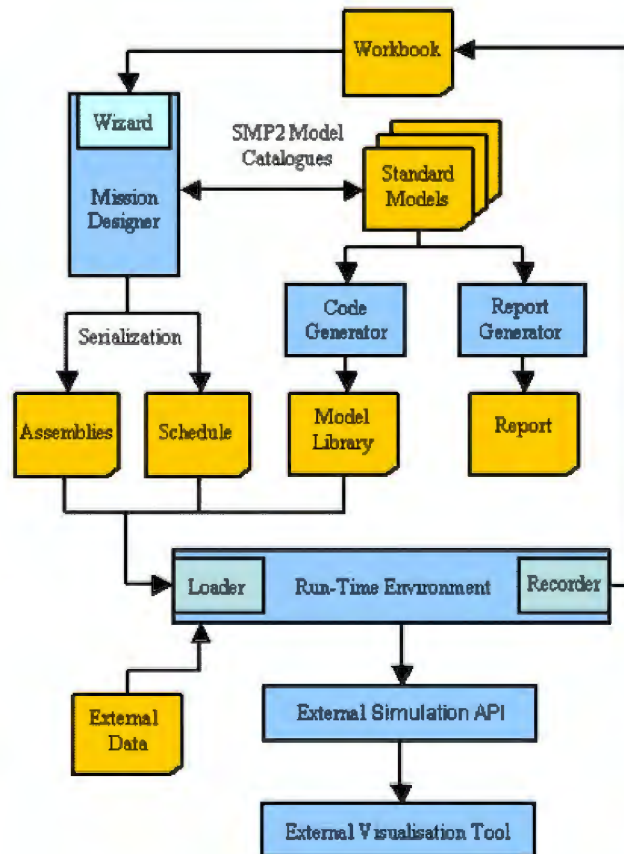


Fig. 7-1 Software Architecture of SIMVIS

The SIMVIS consists of Designer, Run Time Environment and Data processor.



7.1.1 Designer

The Designer is one of the main components of the simulation system. The main task includes,

- Importing of Named Cell Data from the Excel Workbook
- Assembling of Models for the initial Simulation configuration
- Preparing of Simulations for the run time behaviour of the simulator.

The Designer has a modern Human Machine Interface (**HMI**), provides access to the documentation, and makes use of advanced features, including Code Generator, Report Generator, and Wizards. These Wizards include a Mission Wizard, a Simulation Wizard, a Spacecraft Wizard, and a Subsystem Wizard (together called the Project Wizard).

7.2 Simulation of MOVE Mission

This chapter deals with the visualization of, the antenna, ground station interaction and camera contact with respect to Earth. Since (v) - Sys cannot do these calculations; the project requires standard software to analyze these two interactions. Hence SIMVIS was chosen as a tool to analyze these two cases. Even though SIMVIS has the capability to model huge satellites, the MOVE doesn't require all the capabilities of SIMVIS. Since it is new software for the author, a verification tool AGI STK was used to convince the authors understanding and to add weight to the analysis and discussion.

7.2.1 Orbital Definition

The orbital definition in SIMVIS is through an excel workbook. The SIMVIS Excel workbook consists of different sections of spacecraft subsystem. The Workbook starts with the Mission Overview. This Sheet allows entering Name, Type, and Description of the Mission, but its main purpose is to provide an easy navigation mechanism to all other sheets in the workbook.

The pre-defined Mission Types are:

1. Astronomy
2. Earth Observation
3. Communications
4. Interplanetary
5. Lunar
6. Navigation
7. Planet Observation
8. Celestial Body

Each Mission Type is linked to a certain Spacecraft Orbit Type, Solar System Model and Solar System Orbit Source.

Mission management:

The start and end of the mission are initialized with description. The Simulation Start Time defines the Epoch Time when the simulation starts and the Simulation End Time defines how long the Simulation will run. When the simulation reaches the end the Time Manager Model will stop the simulation.

Each Mission consists of different phases. For the MOVE mission, it consists of 3 phases. The first phase will be the launch phase, where the power subsystem is switched off. The 2nd phase

is the Commissioning phase, where the satellite is released from the P-POD. The 3rd phase is the orbiting phase, where the satellite starts to orbit. For the MOVE mission's simulation, the simulation starts from the orbiting phase, since the other two phases are of not much importance. The next step is the orbital parameter. The orbital parameters require,

Semi-major Axis:

The semi-major axis defines the size of the orbit. The value of the semi-major axis is 7003Km, as discussed in the previous chapter of (v)-Sys Simulation.

Inclination:

The inclination determines the slope of the orbital plane with respect to the Equator. The inclination was assumed to be 97.751° . The inclination is the angle between the angular momentum vector and the Z direction of the Unit vector.

Eccentricity:

The eccentricity of the orbit describes the flatness of the orbit. Since the mission requires a polar orbit, the orbit will be a Circular orbit; hence the eccentricity will be 0.

Right Ascension of the Ascending Node:

The right Ascension of the Ascending Node is measure of how the orbit turned in the east west direction. The SIMVIS doesn't have the capability to calculate the RAAN from the given inclination and semi-major axis, hence this parameter is can varied based on the requirements.

True Anomaly:

The True Anomaly indicates the position of the satellite along the orbit. The true anomaly is also an input that needs to be assumed.

The three important parameters that need to be defined to initiate the simulation process are the Semi-major axis, Eccentricity and the inclination.

The screenshot displays the SIMVIS Mission Configuration sheet. It includes a top navigation bar with tabs: Overview, HIS, CB, SON, SCE, GS, PWR, BAT, SA, SP, AGN, SEN, ACT, ANT, REC, TRS, TB, RAD, HEA, and DM. The main content area is divided into several sections:

- Overview:** Mission Name: MOVE Cubesat, Type: Earth Observation, Description: MOVE is an educational project for realization of a Cube Sat mission at the "Lehrstuhl für Raumfahrttechnik". The first satellite "First-MOVE" will verify the satellite platform and new highly efficient solar cells from industry. It is a Cube Sat in single-configuration (selected) which features deployable solar arrays. This design.
- Management:** Management
- Environment:** Celestial Bodies, Orbit and Altitude
- Ground Segment:** Ground Segment
- Spacecraft Modes:** Spacecraft Modes
- Power:** Batteries, Solar Arrays, Solar Panels
- Thermal:** Thermal, Radiators
- Communication:** Antennas, Receivers, Transmitters
- AACS:** Control Modes, Sensors, Actuators
- Propulsion:** Systems, Tanks
- Data Management:** Data Management
- Experiments:** Experiments

At the bottom, there is a table for Mission Types and Orbit Types:

Mission Types	Orbit Type	SSModel	Orbit Source
1 Astronomy	HEO	Use	Keppler Elements
2 Earth Observation	LEO	Use	Keppler Elements
3 Communications	GEO	Use	Keppler Elements
4 Interplanetary	Trajectory	Full	JPL Ephemeris Data
5 Lunar	Celestial Body	Use	Keppler Elements
6 Navigation	MEO	Use	Keppler Elements
7 Planet Observation	Celestial Body	Full	JPL Ephemeris Data
8 Celestial Body	Celestial Body	Full	JPL Ephemeris Data

Fig. 7-2 Mission Configuration sheet



7.2.2 Spacecraft Orientation in SIMVIS

The spacecraft orientation is defined by the attitude pointing with respect to the Earth's frame. Each Attitude Pointing Mode is defined by its unique Name, Type, Pointing Axis, Target Name and Spin Rate. Additionally a Description can be entered. The Type is selected from the following set of available types that are supported by a corresponding model:

1. Star
2. Celestial Body
3. Position (on the surface of a Celestial Body)
4. Inertial Position
5. Ground Station

The **Pointing Axis** defines the axis of the spacecraft that points to the target. The three major axes plus their negative directions are available. The **Spin Rate** defines the spin the spacecraft rotates with around the pointing axis. This is an optional value. Note that an attitude pointing mode either defines a spin rate or uses a Third Axis Control model to define the rotation around the pointing axis.

For the MOVE mission, an Inertial Position type was chosen with different spin-rate. Since the satellite is passively controlled, it spins along the Y Axis, where the X axis and the Z axis are aligned with the Earth's Magnetic field. The rotation of the CubeSat is decided by the spin rate, where the spin rate is decided by the External torques and forces. Hence the spin rate cannot be determined exactly. The spin rate becomes a playable value to determine the contact. In this analysis the spin-rate was assumed to be 1 to 2 rev/minute.

7.2.3 Antenna and Camera Orientation

A sheet for a model-type that inherits from the Hardware Component model type always provides the cells for the Entity model-type plus cells for the Structural Parent Name, the Position and Orientation, the Mass, the Centre of Gravity (CoG) and the Moment of Inertia (MoI) of the model instances. The first table holds the Structural Parent Name that defines a model instance that represents the structural parent of the current model instance. The Position and the Orientation are defined relative to the reference frame of this Structural Parent. In case there is no Structural Parent given, the Spacecraft main body is taken as structural parent. The Orientation is defined in a triplet of Euler angles (ZYX). The corresponding orientation in quaternion's is automatically calculated and shown right to the table. As there are different interpretations on how Euler angles are applied, there is an explanation on how the wizards and the SIMVIS models interpret the Euler angles in the following. The easiest way to think about an Euler rotation might be to think of three rotations that are applied in the order of the letters with a rotating co-ordinate system. Rotating co-ordinate system means that the second rotation is relative to the co-ordinate system that results from the first rotation. The third rotation is relative to the coordinate system that results from the second rotation.

The Figure below shows the orientation of the satellite MOVE dipole-antenna. The direction of A1 is -45° about the Y axis. The direction of rotation will along ZYZ. Hence for the dipole antenna, the orientation will be $(-45 \ 0 \ 0)$. Based on the given configuration the quaternions are calculated.

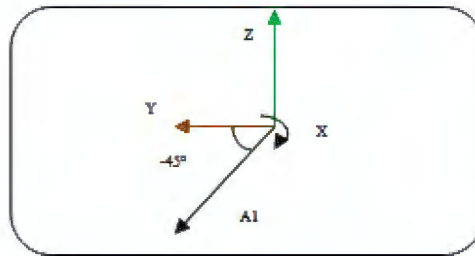


Fig. 7-3 Dipole antenna Orientation

The figure below shows the orientation of the Monopole antenna used for the transmission working at 146 MHz. A similar approach is used, where the X axis needs to be rotated in an anti-clock wise direction to attain the base reference. The A2 orientation is given by the Euler angles (45 0 0). The respective quaternions are calculated.

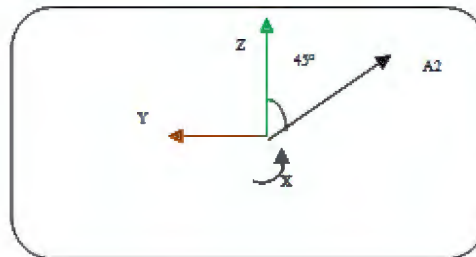


Fig. 7-4 Monopole Antenna Orientation

After updating the respective orientation and positions of the elements to be analysed, the Excel file has to be saved in a mission folder.

7.2.4 SIMVIS Designer Setting

After setting up the Excel sheet, the SIMVIS Designer has to be opened to read the input from the excel sheet. The SIMVIS designer is composed of different settings. Mission, Assembly, catalogue, schedules' and configuration, though schedules and configuration doesn't require much concentration. The catalogue needs to be set before opening the wizard. The *SIMVIS.dll*, *Smp.dll* and *smptool.dll* needs to be imported to have the basic functionalities of the spacecraft.

To import the excel sheet, the Spacecraft wizard needs to be activated. Once on activating the wizard, it asks for the excel sheet. The settings or the inputs given in the excel sheet can be verified in the wizard. There is always a possibility of misreading the excel sheet. Here care must be taken every time when wizard is initiated.

The wizard automatically calls for the catalogue functions. The catalogue consists of the different subsystem functions of the spacecraft. The subsystems are immediately generated in the Assembly section of the designer. Further settings needs to be done in the assembly section. The wizard calls all the parent external files from the type of mission chosen. In this case the Earth Observation mission is chosen.



To set the target for the Antenna and camera, the assembly section needs to be update. The target function needs to be imported under the Antenna and Camera. The targets can be of a Ground station, celestial body or a point on reference. For the antenna, Dipole antenna and the monopole antenna the Munich ground station was chosen.

The ground station setup requires the latitude, longitude and the altitude of the earth on the earth inertial frame. These are given as input either in the excel sheet or in the designer assembly itself. The ground stations can be referred to the target function under the antenna. The camera needs the similar kind of setup, but referring to the celestial body Earth. So when the camera normal orients with the earth the *isVisible* variable under the Target function is set to TRUE. It is of the similar variable used for the antennas too.

SIMVIS has the option of visualizing in the 3D domain. Hence a function called Visualization needs to be included to the assembly. The visualization needs to be initializes in its function. The setup requires the central body Sun, Inertial frame Earth and the satellite. To visualize the field of view cone of the elements on the 3D OpenGL, the satellite elements need to be initialized in the XML file. That will be dealt in the further sections.

After setting up the assembly the mission section needs to be set before the simsat is started. An extra excel sheet needs to be imported for recording the simulated values. After importing the Excel sheet, the recorder needs to be called under the simulation. The recorder consists of variable. The main parameters of the recorder are the time of start, time of stop and the time step. Based on the requirements the output values needs to be segregated between recorders. The Simulator can have 1 or more recorders. The disadvantage of doing this is, as many recorders the mission has, that replicates on the system speed. Hence care must be taken while creating the recorder. The simulation time step is also a reason for the reduction in system speed. The lesser time the mission has, more complexity increases. The variables that are to be recorded needs to be called and linked with the Assembly. For example the ground station visibility is given as *isVisible* under the target function of the Antenna. Similar to that of the Camera the *isVisible* variable is chosen under the camera target function.

The simulator is called with the *Load* function to load the MMI to the SimSat. Before loading to the SimSat, the mission, assembly and the scheduler needs to be saved to start the SimSat.

7.2.5 SimSat Setup

The SimSat is the simulator attached with the SIMVIS software as mentioned earlier. SIMSAT 2000 is the new ESOC infrastructure designed to aid the production of real-time simulations of satellites. It is designed to run under the Windows 2000 Server operating system on the Intel platform.

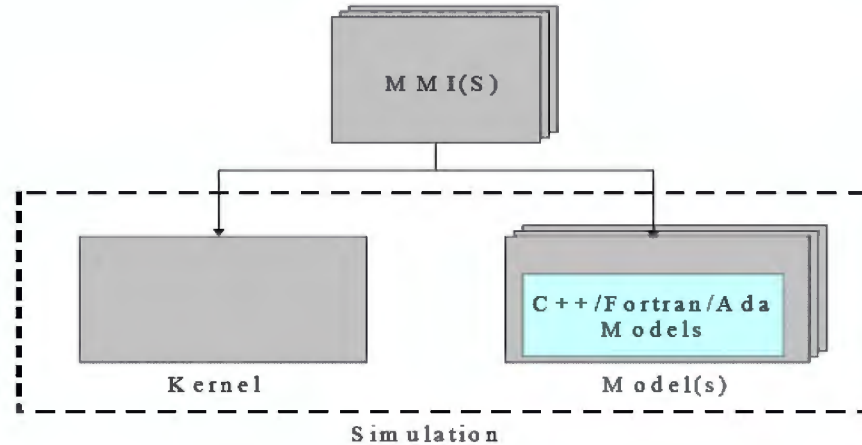


Fig. 7-5 Simsat Architecture

Figure 1 shows the major components that comprise a SIMSAT-2000 system. A simulation comprises of the simulation Kernel and any number of models, controlled by the Kernel. SIMSAT-2000 has been specifically designed to support C/C++ models, although models written in other languages, such as FORTRAN, can be supported. The simulation Kernel and its associated models are controlled by one or more MMI's connected to the simulation.

The SIMVIS Designer creates SimSat file which is used for initialization of the SimSat. Once the file is created the SIMVIS designer has the option to connect the MMI with the simulator. The SimSat is directly activated from the designer to ensure the linkage between the designer and the SimSat. If the MMI is not linked, the SimSat gives an error message of the respective file which creates the crisis. The SimSat consist of the AND file, which converts the alphanumeric data in to the user know data. Hence the values that are read in the AND file does not have the decimal points for the direct understanding. The user has to interpret the values generated. The online simulated data's can be read using a graph plot option as in MATLAB or LABVIEW.

Even though the dynamically varying values are plotted in the graph, the readability makes it inconsistent. The other option of recording these generated values is through the option called Recorder in the Designer. The settings for the recorder are done in the Designer by interfacing the recorder to an *Xml* or *csv* file type. *csv* file type can be read using the Excel sheet. An assembly file is automatically generated when a recorder is created. The recorder consists of variables. The variables can be chosen from or interlinked with the different existing assembly model. For example, Antenna target, Camera target, Power generated values and varying satellite orientation with respect to time. All these values can also be plotted, but as mentioned before due to inconsistency we need to use the recorder option to present the results.

Once on setting the SimSat, before starting the simulation the Epoch need to be set. Even though the Epoch time is given as an input from the Excel sheet, the user gets the freedom to change the Epoch time. The epoch time decides the position of Earth with respect to sun. Hence the mission start time needs to be selected to start the mission; In this case, January 1st 2009 was chosen. Once the simulation starts, the simulator time is synchronized with the

system time. The time factor can be increased. But the disadvantage in SIMVIS is the Time slip. The time slip plays a major role in the results of the simulation. As the Time factor is increased, the time slip between the earth revolution and the satellite orbit differs. Normally this leads to chaos. This disadvantage will be dealt in the concluding sections.

7.2.6 Visualization Setup

The Visualization tool works on XML programming. This uses the most predominantly used graphic viewer OpenGL. Whenever the mission folder is created using the Wizard, an *.igs is generated. The igs file consists of the basic setting to view the spacecraft, orbit, and the target elements (ex. Sensor, camera, antenna etc). The orientations of these elements have to be rewritten as it was set in the excel-sheet. Then the field of view cone needs to be set using its opening angle, orientation matching with the element and the colour of the cone to differentiate. The satellite 3D model can be changed but it requires a file conversion from a local 3D CAD model to the Visualization model. The file can be imported through the igs file. Even though setting up the visualization is easy, the user needs to have a basic knowledge in XML programming to understand the generated code or update the code to the requirements.

7.3 Ground Station Visibility

Each Ground Station is defined by its unique Name, its Latitude, Longitude, Altitude and Masking Angle. The current Workbook supports up to 20 Ground Stations. Figure 4.5 shows an example set of Ground Stations. The Garching Munich Ground station is set to be 48°N latitude and 11°E longitude. The masking angle is set to the worst case of 10° which is otherwise called the minimum Elevation.

	Name	Latitude [deg]	Longitude [deg]	Altitude [m]	Masking Angle [deg]
1	Kourou	5,25143694	-52,80466242	14,561	10,00
2	Kiruna	67,85712518	20,96434169	402,275	10,00
3	Perth	-31,80252491	115,88515564	22,156	10,00
4	Munich	48,08000000	11,35000000	560,000	10,00
5					
6					
7					
8					
9					
10					
11					
12					
13					
14					
15					
16					
17					
18					
19					
20					

Fig. 7-6 Ground station setting MOVE configuration sheet

The ground-station has to be called as a target to the satellite antenna and the satellite to be called as a target to the Ground station. On creating a target to each of the mission element, a variable called *isVisible* is generated. On calling the variable *isVisible* to recorder, the ground-

tracking timings can be recorded in the csv format. As explained before in order to visualize it, the orientation of the antennas needs to be verified in the igs file. The outcome of the Ground tracking visibility study should be the transmission time, number of contacts and the frequency of losing contact with the ground station and vice versa.

7.4 Camera Visibility

The camera visibility studies require the orientation of the camera with respect to the satellite as explained under antenna and camera orientation. The camera visibility gives an idea of how long the camera will be in view targeting the earth surface. Setting up the camera function is similar to that of the antenna. The figure shows the screen shot from the SIMVIS designer tool for the MOVE mission.

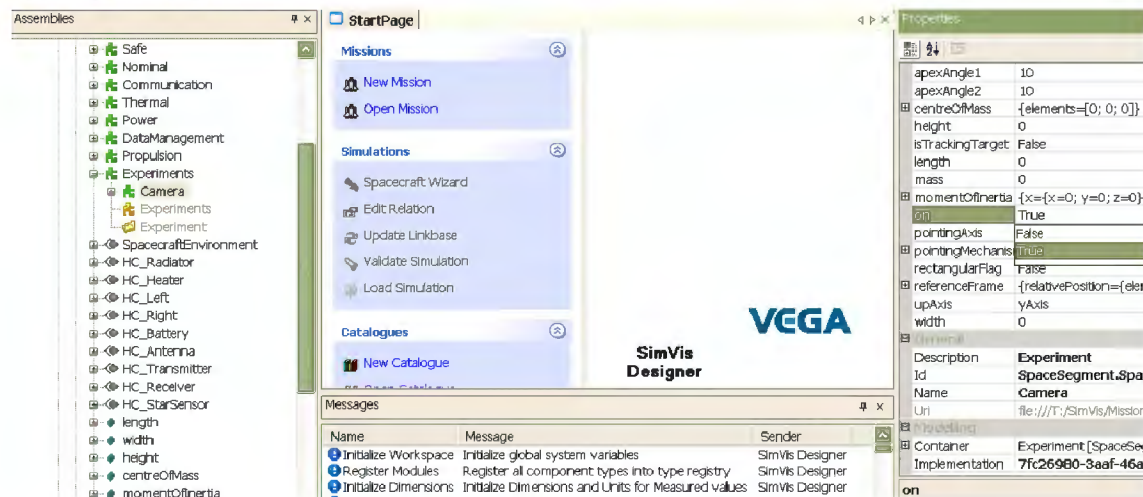


Fig. 7-7 SIMVIS designer

7.5 STK Tool

Satellite Tool Kit is the leading commercial spacecraft analysis and visualization tool. The program is structured around a base module whose core capabilities include generation of position and attitude data, analysis of spacecraft fields-of-view, acquisition time, and sensor coverage. The base module is useful in that it allows orbits to be modelled examining a number of critical factors that affect a number of the spacecraft subsystems. These factors include sensor coverage areas, frequency and length of ground station acquisition periods, and duration and frequency of eclipses. In addition, the module allows the attitude configuration and the orbit path to be visually examined, providing the systems engineer with an additional perspective through which a better grasp of the situation can be obtained. In addition to the core module, the program also has the capability to add other modules which enhance the capabilities of the core module. Analytical Graphics, Inc. (AGI), the manufacturers of STK, provide additional modules to increase the capabilities of the base module. Some of the modules that are useful, from the perspective of the S₂C₂ Functional Division include Astrogator, Attitude, and Coverage.

7.5.1 Setting -Up STK

The satellite tool kit software is the most predominantly used simulator by most of the research institutes and Organisations. Modelling a satellite mission is much more easier compared to other software's. For the MOVE mission the STK tool is used as a verification

tool for comparing the output from SIMVIS. To start with the mission setting, the *Facility* has to be created. The facility can be of a point on the globe. Here in this case, it will be the ground station. The Ground station or the *facility* is defined using the latitude, longitude and altitude. Once the facility is defined, then a satellite has to be created. To create a satellite or a spacecraft, the orbital parameters are to be defined. The orbital parameters are defined as in (v) - Sys and in SIMVIS. The attitude of the satellite is also to be defined. The coordinate systems have to be chosen. Most of the time inertial coordinate system is used to define the satellite attitude.

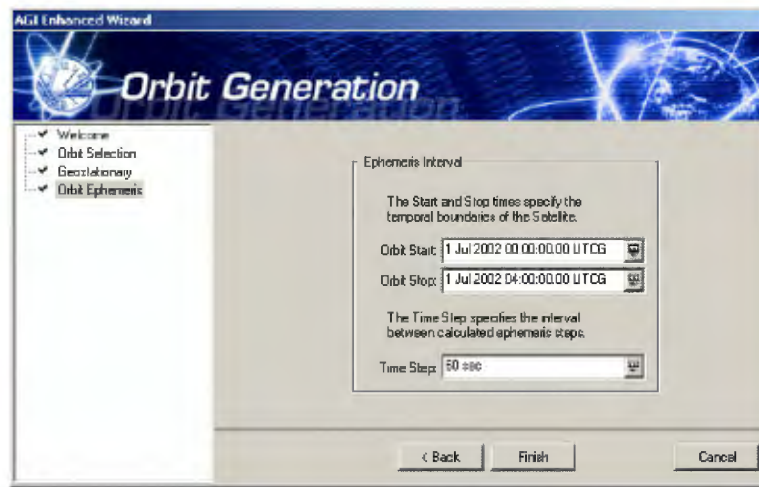


Fig. 7-8 STK Orbit generator

The orientation of the satellite with respect to the inertial frame can be defined either using the Euler angle or quaternions. With reference to the standard orientation the MOVE satellite is defined. The STK has a special feature compared to the SIMVIS, i.e. the spin rate of the satellite can be defined and the stability criterion can also be defined. Since the MOVE satellite is passively controlled, the satellite is expected to rotate along the velocity vector. In general, even though the satellite is passively controlled, still the satellite spin leads to tumbling. The tumbling factor can also be defined in STK, whereas SIMVIS cannot. On setting up the MOVE satellite, the elements that are of concern need to be defined. The antennas and camera are the prospective elements in consideration. Setting up the antenna is of the same as SIMVIS. The orientation of the antenna are to be given and the same with the camera.

The Antenna and the camera are defined with their Euler angles as defined in SIMVIS. The extra feature in STK is setting up the dipole radiation pattern. Whereas in SIMVIS the field of view cones are to be set in the *igs* file. The half wave dipole pattern and the monopole pattern can be well defined using the STK. The beam pattern will vary with the frequency, transmitter power and the Antenna gain. For the camera the normal cone can be set with a half cone angle of 45° , which opens up to 90° . The STK also helped in finding out the contact time in comparison to the elevation angles. Even more precise outputs of the antenna and camera contact times can be simulated with STK. This thesis is restricted to work with Antenna and Camera visualization in STK.

7.6 Results and Analysis

7.6.1 SIMVIS Results

The initial parameters to simulate the ground track over the ground station were set in the SIMVIS designer and the SimSat. To analyse the ground station entry time, 2 cases were chosen setting the start date to be Jan 1st 2009. The first case is that the simulation start time to 12'o clock and the second case the simulation start time to be 18'o clock. As mentioned in the earlier chapter, the initial orbital design value shows the 18'o clock orbit has better visibility of the satellite rather than the 9'o clock orbit. To make the verification, SIMVIS was used to analyse the same for the antenna visibility studies and camera visibility studies. Since the SIMVIS takes longer time to conduct the simulation, a shorter period of analysis was chosen. Even though, SIMVIS had the facility to increase the simulation pace, it equally had the disadvantage of time slip. Hence a short period of 7 days was chosen initially. As shown in the bar chart, the 18'o clock orbit had a better visibility than the 12'o clock orbit.

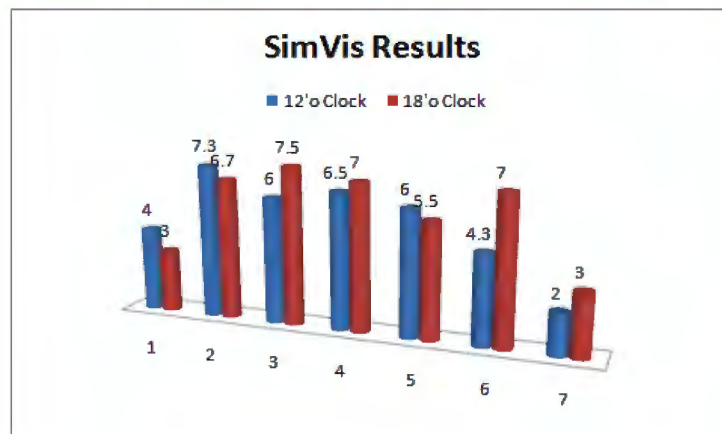


Fig. 7-9 SIMVIS Access study

The 7 day simulation gives a mean idea of the number of entries. Hence a 30 day simulation was conducted with a small increase in time factor. The chart below shows the simulation results for the first 30 days. On the whole the 18 o' clock orbit has more entries.

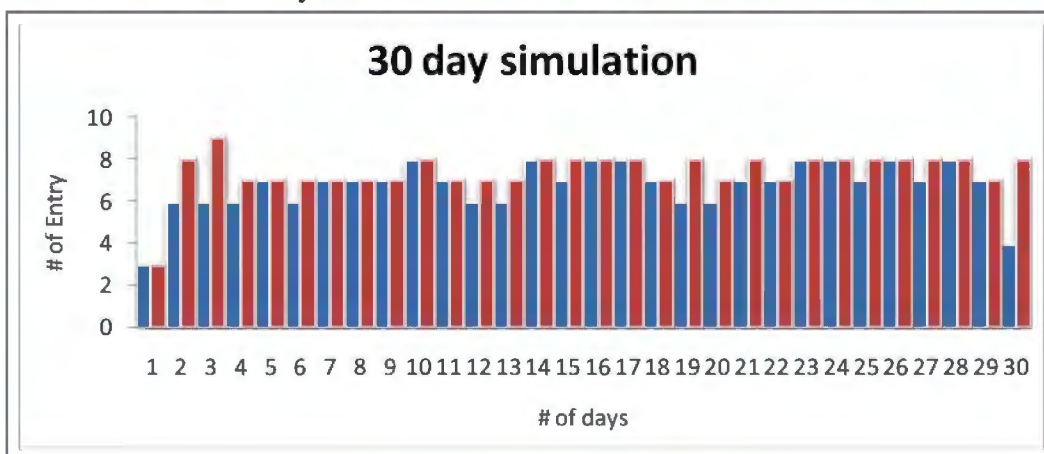


Fig. 7-10 SIMVIS 30 day simulation

The next levels of analysis will the total time per entry, hence a 24 hours scenario was assumed. Taking the best day from the 30 days analysis as 15th of January 2009 where 8 entries take place.

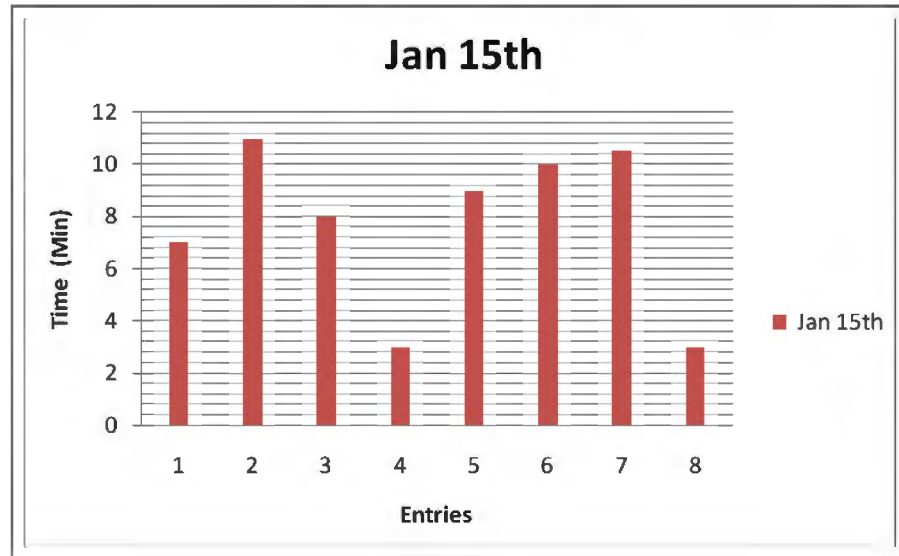


Fig. 7-11 Jan15th results

Since the SIMVIS doesn't have facility to visualize the ground track either on a 2D view or a 3D view to distinguish the eclipse period and the sunlight period, the solar intensity from the satellite point of view was recorded from the simulation. Hence the solar intensity values give an idea about the satellite entry in to the daylight and the satellite entry into the eclipse. The scenario can be well expressed when an average time scheme is developed.

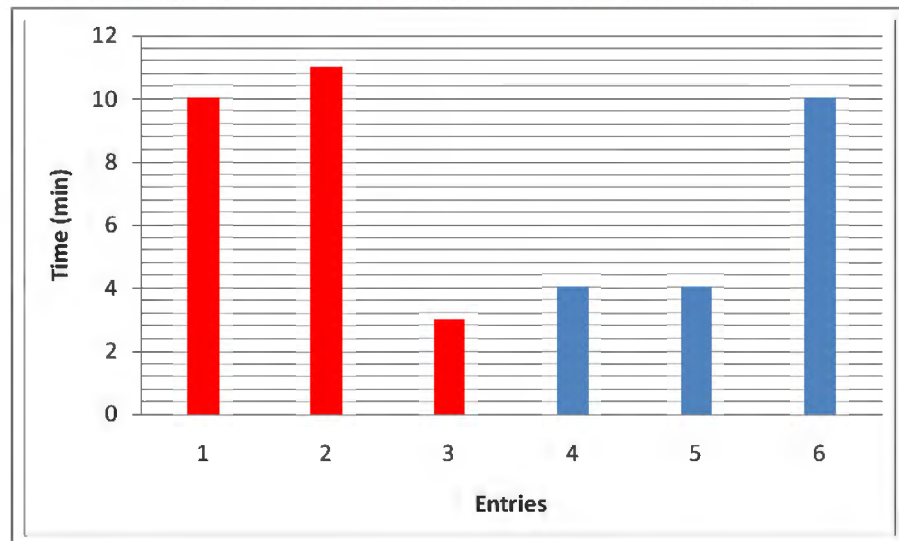


Fig. 7-12 Average access per day

The above figure shows the difference in the sunlight period and eclipse period on an average per day. Hence with the above data, the transmission can take place twice effectively. This will directly replicate the values of the antenna usage per day for transmission. But the above

analysis is fixed values for antenna visibility study. To have even more realistic values, the satellite spin has to be implemented.

Along the Y-axis a spin rate of $12^{\circ}/\text{sec}$ was assumed keeping in mind the earlier assumption of 2 rev/min, along the X axis a spin rate of $5^{\circ}/\text{sec}$ was assumed and along the Z-axis a spin rate of $3^{\circ}/\text{sec}$ was assumed. Setting up the opening angle of the antenna directed radiation was not possible with the SIMVIS. Whereas, on setting the spin-rate with respect the inertial axis in the SIMVIS designer, the simulator can recognize the values for its dynamic calculations and simulation. But it cannot be realized, in terms of a 2D view or a 3D view. On an average study on the link loss with respect to antenna, the satellite loses at least 2 min per access on the satellite tumbling for the dipole antenna.

Tab. 7-1 With tumbling & Without Tumbling (Dipole)

	Without Tumbling	With Tumbling
Access	Link time1 (min)	Link Time2 (min)
1	10	9.5
2	11	8.7
3	3	1.7
4	4	2.2
5	4	2.3
6	10	8.3
7	6	4.9
8	8	7.1
9	10	7.8

For the Monopole antenna the total time loss is much more than the dipole antenna, because of the radiation patterns and the orientation of the antenna. The total time loss with respect to the monopole antenna will be approximately 3-4 min in comparison to a non tumbling case. These are not the exact values since the radiation patterns cannot be imported or used in SIMVIS. These values can either be an assumption or an approximation from the obtained simulation results. Hence it requires a verification to use these values.

Tab. 7-2 With tumbling & Without Tumbling (Monopole)

	Without Tumbling	With Tumbling
Access	Link time1 (min)	Link Time2 (min)
1	10	5.5
2	11	6.1
3	3	1
4	4	1.5
5	4	1.5
6	10	5.7
7	6	3.5
8	8	4.5
9	10	6.1

A same kind of analysis was set for the camera also. During *Without Tumbling* mode the camera rarely get the opportunity to visualize the earth surface, since the camera is facing towards the Y-axis. Even then with an apex angle of 90° , the camera should be in view with the earth for at least few seconds. The time step for the antenna and the camera was initially set to 30 seconds, since to find out exactly the total time of the camera visibility, the time step was reduced down to 10 seconds. On doing this, the Earth was visible to the camera for maximum of 10 seconds for every access. In the tumbling scenario the, the camera visibility was fluctuating between 5 seconds to a maximum 19 seconds, which is more than enough to take picture of the earth surface. The total time in view was calculated to be 65min per orbit.

The visibility of the sun sensors were also studied using the SIMVIS software, since the MOVE team has plans to add a set of attitude determination sensors as a payload. 4 sun sensors are being planned to be placed in the CubeSat. The Sun Sensors are from Technical University of Denmark. The sensors are planned be positioned on the possible 4 sides of the cube to get the best results. The sun sensors have field of view of 120° . The results show that the sun sensor will be in view with the sun for 10 seconds every time when facing the sun. The sensors will be visible to sun only during the sunlight period.

7.6.2 STK Results

The STK software is much more sophisticated than the SIMVIS software. The results from the STK are as follows. The below figure show the 2Dview of the ground tracks with the ground station entries marked in black. The results of the STK access time include the least possible elevation angles also. Hence as per the access results achieved, it has at the least 11 entries per day including the elevation angle below 5° . On excluding the elevation angle less than 5° leads to have a minimum of 5 entries and a maximum of 8 entries. After increasing the elevation angle to 10° , the access reduces to a minimum of 4 entries to a maximum of 7 entries.

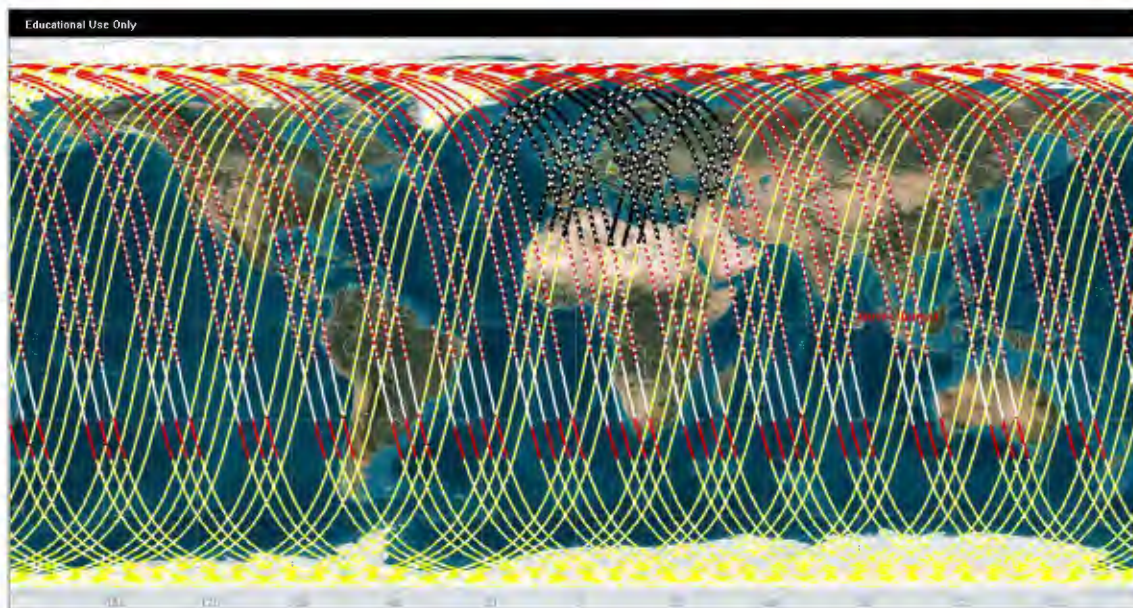


Fig. 7-13 Ground Track STK 625 SSO

The figure below gives an illustration of the ground tracks in a 3D view. The Yellow colour refers to the sunlight side and the red colour refers to the eclipse side. The eclipse period can

be differentiated into 2 parts, penumbra and umbra. The Penumbra is the transition period from the sunlight period to the eclipse period (partially dark), whereas the umbra period is a complete dark period. In this case the penumbra is a very short period compared to the umbra. The penumbra plays a major role when α angle is increased from 0 to 90° . Hence in this case the penumbra period is not of much relevance.

The simulation start time was fixed similar to that of the SIMVIS and initial 7 day analysis was conducted, access results are much similar to that of the SIMVIS results as shown in the bar chart below.

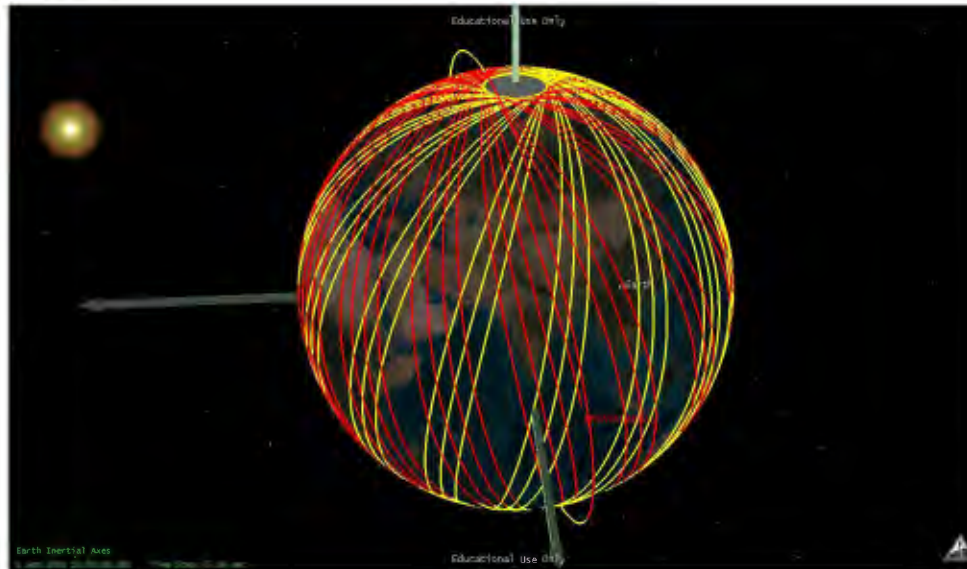


Fig. 7-14 3D view in STK

The simulation of the 18'o clock orbit was extended from 7 days to 30 days. The simulation values from the STK shows even more precise and accurate values compared to the SIMVIS, Above all, the interpretation of the results was much easier than the SIMVIS. The following charts and table shown are the results from the STK.

STK-results

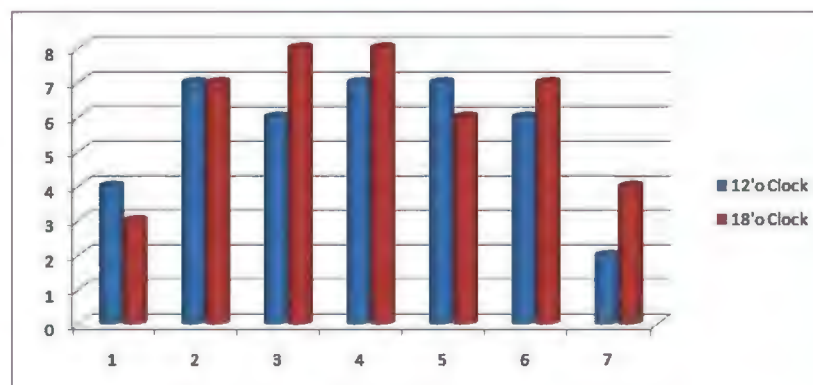


Fig. 7-15 Access study results in STK

The 30 days (Fig.9-15) Simulation shows a much similar profile to that of the SIMVIS results except few deviations. This shows the MOVE mission can expect an average access attempt of 7. The next levels of verification will the access time.

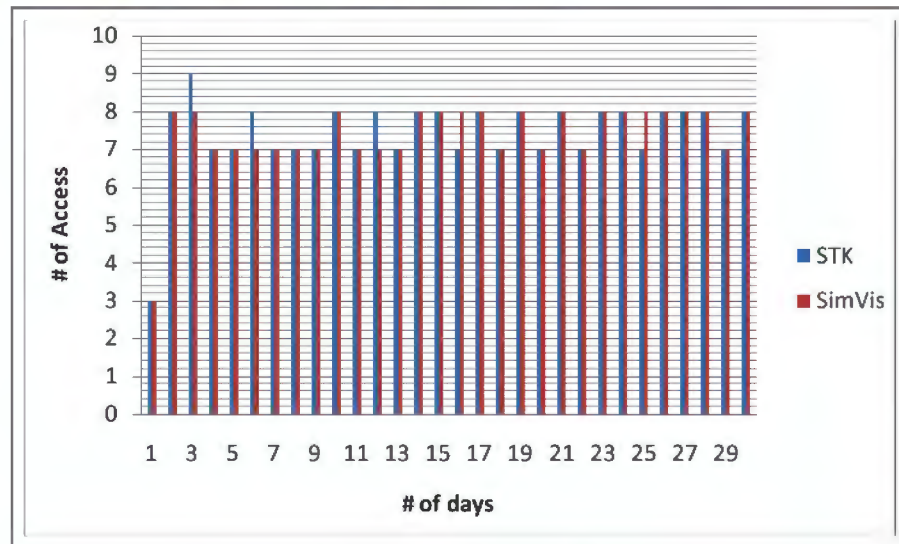


Fig. 7-16 Access Result Verification

The Below chart shows the distribution of the range with respect to time, the threshold level of the range is around 2400km, with 5° elevation and 1800Km with 10° elevation. The occurrence of 70-90° elevation (directly above the ground station), is of a maximum of once in 2 days. The disadvantage of low elevation angle is of space loss, since the range is too high, the propagation losses mounts, whereas at high elevation angles

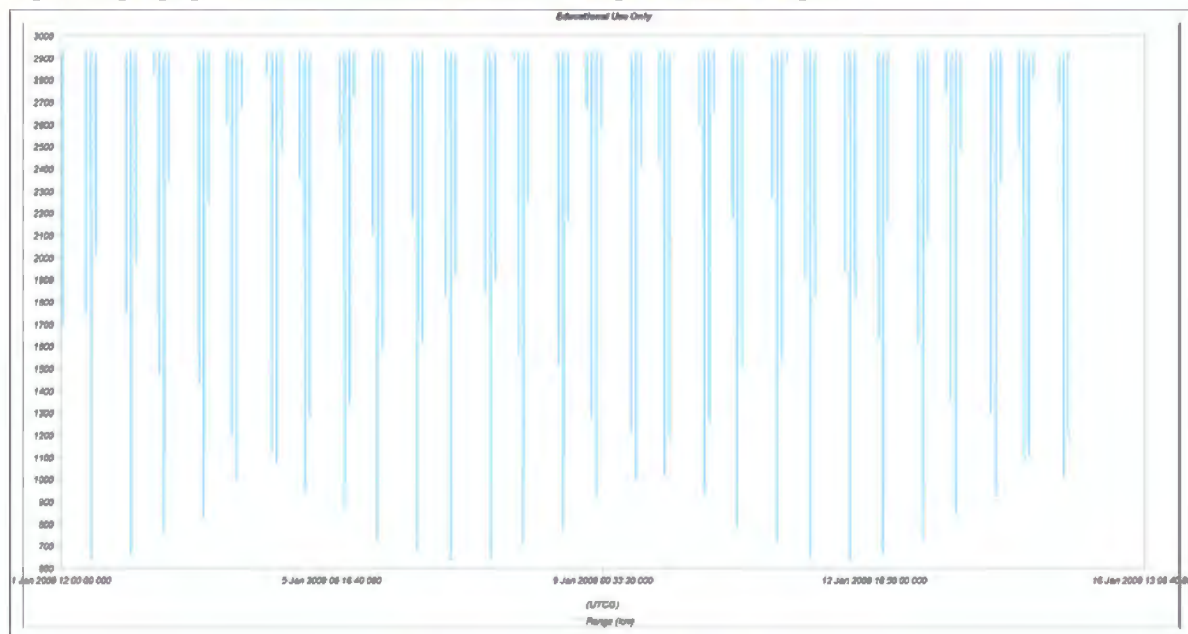


Fig. 7-17 Range plot from STK

The time of access during that period will be around 11 minutes, which is much viable for the transmission of payload data. The (Fig 4-16) illustrated the combination of elevation angle with respect to range and time. The purple colour indicates the elevation angle and the yellow colour indicates the range.

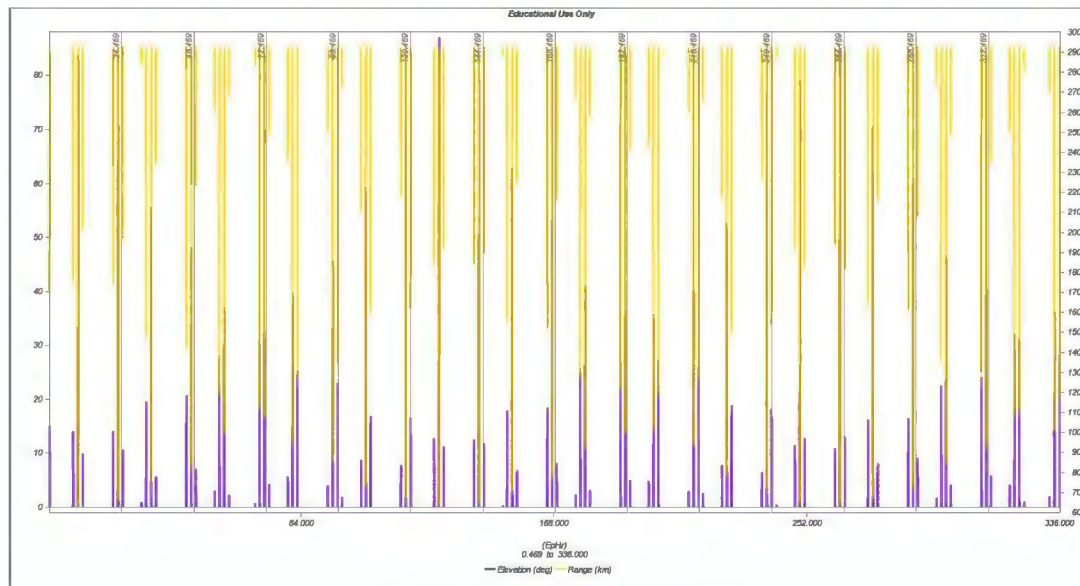


Fig. 7-18 Range vs. Elevation vs Time

The chart below shows the elevation angle with respect to the solar intensity. This graph shows that there are 3 possible transmissions in sunlight and 3 possible transmissions in Eclipse. We are not interested about the transmission in Eclipse since the power consumption is always a problem to think about.

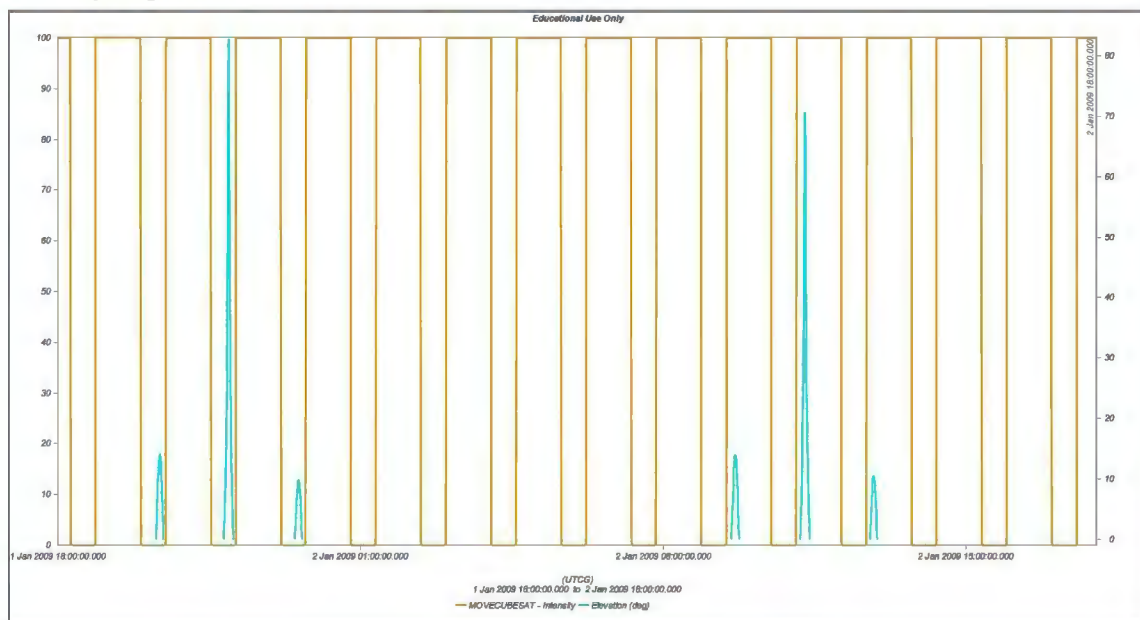


Fig. 7-19 Sun Intensity vs. Elevation vs time

The table below gives a comparison between the STK and SIMVIS in a Summarized form, which includes the antennas, camera and the sun sensor results. The figure below shows the representation of the monopole antenna beam. At this position, the satellite loses connection,

since the satellite transmitter is radiating in the opposite direction.

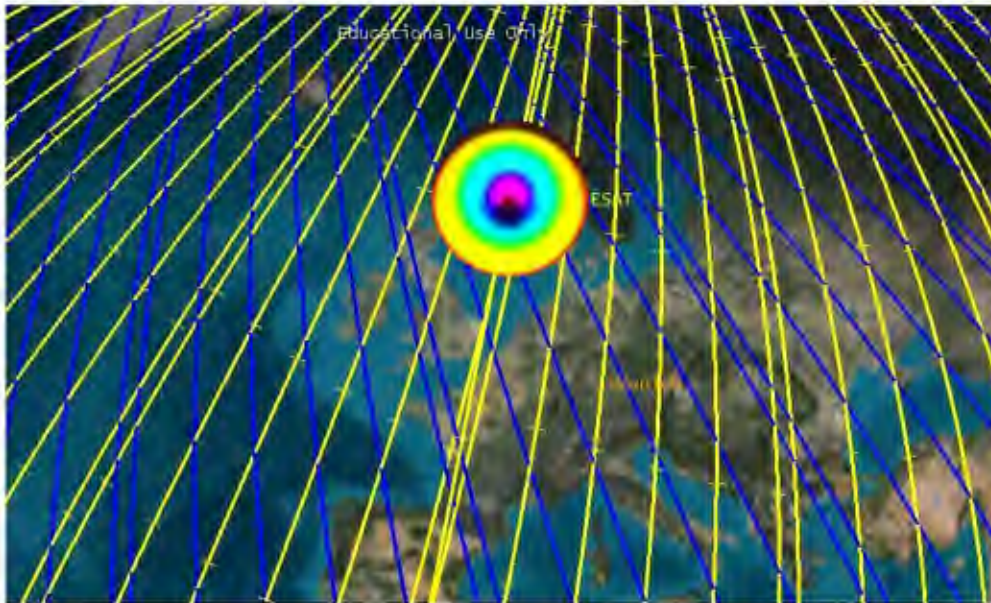


Fig. 7-20 Monopole Beam Representation in STK over the Munich Ground station

While the below figure give the representation of the monopole antenna beam, which is in contact with the ground station. The monopole antenna has a half donut shape beam. Every half revolution the satellite loses contact with the ground station. The colour scheme represents the intensity of the power radiated. The red region is of the maximum radiated power.

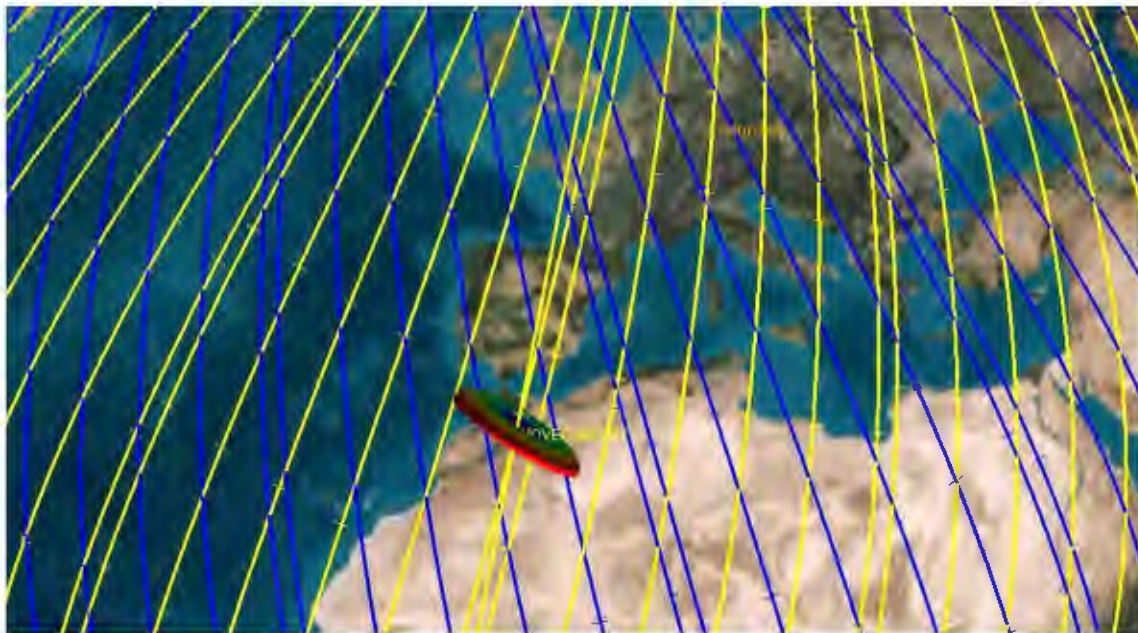


Fig. 7-21 Monopole antenna in contact with the Munich Ground station

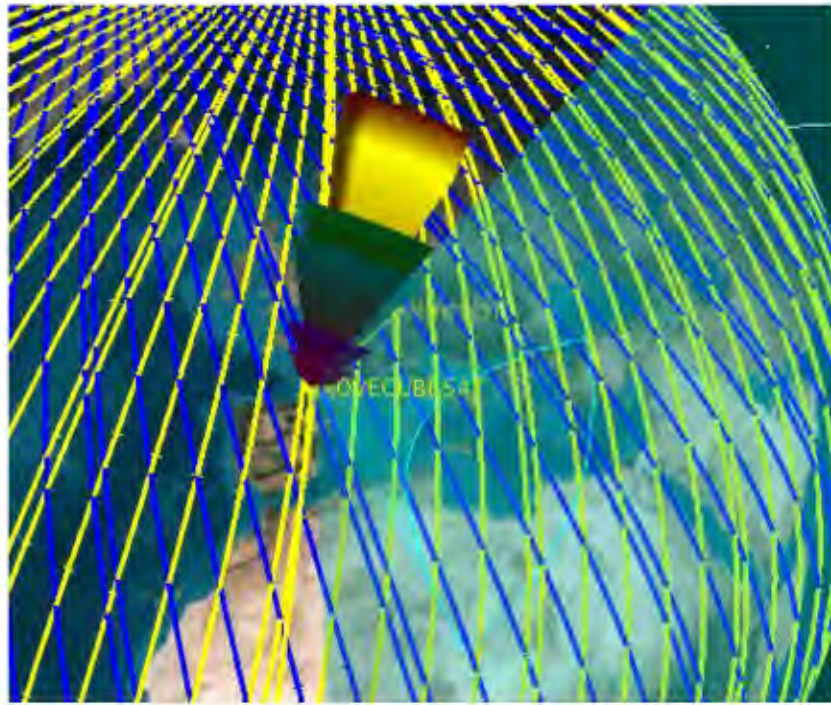


Fig. 7-22 Dipole receiver antenna beam in view with the Munich GS

The above figure shows the beam representation of the dipole antenna beam with respect to the ground station.

Tab. 7-3 Verified Results

Access	Antenna(min)		Camera (sec)	Sun Sensor(sec)			
	Dipole	Monopole		Sun Sensor1	Sun Sensor2	Sun Sensor3	Sun Sensor4
1	9	6.6	10	8	10	4	0
2	8.1	7.1	9	5	0	8	11
3	1.4	1	10	0	6	6	6
4	2.3	1.3	12	10	12	0	12
5	2.7	1.7	13	7	0	14	14
6	8.4	6.6	10	9	11	0	6
7	4.9	3.3	13	6	0	13	0
8	7.1	4.3	9	0	8	4	6
9	8.1	6.2	13	5	5	5	5

The above table gives a lean idea of the access time of the various elements considered for visualization. These results are compared and approximated from both the software approaches, since software's has a level of bug that reflects in the simulated outcomes. The camera visibility studies show that the average time in view will be 65 min/orbit. This gives sufficient time to take picture of earth both in sunlight as well during the eclipse.

7.7 Conclusion

In this chapter various analysis results are discussed based on the MOVE mission. Different software's were used to analyse and verify the ground tracking and access times while orbiting. This analysis gives the team engineers a plan to work on the mission phases and modes and switching between them. The antenna visualization results from SIMVIS shows, the number of access and the access time. This has helped in optimizing the orientation and also in verifying the same. The initial estimates show that, the average number of access per day is about 6 and an average access time of 7 minutes. From the dipole antenna point of view with tumbling effect, the STK results show that the average access time reduces by 1 minute. In case of monopole antenna with tumbling, the results show, that the average time in view is even worse compared to the dipole. There remains a slight difference in the results of the SIMVIS and STK. The SIMVIS results shows deviation with respect the STK results because, during the simulation the always a time slip remains between the earth revolution and the satellite motion, hence this leads to the time difference between the results. Based on the results the recommendation will be to have a multiple monopole antenna or the orientation has to be changed. This result has let to serious discussion on the monopole antenna and the orientation which are yet to be decided. The Camera studies show that an average of 65minutes/ orbit and with the sun sensors the visibility study shows average of 10seconds/access with the sun during the sunlight period. Even though detailed studies on the sun sensors are not discussed in this work. Based on these results, different dynamic scenarios were designed and assumed to use the MOVE (v)-Sys model for mission verification.

8 Dynamic aspects on (v)-Sys Model

The (v)-Sys uses an object oriented approach to carry out the static calculations. The calculations are based on relation between the elements, subsystem and system. For a satellite, the most important parameters to be dimensioned and verified are the mass, link, power and attitude determination. The major factors of concern to MOVE mission are the mass, link and power, whereas the attitude determination is still in trials. Even though there are lot of other institutes progressing to conduct CubeSat missions with attitude determination and controls, the MOVE CubeSat is of first of its kind with minimum provision for attitude determination and control. As a stepping stone, the sun sensors are being tested with future prospects in mind. In this chapter, the mass budget power budget and link budget will be of major concern for discussion using (v)-Sys and its present capabilities. Suggestions are also provided to improve the software to support the market needs in Space sector.

8.1 Mass budget

The elements that can be realized in a satellite have a physical property called mass. In (v) - sys, the elements are related to each other in terms of connectivity or operation. Once on choosing the elements and subsystems of the CubeSat from the data base, the parameters from the data base gets uploaded to the parameter of that element or subsystem. As mentioned earlier every element has a property mass. The mass gets summed up with respect to the subsystems. Example, the structural elements gets summed up under the *object* structure. In a similar fashion the other elements are grouped under their relevant satellite subsystem. The satellite does not carry any volatile mass like gas or liquid, hence the mass of the satellite does not vary dynamically. The dynamic aspects are neglected for the mass budget calculation.

Following the CubeSat specifications, the overall mass of a CubeSat is limited to 1kg. The structure of a CubeSat typically weights some 300 to 325g, and another 300 to 325g should be reserved for the payload, leaving only 400 to 450g for the energy collection, energy storage and the electronics itself. The commercially available CubeSat-Kit, containing the structure (190g) and electronic on-board-unit of a CubeSat ready to be used for a CubeSat mission, weights 310g including transceiver. 690g are left for the electrical system and the payload (not being part of the CubeSat-Kit).

Based on the calculations and estimations being made in this chapter, the overall mass budget of the CubeSat is outlined in the following table:

Tab. 8-1 Mass budget

SubSystem	Allocated Mass(g)	Current Mass(g)
Structure	330	275.5
Mechanism	50	46
Communication	50	124
Power	220	215
Command and Data Handling	230	220
Payload	100	110
Attitude Control	20	8
Total	1000	998.5

The Mass Budget is a summing up operation, defined in the excel sheet. The worksheet of the Satellite module defines various parameters from different subsystems.

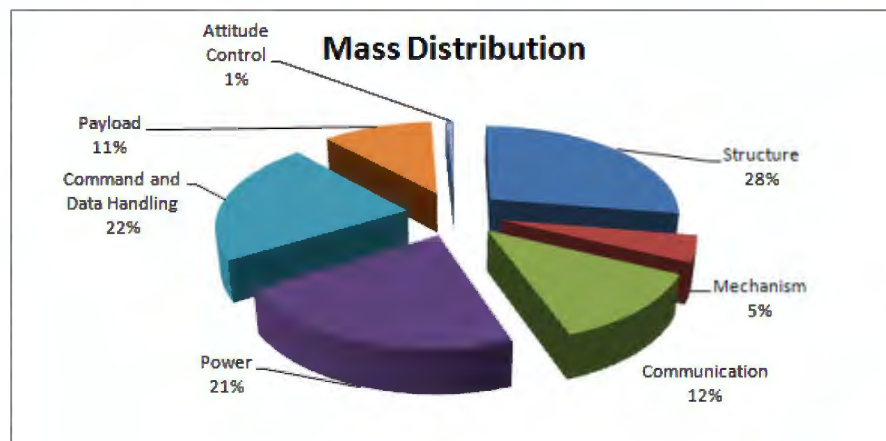


Fig. 8-1 Mass Distribution (998.5 gms)

The Satellite *object* sums up all the mass property values of its subsystem. A table is generated in (v) - Sys, which indicates the values of the each subsystem. The summing process is hierarchical. The mass value is changes based on the selected data sheet and the data entry of each component. The Mass variable is globally declared, such that the change in the subsystem automatically reflects on the Satellite module.

The initial mass estimates worked out to be 0.95 kg. Later after making changes with the structural design. The mass of the total system was raised up to 1.075kg. The Commercial CubeSat kit is built of aluminium structure covered in all six sides of the cube. In case of MOVE, to reduce further mass with the structural elements, two of its aluminium side plates (Front & Back) were replaced by carbon fibre. The other two sides were structurally optimized to reduce its mass. The reason for the increase in mass was due to the addition of a Transceiver capable to handle 2 antennas with 2 different frequencies.

8.2 Power Budget

Power consumption is a very important consideration when designing hardware for a CubeSat mission, because the available power is very limited. Other CubeSat missions and its power calculation define a total power consumption of less than 2.6 watts for the whole CubeSat, including on-board-computer and telemetry. It is obvious that only little power is left for scientific and technical payload. Power consumption of a CubeSat varies depending upon the orbit and operation scenarios. For power budgeting, the worst case scenario has to be assumed. This worst-case assumes a complete operation per orbit, with all subsystems and payloads being used at their respective allocated time. The highest amount of energy is needed for communications. For example, the ISIS radio used in MOVE consumes an electric power of 1.9 W while transmission.

8.2.1 Scenario

The power system Budgeting is based on the following assumptions,

- As show in the figure, the total number of transmission and reception are assumed for effective static analysis with dynamic aspects of the power system.

- A total time of 1 day was taken, of which the number of satellite contact with respect to the ground station varies between 3 to 6 entries, most often 5 entries can be expected.
- Since the satellite works on different mode proportions, a scenario has to be fixed and studied.
- The satellite transmission system needs at least 9 minutes to transmit the scientific and telemetry data at a rate of 1200bps.
- The satellite should send a beacon every 1 minute for 5 seconds at a rate of 300bps.
- The receiver should always be ON every 2 min for 10 sec throughout the orbit to verify that the satellite is not lost.
- The satellite will be in the sleep mode for the remaining part of the orbital time.

The scenario shown below gives the different modes for the time period of 24hrs. The orbital time is calculated to be 97.5 minutes; it makes approximately 14 orbits a day. As per the above assumption, the satellite will be in contact with the ground station 6 times a day, of which 3 will in the eclipse period and 3 in the daylight approximately. The link time varies between 3 min to 13min per access, on an average 9min/link. To handle the power system effectively, we need to use the communication link during the daytime rather than using in the eclipse as a power saving option. The blue shade in the figure is the link during the eclipse period and red shade is the link in the sunlight period. The above assumptions and the following assumptions are based on the visualization studies conducted and discussed in the chapter 7.

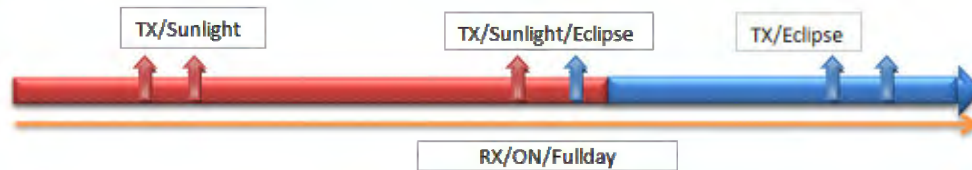


Fig. 8-2 Ground link Scenario

Tab. 8-2 Power Budget

S.No	Sub System	Bus Voltage	Modes				Units
			Beacon	Tx	Rx	Sleep	
1	COMM	7 V	1.9	1.9	0.2	0.2	(W)
2	EPS	5V	0.1	0.1	0.1	0.1	(W)
3	OBDH	5V	0.5	0.5	0.5	0.1	(W)
4	PL	5V	0.1	0.1	0.1	0	(W)
Total Power			2.5	2.5	1	0.4	(W)
Time			120	12	120	1188	(min)
Proportion			8	1	8	83	(%)

The power consumption during different modes is compared in the above table. Since the different modes are the key to define the working time schedules of the satellite. The maximum power is consumed during the Beacon and the Transmission mode. The transceiver consumes the maximum power of 1.9 Watts. But since the working of these modes are not continuous, this will not harm much. But as a safety factor the power generation system should be able to support the requirement. On contrast with other CubeSat mission the power

generation is almost similar, whereas small changes can make a huge difference with the power generation, that will be discussed in the following sections.

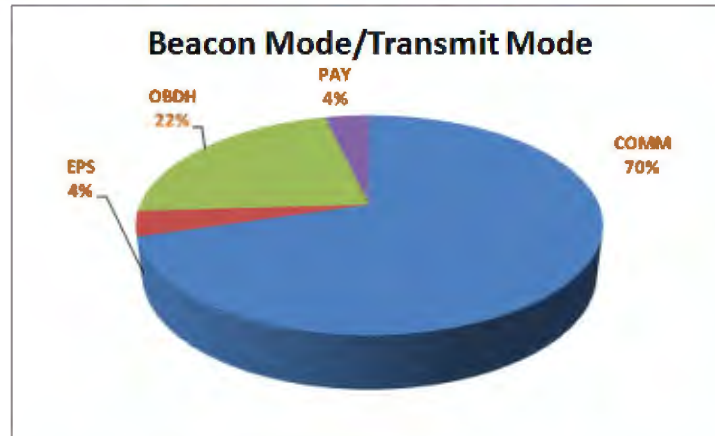


Fig. 8-3 Distribution of power consumption during Beacon/TX (2.5 W)

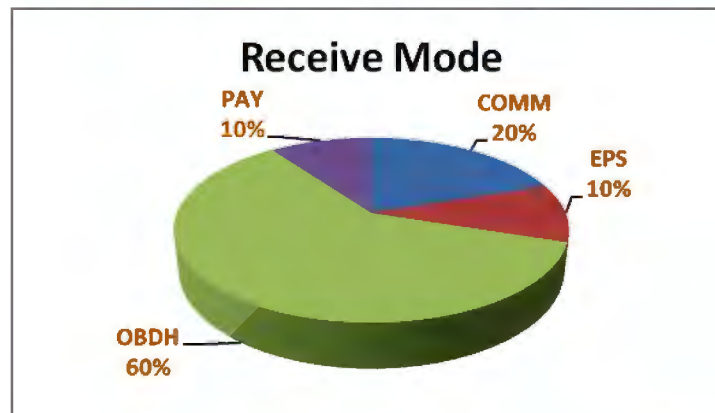


Fig. 8-4 Distribution of power consumption during Receive Mode (1 W)

The pie chart illustrates the distribution of the power consumption in during different modes and also the time distribution of the modes. The results show that most of the time the satellite will be in the sleep mode, since the power generation needs to be monitored to switch modes. Unless sufficient power is not stored, the satellite needs to be kept in the sleep mode.



Fig. 8-5 Distribution of power consumption during Receive (0.4W)

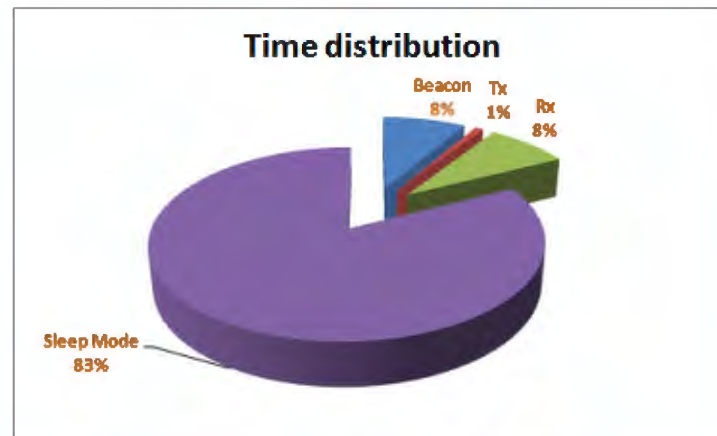


Fig. 8-6 Time distribution of satellite power modes (1440 min)

With the above analysis, the transmitter consumes the major portion; and the payload need to be considered, hence in the nominal mode of operation, there are 6 different possible power consumption states: 4 during daylight and 2 during the eclipse. In each part of the orbit the satellite can be either transmitting or not and taking pictures or not. The different states are listed in table below with their corresponding energy consumption.

Tab. 8-3 Scenario Vs Energy consumption

DwwP	Daylight WITH Transmission and WITH Payload	2.57Wh
DwoP	Daylight WITH Transmission and WITHOUT Payload	2.4Wh
DowP	Daylight WITHOUT Transmission and WITH Payload	0.51 Wh
DooP	Daylight WITHOUT Transmission and WITHOUT Payload	0.42Wh
EowP	Eclipse WITHOUT Transmission and WITH Payload	0.3Wh
EooP	Eclipse WITHOUT Transmission and WITHOUT Payload	0.25Wh

In order to have power modes for complete orbits, the different cases above have to be combined. The combination gives 8 (4x2) different power states. Table 8-4 shows the sum of the different states (**unit Wh**).

Tab. 8-4 Total energy consumption sunlight period Vs eclipse period

	EowP	EooP
DwwP	2.87	2.82
DwoP	2.7	2.65
DowP	0.81	0.76
DooP	0.72	0.67

Tab. 8-5 Predicted Scenario

Phase		No.of Orbit	Energy/Phase	Energy	Unit
DwwP	EowP		2.87	0	Wh
DwwP	EooP		2.82	0	Wh
DwoP	EowP		2.7	0	Wh
DwoP	EooP	2	2.65	5.3	Wh
DowP	EowP		0.81	0	Wh
DowP	EooP	1	0.76	0.76	Wh
DooP	EowP		0.72	0	Wh
DooP	EooP	11	0.67	8.04	Wh
Total		14		14.1	Wh

The total energy consumption for a day is calculated to be **14.1 Wh**. As per the scenario assumed, the transmission mode will be for 2 orbits and 1 orbit for using the camera and the remaining 11 orbits the satellite will be in sleep mode. Hence the total energy generated should be greater than the requirement.

8.3 Solar panel budgeting

8.3.1 Scenario

While calculating the solar power generation in (v)-Sys, different parameter control the power generation. Parameters like, sun vector, surface normal vector, number of cells facing the sun, incident angle etc. The following scenario gives the reader to get an understanding about the scenario assumed for the solar panel budgeting. The Scenario's were based on the visualization studies discussed in the previous chapter, the STK results and 3D visualization give an idea of the way in which the satellite will tumble in space. When calculating the power provided by the solar cells, the orientation of the CubeSat with respect to the sun has taken into account. Depending on the orbit, for a certain period of time the satellite is in eclipse and therefore not receiving any radiation power of the sun. For these calculations a 625 km sun-synchronous orbit is considered, with the satellite being in eclipse for less than 50 % of time. Because the satellite is not stabilized, a statistical analysis is performed; either one, two or three sides of the satellite are radiated by the sun, the efficiency (25-28 %) and size (0.00602m^2) per side of the satellite) of the solar cells. At first, the generated power per square meter is calculated, depending on the Solar Constant, efficiency and expected degeneration of the solar cells, and the average angle of the scenario. Using this generated power per square meter and the size of the solar cells of each side, the actual power for each scenario (one, two, or three sides of the satellite are radiated by the sun) is calculated. Finally the probability of each scenario and the average time in daylight are taken into account and the available power per orbit is calculated. This calculation, which is performed in the following table for the MOVE CubeSat radiated from one, two or three sides, clearly shows that while a peak power of nearly 4.3 watt will be generated by the solar cells at some time, the average power during a whole orbit period will be only about 2.2 W, therefore the available power is very limited. Please refer to the table itself for the exact numbers used in this calculation:

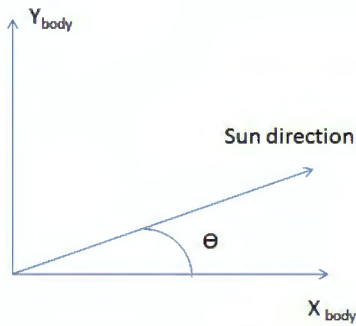


Fig. 8-7 Sun incident angle

Where Generated Power P per square meter is given by,

$$P = SC_{eff} * I_d * SC * \sin(\theta) \quad (8-1)$$

Total Generated power P_{total} is given by,

$$P_{total} = P * n * A \quad (8-2)$$

Available Power per orbit

$$P_{av} = P_{totalavg} * T_d \quad (8-3)$$

Case I:

The case I assume the spin along the X axis, the direction of the flight and limited spin along the other 2 axis. The average sun angle and the possible orientation are assumed based on the visualization studies.

Tab. 8-6 Case 1 with respect to satellite tumbling

Parameter	Scenario			
	0°	45°	60°	90°
Solar Cell Efficiency(SC_{eff})	0.25	0.25	0.25	0.25
Inherent Degeneration(I_d)	0.9	0.9	0.9	0.9
Solar Constant(SC , [W])	1367	1367	1367	1367
Average Sun Angle(a [degree])	0	45	60	90
Generated Power per Square Meter[W] P	0	217.488	266.368	307.575
Number of Exposed Sides(n)	0	3	3	3
Area per side of Satellite A	0.006	0.006	0.006	0.006
Generated Power[W] P_{total}	0	3.91479	4.79462	5.53635
Probability of Scenario(x)	0.1	0.35	0.30	0.25
Average power per Scenario(P_{avg})	0	0.9787	1.19865	1.38409
Total Average Power($P_{totalavg}$)	3.56144	W		
Time in Daylight(T_d)	0.63			
Available Power per Orbit [W]	2.24371			

Case II:

The MOVE orbit is divided into 3 zones, the maximum portion of the orbit is covered by Zone A and Zone B, C covers the remaining. In this case satellite is taken in along with the magnetic field. In the Zone A the satellite faces the sun with an incident angle of 90° according to the formula given above. The Zone A is split in to 2 a 90° zone and a 60° degree zone. The Zone B is assumed to have 45° and the Zone C is assumed to have 90° again, since it will flip at the poles. The other parameter that differentiates the scenario is the incident angles and the number of sides facing the sun along these zones.

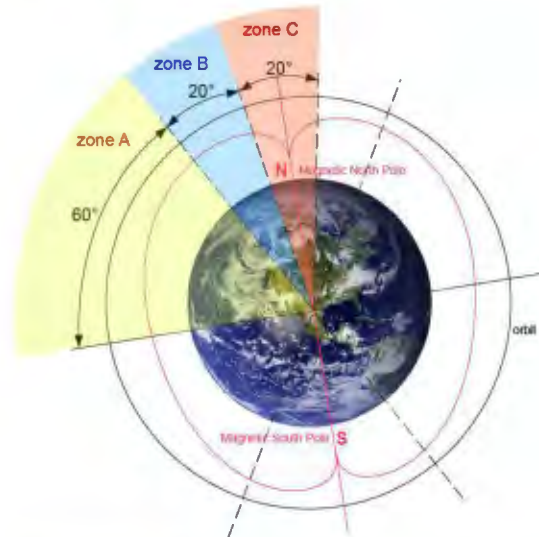


Fig. 8-8 Satellite Orientation Scenario (Assmann, 2008)

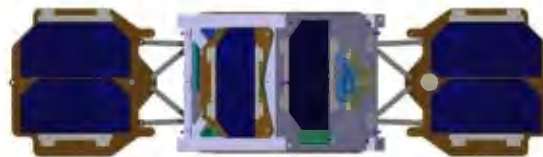


Fig. 8-9 Zone A Orientation 90°

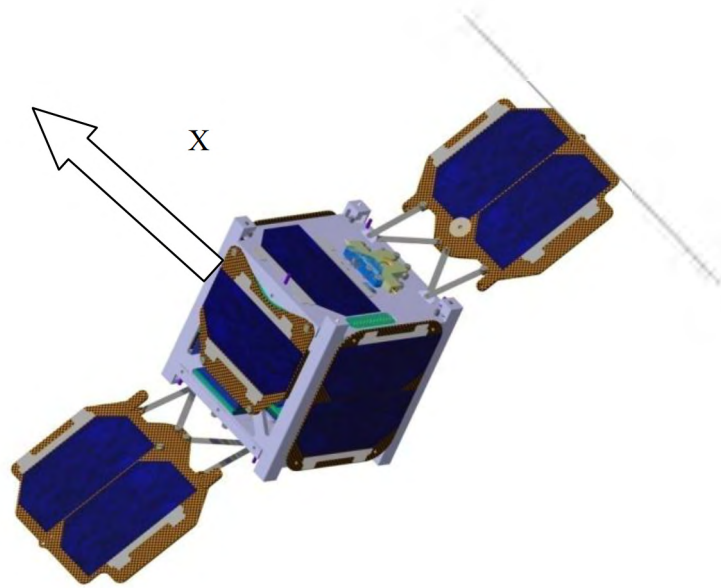


Fig. 8-10 Zone B 33° to 45°

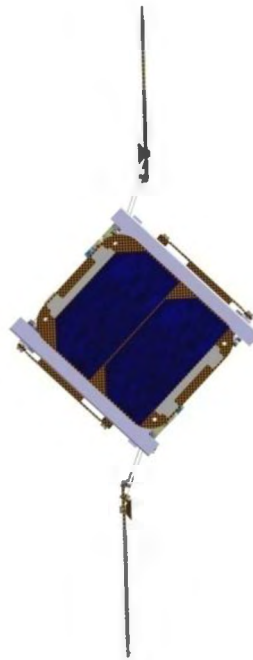
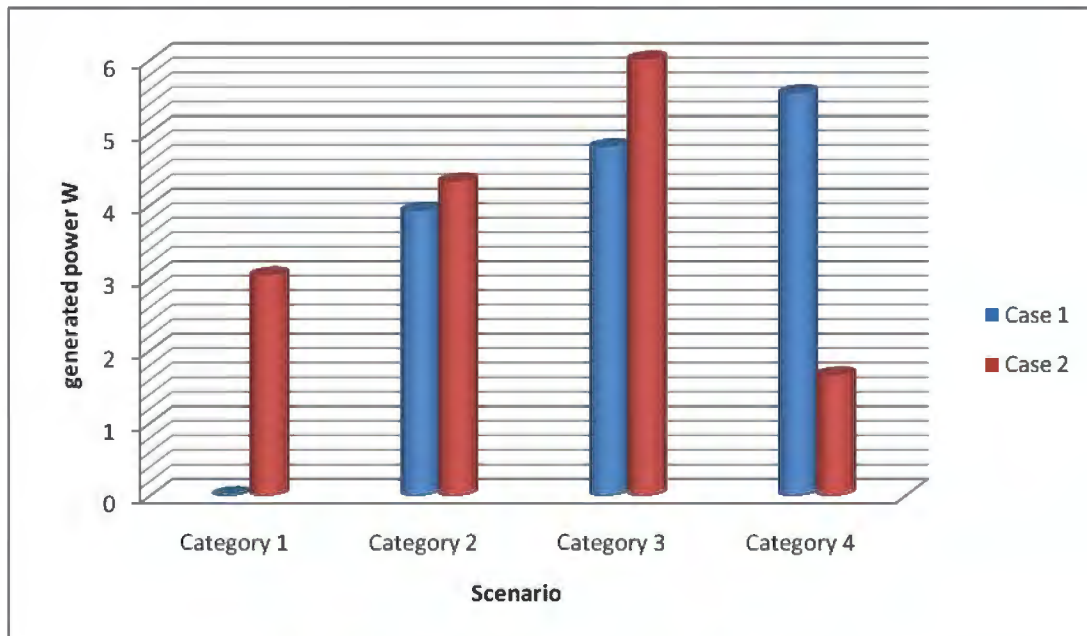


Fig. 8-11 Satellite flip near the poles/ Sun incident angle 90° Zone C Orientation

Tab. 8-7 Case 2 Scenario with respect to the orbit/magnetic field

Parameter	Scenario			
	One	Two	Three	Four
Solar Cell Efficiency(SC_{eff})	0.25	0.25	0.25	0,25
Inherent Degeneration(I_d)	0.9	0.9	0.9	0,9
Solar Constant(SC , [W])	1367	1367	1367	1367
Average Sun Angle(a [degree])	90	60	33	90
Generated Power per Square Meter[W] P	307.575	266.4	167.517351	307.575
Number of Exposed Sides(n)	2	3	3	1
Area per side of Satellite A	0.006	0.006	0.006	0.006
Generated Power[W] P_{total}	3.6909	4.795	3.015	1.845
Probability of Scenario(x)	0.6	0.2	0.15	0.05
Average power per Scenario(P_{avg})	2.214	0.959	0.452	0.092
Total Average Power($P_{totalavg}$)	3.718	W		
Time in Daylight(T_d)	0.63			
Available Power per Orbit [W]	2.34	W		


Fig. 8-12 Comparison between case 1 and case 2

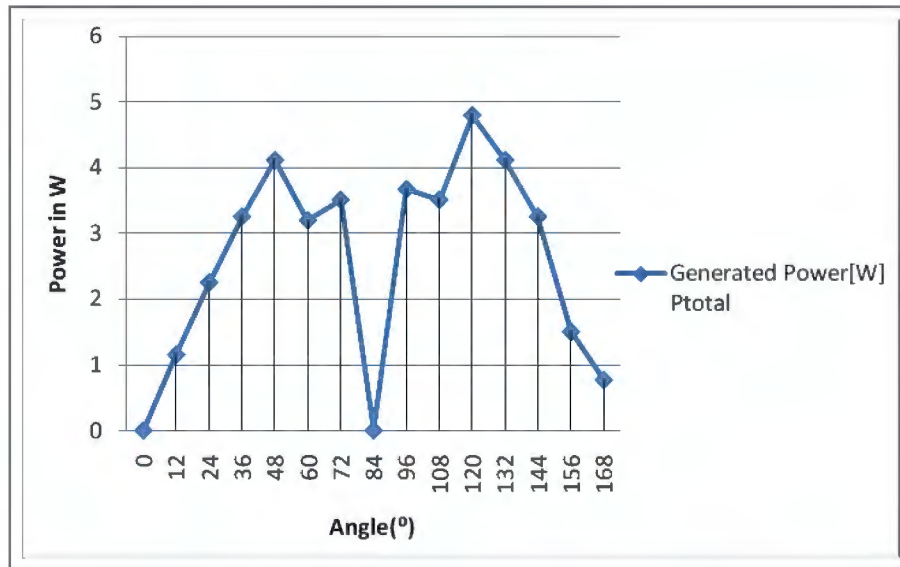


Fig. 8-13 Generated power profile in ½ revolution

As an interpretation from both the cases, the generated power profile shows that the average power generated will be 2.44 Watts in ½ revolutions about its spin axis. On manipulating the same with a probabilistic theory the mean generated power during the sunlight period will be **2.39W** approximately. On calculating the total energy generated in one full day will be **33.4 Wh**.

For the solar panel dimensioning, the (v)-Sys uses a step by step approach to design the system based on the requirements. Mission life and the average power requirements are the two key design considerations in sizing the solar panel. (v)-Sys sizes the photovoltaic system to meet the power requirements at EOL, with resulting solar array often oversized for power requirements at BOL. The longer the mission life, the larger the power requirements (Wertz J. , 1999). Since the MOVE mission is going to last for a minimum of 4 months, the power requirements are not much compared to larger space missions.

The inputs for the (v)-Sys to size the power system are the, average power requirements during daylight and eclipse. Orbit altitude, eclipse duration and the life time of the mission. The solar panel illumination intensity depends on orbital parameter such as the sun incident angles, eclipse period, solar distance and concentration of solar energy. If we mount the cells on the body of the spacecraft, we must orient them so they will generate adequate power throughout the mission (Peter Fortescue, 2002). But in the MOVE mission, since no solar tracking mechanism is used statistical orientation assumptions were made to select the exact layout design.

The output from the dimensional model gives the power that must be produced by the solar section, minimum number of solar cells, number of solar array sections and the maximum number of parallel solar cells in a section. Based on these results the product architectural model is compared and verified.

8.3.2 Solar Generator

In (v)-Sys power generator consists of the solar cell, solar section, Solar panel and the Battery charge regulator. A set of solar cells forms a solar section, a set of solar section forms a Solar panel. Since the MOVE Satellite uses only 16 cells inclusive of 4 test cells, in (v)-Sys only the solar cells and the solar sections are designed. The test cells are from Astrium/EADS ATJ, so the parameter values are chosen from the Excel datasheet under the product architecture. The tables show the design parameter values under the solar cell design.

Tab. 8-8: Solar generator input parameter

S.No	Input	Values
1	Short Circuit Current_Solar Array Section	0.5089 A
2	Number of Solar Array Sections	2
3	Open Circuit Voltage_Solar Array Section	5.4766 V
4	Specified Minimum Solar Generator Power Supply	3.8615 W
	Parameter	
5	Number of solar cell in series	4
6	Number of solar cell in parallel	1
7	Sun Incident angle	0°
8	Packing Factor	0.6
9	Solar Panel Thermal Absorptivity	0.85

As shown in the parameters table, the values were chosen from the datasheets of ATJ EADS cells. Since the solar cells are mounted in all the directions, the maximum possible combination will be 4 cells in view of which 4 in series and 1 in parallel with one section and the other section will have 4 cells connected in series. The minimum possible solar cells in view will be 2 cells, 2 in series. Since the satellite is passively controlled, the magnets are positioned in such a way that the best case is in view with the sunlight (4 Cells).

Tab. 8-9: Solar generator output

S.No	Calculation of	Results
1	Total Number of Solar Cells	4
2	Solar Cell Area per Panel	0.00608 m ²
3	Total Solar Panel Area (Substrate Area)	0.018 m ²
4	Amount of Solar Power absorbed by the Solar Panel	21.080 W
5	Power at MPP - Solar Panel	4.75 W

The solar panel calculation give the amount of solar power absorbed by the solar panel is 21.08W and power generated during MPP is 4.75 W, where we require 3.8 W as power at MPP.

On applying the probabilistic approach on the (v)-sys parameters keeping the specification of the solar cells to be constant, the values attained are similar to that of the scenarios assumed. The difference in the values of the assumed scenario and the calculation in (v)-Sys are illustrated in the below graph.

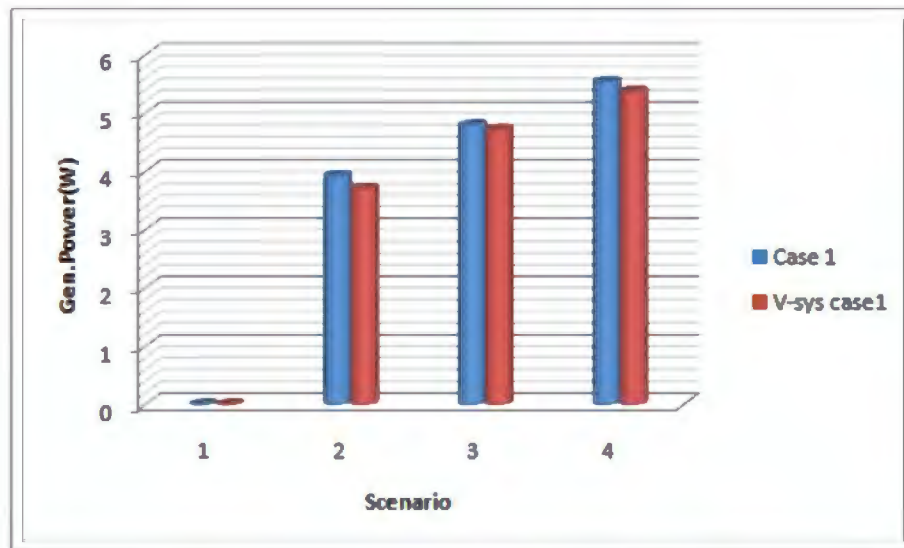


Fig. 8-14 Case1 and (v) - Sys Results comparison

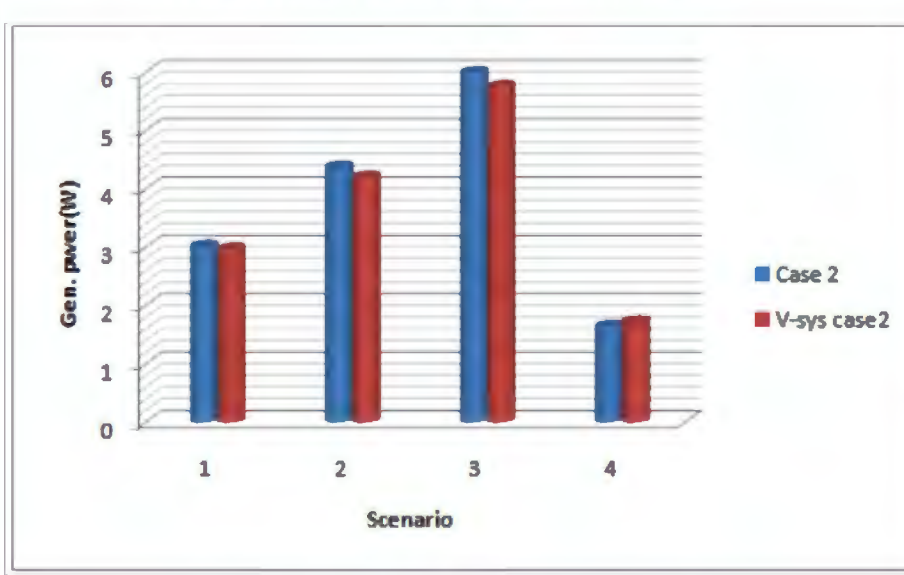


Fig. 8-15 Case2 and (v)-Sys results comparison

Even though there are slight differences between the scenario and the (v)-Sys model, on an average the values remain the same. (v)-Sys model is even more precise compared to scenario, since the values of the solar cell and its characteristics are adopted in (v)-Sys, whereas in the scenario, it is more a generalized methodology.

On applying the probabilistic approach on the above analysis in (v)-Sys, the average power that is generated is about **2.29 W**.

As per the above analysis, the **effective area** of the solar cell array would be 3 sides with 6 cells facing the sun at 90° . This is not possible in the current MOVE mission. Currently a maximum of 2 sides with 4 cells faces the sun with the area of 0.0121m^2 , whereas when adding the payload cells to the benefit, then the average power generated will be **3.03 W with an area of 0.0181m^2** . Which is much better compared to the current scenario. In the future MOVE mission, on adding these two sides will make a dramatic change in the power shortage problems of MOVE CubeSat, may be all the other CubeSat missions too.

8.4 Battery budgeting

The energy collected by the solar cells is stored inside batteries in order to provide power even when no radiation from the sun is received by the satellite (e.g. when the satellites operates at the 'Night-side' of the Earth). A number of different battery types exist these days, being based on different chemistries and having different properties. In the following table a comparison of battery chemistries and their advantages and drawbacks is given:

Tab. 8-10: Comparison on different battery types

Characteristics	NiCd	NiMH	LiIon	LiMetel
Nominal Voltage[V]	1,2	1,25	3,6	3
Gravimetric Energy Density [Wh/Kg]	45	55	100+	140+
Volumetric Energy Density [Wh/l]	150	180	225+	300+
Self discharge Range[% Month]	25	20 to 25	8	1 to 2
Temperature Range[°C]	0 to +50	-10 to 50	-10 to 50	-10 to 50

Lithium-Ion batteries have high energy densities and high specific energies. In addition to this, they are relatively easy to produce in prismatic rather than cylindrical packs and have a higher per cell voltage than NiMH and NiCd batteries. On the downside they are more expensive, require a specialized charging circuitry, and have a slightly reduced cycle life. The reduced cycle life is acceptable for a CubeSat mission, because the satellite initially has a shorter mission life (about 2 years).

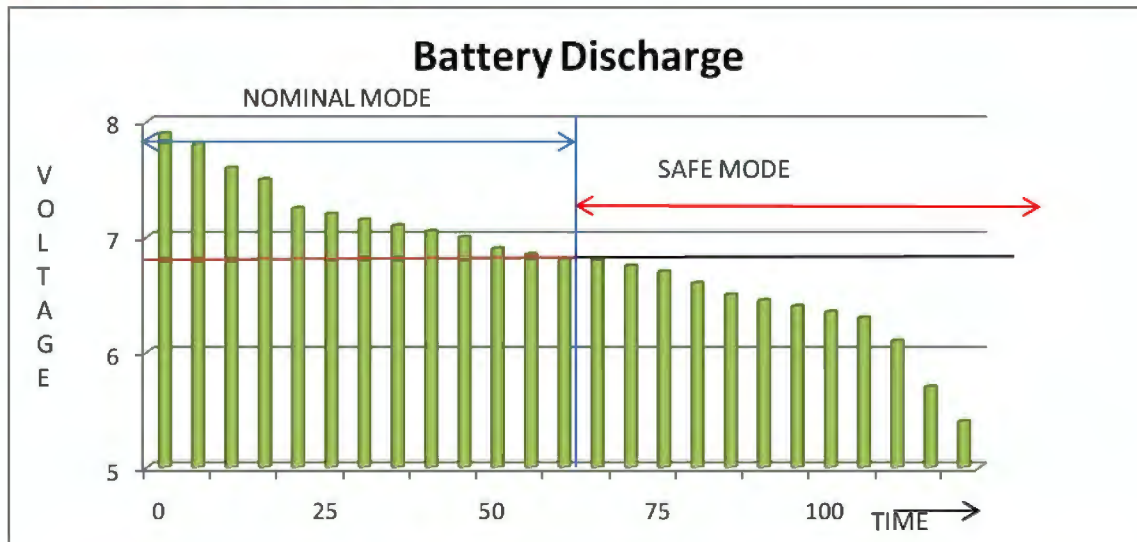


Fig. 8-16 General Threshold voltage characteristics with respect to time

A battery consists of individual cells connected in series. The number of cells required is determined by the bus voltage. The lithium ion batteries come along with Clyde space Electrical power system.

8.4.1 Battery Cell Calculation

The following table shows the inputs and parameter for verifying the product element with the design dimension. The inputs and the parameters are based on the chosen data sheet with respect to the dimensional requirements.

Tab. 8-11: Input parameters for Battery cell product VARTA

S.No	Input	Values
1	Nominal Voltage	3,7 V
	Parameter	
2	Total Electric Capacity BOL	5,8 Ah
3	Mission Duration	4 Months
4	Degradation of Capacity of Battery per Year	1%/Yr
5	Total Number of Battery cells	4

The battery cell capacity at EOL is used as an input to calculate the specified usable cell energy in the dimension model. The battery cell Energy at EOL is 74942 J which is far much greater than the requirement.

Tab. 8-12: Calculation results

S.No	Calculation of	Results
1	Battery Cell Capacity at End-of-Life	5.684 Ah
2	Battery Cell Energy at End of Life	74942.496 J

8.4.2 Battery Package

The battery package used for the move mission uses a Lithium Ion battery (VARTA). The mass of each cell weighs about 12 gms with a cell voltage of 3.7V. The battery package comes along with the EPS Module, of which the package consists of 2 batteries connected in parallel, where each battery is packed with 2 cells in series.

Tab. 8-13: Battery package Input parameters

S.No	Input	Values
1	Total Electric Capacity EOL	5,684 Ah
2	Specified Minimum Useable Battery Capacity	0.1074 Ah
3	Mass Battery Cell	12 gm
	Parameter	
4	Depth of Discharge allowed	30%
5	Cell voltage	3.7 V
6	Number of Battery cells in Series	2
7	Number of Battery cells parallel	2

The calculation results show that the battery package can provide a constant 7.4 V. With a total end of life capacity of 11.4 Ah and a usable capacity of 3.4 Ah. The Total energy and the usable energy are also comparatively high with respect to the dimensional model.

Tab. 8-14: battery package Results

S.No	Calculation of	Results
1	Battery Voltage	7.4 V
2	Total Capacity at End-of-Life	11.368 Ah
3	Total Usable Capacity at End-of-Life	3.4104 Ah
4	Total Energy at EOL	302843.52 J
5	Total Usable Energy at EOL	90853.056 J
6	Total Amount of Battery Cells	4
7	Battery Mass	48 gm

The inputs required to dimension the energy storage system in (v)-Sys are the mission length, Eclipse length, eclipse frequency, bus voltage, Total electric capacity and allowed depth of discharge. The outcome of the battery dimensioning model are the specified number of battery cells in series and parallel, specified minimum usable energy, specified minimum total capacity and Specified minimum unusable capacity.

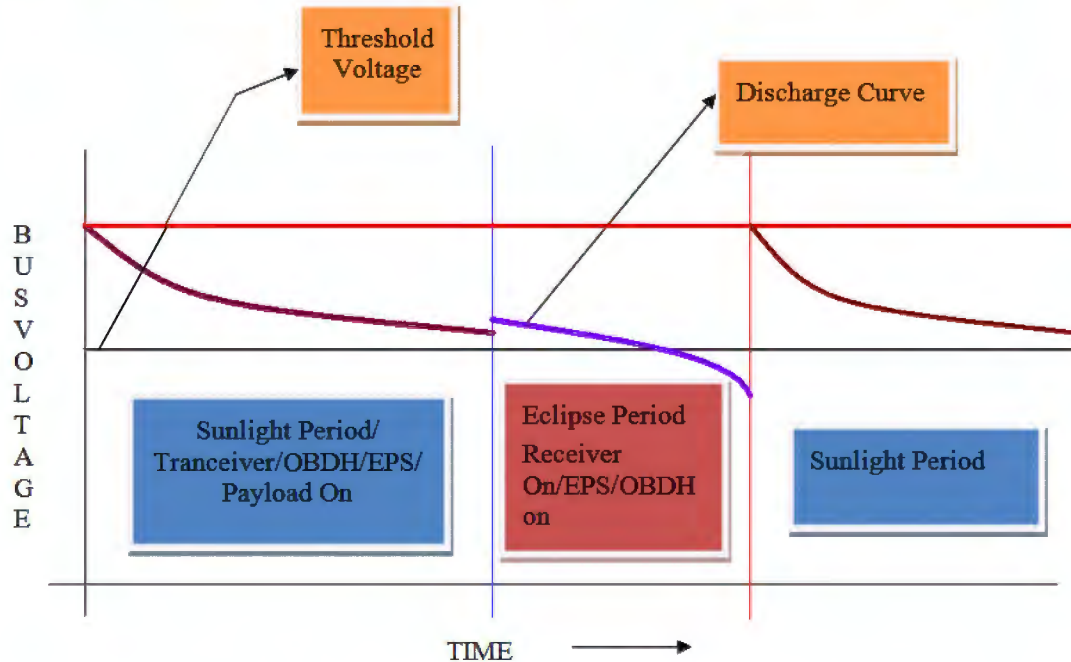


Fig. 8-17: General Discharge/ charge characteristics

The above figure shows the general characteristic curve of the battery discharge and battery charge of the satellite missions. Currently the (v)-Sys does not have the capability to conduct dynamic modelling. The dynamic modelling of the power system will be dealt under the Open Sim Kit.

8.5 Link Budget

8.5.1 Ground Station antenna design verification

Tab. 8-15: Helical Antenna Design Verification

Helical Antenna design verification		
INPUT		
Frequency(f)	4E+08	Hz
Number of turns(n)	9	N
Pitch angle(α)	15	°
OUTPUTS		
Wavelength	0,6897	M
Diameter of Base plate(D)	0,7241	M
circumference of the antenna coil(s)	0,6897	M
Area of the base plate(A)	0,4118	m ²
diameter of the antenna(d)	0,212	M
Spacing between turn(b)	0,1785	m
Distance between base plate and antenna coil(a)	0,1034	m
distance between base plate and start of coil(a')	0,1379	m
Length of antenna(L)	1,7099	m
Gain of the antenna(G)	13,48	dB

The below given set of formulas were used to verify the ground station antenna design,
The wavelength is given the general equation,

$$\lambda = \frac{c}{f} \quad (8-4)$$

The diameter of the base plate should be between factors of 0.75 to 1.33.

$$0.75 < \lambda < 1.33 \quad (8-5)$$

The circumference length of each turn is give by,

$$s = \lambda \quad (8-6)$$

The diameter of the antenna is given by,

$$d = \frac{s \cdot \cos \alpha}{\pi} \quad (8-7)$$

The spacing between each turn is given by,

$$b = s \cdot \sin \alpha \quad (8-8)$$

The starting distance between the base plate and the antenna coil is given by,

$$a = 0.15 \cdot \lambda \quad (8-9)$$

The distance between the base plate and the start of coil is given by,

$$a' = 0.2 * \lambda \quad (8-10)$$

The length of the antenna is given by,

$$L = a + n.b \quad (8-11)$$

The Gain of the antenna is given by,

$$G = 10.\log (8.5 * L + 7.75) \quad (8-12)$$

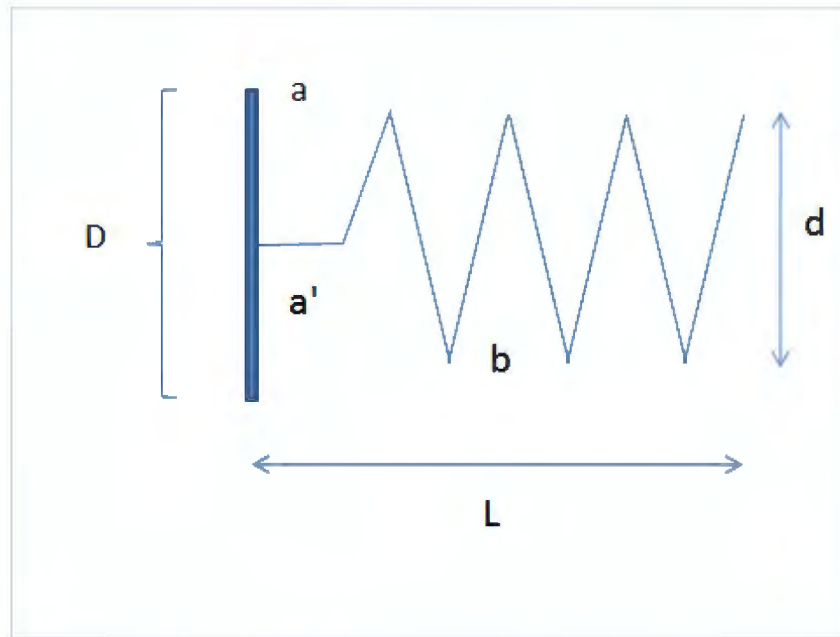


Fig. 8-18 Helical antenna Layout (Harder, 2007)

8.5.2 Scenario

For the Link Budget, different scenarios were considered. Since the Ground station parameters are fixed, this does not require a change in design. Hence the concentration is given on the Spacecraft link design. The antennas used for the communication are a monopole and a dipole antenna as discussed in the previous section. The gain of the antenna and the directivity of the antenna is never a constant. There are 2 cases which lead to the instability, one is due to the tumbling of the spacecraft and the other is due to the uneven directivity. Even though the dipole antenna is of an isotropic kind, but reality it fact is not so. Hence a different scenario of changing the antenna gain has been in place for the analysis part. An antenna gain of varying from -10dB to 2 dB has been assumed. Most of the dipole and monopole antenna experiments show an antenna gain of 2dB. But to augment the outcome of the analysis a gain of -10 dB was assumed.

8.5.2.1 Downlink 146 MHz

The following assumptions were made:

- For the slant range a worst case orbital altitude of 2891 km was assumed with a minimum elevation of 5° .
- A transmit power of 0.2 Watts was assumed with an antenna gain of 2dBi on the satellite side.

- The downlink antenna being a Monopole there is a theoretical zero in the axis of the antenna.
- For the antenna pointing loss, 3dB were assumed.
- The satellite's antennas being linearly polarized a circular polarization was assumed on ground, the worst case loss is 3dB.
- The chosen modulation scheme is BPSK: For a bit error rate of 10^{-5} and E_b/N_0 of 10.33 dB is required.
- Maximum data rate is 1200 bit/sec.
- A stack with Yagi antennas for downlink was assumed at the ground station yielding a conservative gain of 12dBi.

Tab. 8-16: Downlink Budget

Downlink Link Margin			
S.No	Parameter	linear Value	dB
1	satellite altitude Km	625	27,96
2	Radial Distance Km	7003,34	38,45
3	Bit Rate bps	1200	30,79
4	Bit Error Rate	1,00E-04	-40,00
5	Modulation Method	FSK	
6	Min Elevation angle deg	10	10,00
7	Link Frequency Hz	146000000	81,64
8	Ground station antenna gain(dB)	31,6227766	15,00
23	GS Noise figure(dB)	3,16227766	5,00
10	GS Noise Temperature K	627,060521	27,97
11	Boltzmann constant	1,38E-23	-228,60
	C/N req		
12	bandwidth(bps)	2400	33,80
13	E_b/N	3,55E-03	-24,50
14	C/N req	4,26E+00	6,29
	C/N act		
15	Max Distance (m)	2891980,29	64,61
16	Wave length(m)	2,05479452	3,13
17	Max free space loss	3,20E-15	-144,95
18	S/C Transmit power P_t (W)	0,20	-6,99
19	S/C antenna gain G_t (dB)	1,00E-01	-10,00
20	S/C EIRP	2,00E-02	-16,99
21	GS Antenna Gain G_r (dB)	1,00E+01	10,00
22	GS received power P_r	6,39E-16	-151,94
24	G/T	0,01594742	-17,97
25	E_b/N_0	10	17,48

23	C/N act		14,88
24	Link Margin		8,59

Tab. 8-17 Transmitter Gain vs. Receiver Gain

Downlink		
Gt	Gr	Link Margin
-10	15	13,59
-10	10	8,59
-10	8	6,59
-10	5	3,59
-10	2	0,59
-10	0	-1,41

The downlink link margin shows a minimum margin of 11dB to a maximum of 23.6dB, with a varying antenna gain. As the satellite tumbles at certain rate, the dynamically varying patterns. The transmitter gain varies based on the orientation of the antenna with respect to the ground station. The above calculations are dealt with the worst case scenario rather than the best case. If the transmitter antenna is horizontal to the earth surface, a maximum gain can be expected with 2dB, whereas the worst case goes in to negative gain. On assuming -10dB the link margin still remains well inside the limits of 3dB margin.

8.5.2.2 Uplink 435MHz

The same assumptions were made for the Uplink with the following exceptions:

- Transmitted power: 50W.
- Double Helical stack: 15 dBi.
- On the satellite side, a dipole with a gain of 2.2 dBi was assumed.

Table summarizes the generated TM/TC budget. For a 625 km the path losses decrease by about 5 dB. The downlink budget show close to 35 dB margin, but further refinements need to be done in this area.

Tab. 8-18 Uplink Budget

Uplink Link Margin			
S.No	Parameter	linear Value	dB
1	satellite altitude Km	625	27,96
2	Radial Distance Km	7003,34	38,45
3	Bit Rate bps	1200	30,79
4	Bit Error Rate	1,00E-04	-40,00
5	Modulation Method	BPSK	
6	Min Elevation angle deg	10	10,00
7	Link Frequency Hz	435000000	86,38
8	Ground station antenna gain(dB)	31,6227766	15,00
23	SC Noise figure(dB)	3,16227766	5,00

10	SC Noise Temperature K	627,060521	27,97
11	Boltzmann constant	1,38E-23	-228,60
	C/N req		
12	bandwidth(bps)	1440	31,58
13	Eb/N	5,91E-03	-22,28
14	C/N req	7,10E+00	8,51
	C/N act		
15	Max Distance (m)	2891980,29	64,61
16	Wave length(m)	0,68965517	-1,61
17	Max free space loss	3,60E-16	-154,44
18	GS Transmit power Pt(W)	2,00E+01	13,01
19	GS antenna gain Gt(dB)	1,00E+01	10,00
20	S/C EIRP	2,00E+02	23,01
21	S/C Antenna Gain Gr(dB)	1,00E-01	-10,00
22	S/C received power Pr	7,20E-15	-141,43
24	G/T	0,00015947	-37,97
25	Eb/No	0,1	20,74
23	C/N act		27,62
24	Link Margin		19,11

Tab. 8-19 Uplink Transmitter Gain vs. Receiver Gain

Uplink		
Gt	Gr	Link Margin
15	-10	24,11
10	-10	19,11
8	-10	17,11
5	-10	14,11
2	-10	11,11
0	-10	9,11

The Link Margin calculation show that, even on dynamically varying the antenna gains the Link margin is far above the 3dB Margin.

8.5.3 Transceiver module

The product design module consists of the Transceiver module, Downlink, Uplink and the ground-station module. While the Ground station and its facilities are available, the model remains the same. The transceiver *object* is the same for both GS and S/C.

Tab. 8-20: ISIS Transceiver Parameters

S.No	Input	Values
1	Antenna Frequency downlink	146 MHz
2	Uplink Frequency	435 MHz
3	Downlink Data rate	1200 bits/sec
4	Antenna Gain Transmitter	0 dB
5	Antenna Gain receiver	0 dB
6	Transmit Power	0.2 W
7	Uplink Data rate	1200bits/sec
	Parameter	
8	Type of Modulation for Downlink	BPSK
9	Type of Modulation for Uplink	FSK

The ISIS transceiver was chosen based on the dimensional model requirements. The input parameters shown in the above table are based on the Datasheets available. The antennas used are dipole antenna for receiving and a monopole antenna for transmitting, as discussed in the previous chapter.

The calculation results show that possible bandwidth with the transceiver system, the antenna length to be used, transmitter EIRP and Actual Carrier to Noise ratio.

Tab. 8-21: Transceiver results

S.No	Calculation of	Results
1	Downlink Bandwidth (BPSK)	1.44 Kbps
2	Uplink Bandwidth (FSK)	2.4 Kbps
3	Transmitter EIRP	0.2 W
4	Receiver Power	2.15 W
5	Energy bit to Noise ratio	2.04E-13
6	Bit Error Rate	10^{-5}
7	Actual Carrier to Noise ration	24.88

8.5.4 Uplink model

The uplink model calculates the signal wavelength used for transmission, expected efficiency of transmission due to space Loss and the Link Margin. The link margin will decide the link budget achievement.

Tab. 8-22: Uplink model

S.No	Calculation of	Results
1	Transmission Signal wavelength	0.663 m
2	Space loss	3.60E-16
3	EIRP Uplink	7.20E-14
4	Link Margin	29.11



The calculation results show the uplink margin is calculated to be 37dB, when reducing the transmit power to 20W the link margin went down to 19dB.

8.5.5 Downlink Model

Tab. 8-23: Downlink Model

S.No	Calculation of	Results
1	Transmission Signal wavelength	2.05 m
2	Space loss	3.20E-15
3	EIRP Downlink	6.39E-17
4	Link Margin	18.59

The link budget shown under link scenario has been represented or translated in terms of (v)-Sys and the results from the scenario and (v)-Sys remains the same since the mathematical model used for the scenario in the previous section and the model in (v)-Sys are the same. Hence the values were verified based on the different antenna gain and vice versa.

The Link Budget results from (v)-Sys shows, the link margin to be well over the limits of the 3dB margin. Since the availability and choice of the COTS component for the transmission system is less, based on the cost and reliability factor the ISIS module was chosen. With the outcomes of the current mission the team has future plans to build a transmission module for upcoming CubeSat missions.

8.6 Conclusion

The chapter gives an introduction to the (v) - Sys modelling software and the various scenarios assumed for the MOVE satellite and its subsystem through the dynamic aspects assumed through the SIMVIS and STK results. Various scenario calculations are shown for each and every subsystem. The mass budget, power budget, Link budget are well illustrated in this chapter. The next chapter will deal with the results from the (v) - Sys model and a comparison with the assumed scenario calculations. The power system budgets show that the maximum power consumed will be 2.7 Watts at the Beacon and the Transmission phases. The tentative mass budget shows the total mass of the satellite is calculated to be 998.5gms this may increase with a margin 5% works to 1045gms inclusive of the deployable panels. The best case results of the link budget shows a downlink margin of 23dB with 2dB S/C antenna gain and an uplink margin of 39.75dB with 2dB S/C antenna gain. The worst case scenario show the link margin with downlink margin of 11dB for -10 dB antenna gain and an uplink margin of 27 dB. The inertial calculation and the solar torque calculation are of not much effect in static calculations. A mathematical model and transformations will be discussed in the chapter 10.



9 (v)Sys Future Prospects

The functionalities developed under (v)-Sys for satellite modelling is not complete. Hence the (v)-Sys modelling does not have all the functionalities to test the satellite model. This chapter deals with the future prospects of (v)-Sys and its satellite modelling abilities.

9.1 Data budget

The word telemetry means “measurement from a distance.” Strictly speaking, all signals arriving from a satellite constitute telemetry, whether they originate in the satellite payload or in the platform. The word telemetry usually refers specifically to the platform. To distinguish the platform data from the payload, they are generally called as housekeeping telemetry and scientific telemetry.

Tab. 9-1: Input/ Output Budgets

OBDH Input/output Budgets						
Components	# of Components	Power Line	Operational Hour per day	Intervals (min)	Averaging	Bits per sample
Scientific Data						
Camera	1	1	24	60	NO	
solar cell Voltage	4	0	24	5	Yes	10
Solar cell Current	4	0	24	5	Yes	10
Sun Sensor data	4	4	24	5	Yes	10
House Keeping Data						
Temperature sensor	8	8	24	4	Yes	10
Battery Voltage	2	0	24	3	Yes	10
Solar Panel Voltage	6	0	24	3	Yes	10
Control Power						
Antenna Release Control	1	1	-	-	-	-
Power Distribution Control Line	1	1	24	3	NO	10
COMM						
Transceiver	1	1	Varies	-	-	-
Data Bus	1	0	24	5	-	10

The above table gives an idea about the electronic systems and components used and its physical parameter. It also gives an idea about the measurement cycles and the bits per sample.

In (v) - Sys, the telemetry and scientific data budget are to be dimensioned and the memory storage device has to be selected based on the dimension. Already, in (v)-Sys the electronic elements has a parameter telemetry data rate and scientific data rate. It is summing operation under the satellite object model. With these parameters the total telemetry data rate and scientific data rate can be calculated for the whole satellite.

Tab. 9-2: Telemetry packet budget

OBDH Data generated		
Variables	Bits/min	Description
Time	32	System time at the start of the packet
Solar Cell Voltage	10	The voltage of the solar cell Bus
Battery 1 Voltage	10	The voltage of the first battery
Battery 2 Voltage	10	The voltage of the second battery
7V Bus Voltage	10	The voltage of the BCR DC- DC convertor
Solar Cell Temperature	60	Temperature of the thermister
Battery 1 Temperature	10	Value from the EPS battery 1
Battery 2 Temperature	10	Value from the EPS battery 2
Transceiver Temperature	10	ISIS transceiver temperature
Test Solar cell temperature	10	Payload solar cell temperature
Solar cell Voltage/Current	20	Payload solar cell voltage/current
Sun Sensor Data	40	Sun Sensor values
OBDH Temperature	10	OBDH board temperature
Total	242	Bits/min (Excluding Camera)
	30.25	Bytes/min

(v)-Sys should have the option to change the data rates at different modes. With this value, the user will have option to dimension the memory unit based on these results. A mathematical function has to be implemented to integrate the memory accumulation with respect to time. With the known number of orbits per day, we can calculate the total memory accumulated in a day. With this value we can dimension the memory unit. Later the compatibility of the memory unit has to be checked with the OBDH board.

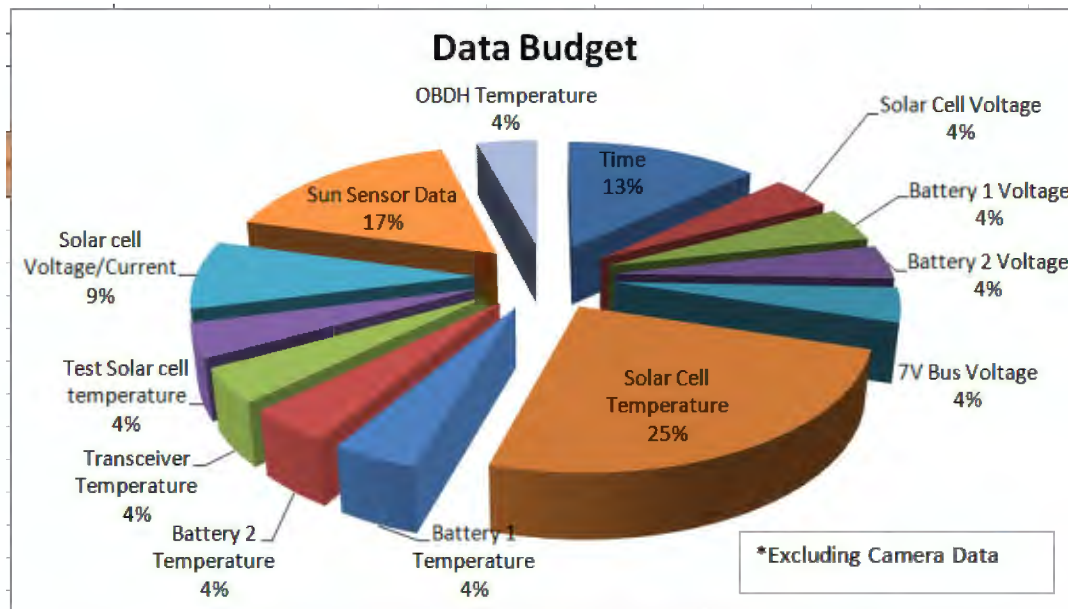


Fig. 9-1 Telemetry Data Budget (30.25 bytes)

The above table and the figure give an idea of the data generated from each element and its distribution. This excludes the data generated from the camera. Since the camera contributes a major portion of the data generated, it will be explained in the coming section.

9.1.1 Camera design and data compression

The camera to be used for the MOVE mission is still under consideration. Even though it is decided to carry a camera, the product is yet to be decided along with the data rates and compression. The actual sensing device will be chosen considering the demands around image size, pixel- and radiometric resolution which all have an impact on the chip, camera and lenses used. Based on the expected orbital height, calculations concerning image sizes and focal length will be performed in this chapter as well.

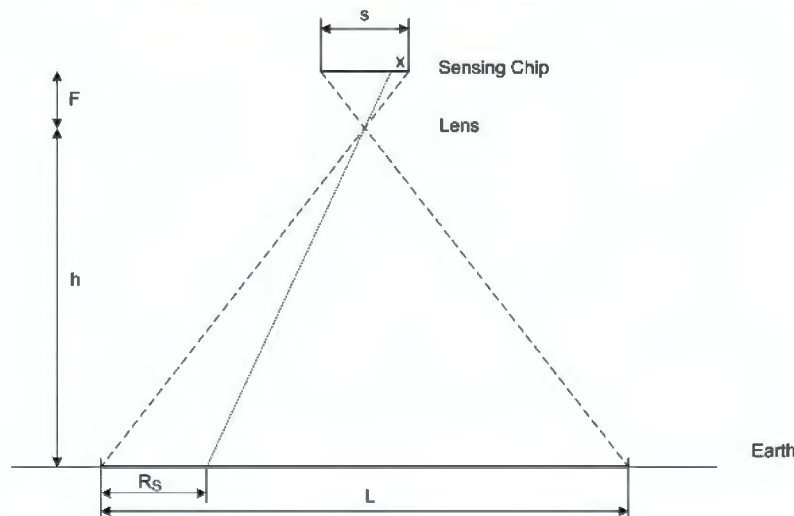


Fig. 9-2: Focal length design layout



The focal length is a very important parameter for the design of the remote sensing payload, because it is the major parameter specifying the size of the experiment. Figure 2.19 shows in a schematic point of view how the focal length is influenced by the satellite's height, the size of the chip and the field-of-view. In addition to this the spatial resolution can also be derived from this picture. Please note that the curvature of the earth is not included in these calculations.

Based on the desired field of view (L), the satellite's orbit (h) and the size of the chip (s), the focal length (F) can be calculated using the following formula derived from figure 2.19:

$$F = \frac{s * h}{L} \quad (9-1)$$

For a proposed orbit of 625 km, a chip size of 8 * 8mm and a desired field-of-view of 200 * 200km, we can calculate the following focal length:

$$F[mm] = \frac{8mm * 625 * 10^6 mm}{200 * 10^6 mm} = 24.8mm \quad (9-2)$$

Given an expected payload size of 5 * 5 * 5 cm and a chip size of only a few millimetres, a sensing device with this focal length can be easily integrated into the MOVE CubeSat.

The spatial resolution depends on the field-of-view (L) and the number of picture elements (x) of the scanning device. Typical CCD devices feature some 1.3 million picture elements (pixels), which are about x = 1200 pixels per line, resulting in a spatial resolution of

$$R_s = \frac{L}{x} = \frac{200 * 10^3 m}{1200} = 167m \quad (9-3)$$

This spatial resolution should be sufficient for the needs of this CubeSat mission. By changing certain parameters it can be calculated how the field-of-view, spatial resolution, focal length, orbital height and number of picture elements relate to each other. If a bigger field-of-view is requested, the focal length gets smaller, but also the spatial resolution. If the given spatial resolution has to be kept, the number of pixels inside the chip has to be increased. On the other hand, if the orbital height is changed, the focal length has to be changed proportionally in order to get the same resolution.

Scenario

Recent developments in digital camera design have resulted in high resolution CCD and CMOS cameras providing picture of 1.3 and more million pixel. Although such a pixel is taken in a fraction of a second, it takes a reasonable time to transmit these pictures with the narrow power and bandwidth transmitter being part of a CubeSat. At a typical 625 km orbit, the satellite is orbiting the earth once every 97 minutes. From a given ground station, the satellite however is only visible for about 10 minutes, the other time it is below the horizon and data cannot be received. Considering a data rate of 1200 bit per second (bps) and a time-window of 600 seconds, a total of **1200bps * 600s = 7, 20,000bits** or 90kByte can be transferred when passing by the ground station. On the other hand, a single colour picture with a spatial resolution of 1.3 mega-pixel and a spectral resolution of 1.3 * 8bit would already need some **1.3 * 1024 * 1024 * 3 * 8 = 4.09MBytes**, thus only less than 20 % of the picture could be transmitted during on pass-by.

In order to fully transmit a picture within 10 minutes, the raw data needs to be encoded and compressed. One very common algorithm for this task is the JPEG (Joint Pictures Experts

Group) algorithm, which is based on inverse cosine-transformation and provides variable compression rates of up to 20: 1 (and more). The higher the desired compression ratio, the more information is removed from the picture to be transmitted, therefore the quality is decreased. Thus, the JPEG-algorithm is considered to be a “data-reduction” algorithm, because actual information is removed from the picture in order to decrease size.

It was calculated above that the maximum size of a picture to be transmitted, must not be bigger than 90 Kbytes. If high-resolution, 1.33 mega-pixel pictures are desired, a high compression must be performed on them, or, the resolution must be decreased. If the data rate is increased to 19200 bps, a total of 1406 Kbytes of data could be transferred, allowing the use of lower compression rates for a given picture size. In the table below the necessary compression rates in order to transmit a picture using 1200 bits per second, 9600 bits per second and 19200 bits per second within 10 minutes are listed:

Tab. 9-3: Camera Data compression analysis

Resolution	Raw Data(Kbytes)	Compression Rate required		
		1200	9600	19200
1	3072	34.95	4.37	2.18
1.3	3993.6	45.44	5.68	2.84
2	6144	69.91	8.74	4.37
2.4	7372.8	83.89	10.49	5.24
3	9216	104.86	13.11	6.55
5	15360	174.76	21.85	10.92
	Picture size(Kbytes)	87.9	703	1409.17

The data rate was chosen to be 1200bps, since the ISIS Transceiver uses 1200 bps. The ISIS has the capability to use 9600bps, but it need to be tested and as per the component manufacturer the 1200bps is recommended.

9.1.2 Data Budget Analysis

The data budget analysis in (v) - Sys is a simple summation operation from different subsystem. The data sheets chosen for the payload sensor and the camera data are entered as per the estimated shown in the previous section, the total house-keeping data rate is estimated to be 30.25 bytes of telemetry information and the scientific data rate from the payload is calculated in (v)- Sys as show below;

Tab. 9-4: Payload Data Budget

Payload	# of components	Data	Unit
Camera	1	85	Kbytes
Temperature sensor	8	0.010	Kbytes
Sun Sensor	4	0.05	Kbytes
Total	13	85.015	Kbytes

The transmission of camera data needs a separate link, since it generates 85Kbytes. Hence the other payload data on calculating for the whole orbit is estimated to be 2.93Kbytes with a measurement interval of 1 min. With the data rate of 1200bps, it takes a maximum of 20 seconds to transmit the data.

$$R_{Tx} = \frac{(R_{sc} + R_{Tm}) \epsilon_{orbit}}{\epsilon_{link}} \quad (9-4)$$

The above formula works to the continuous generation of the data for an orbit. Using the maximum data rate specified in the transceiver data sheet, it takes 9.7 min to transfer the payload data or the scientific data. The telemetry data is calculated to 2.9 Kbytes and the system software data storage of 512 Kbytes.

Tab. 9-5: Total Data generated per orbit

Data generator	Data generated /orbit (Kbytes)
Scientific data capacity	85.015
Telemetry data capacity	2.93
Miscellaneous	5
Total	92.94 Kbytes

The **minimum volatile memory** required after calculating the total data generated/orbit to total data generated /day will be **1.30 Mbytes/day**. The major portion of the data is the camera data. With 2 possible transmission attempts/ day will have 20 minutes of effective transmission. With a data rate of 1200bps, it will take **140 minutes** of transmission time to transfer 1.30 Mbytes of data Approximately 14 transmission orbits/day, which is not possible. Hence the data generated per day should be reduced to support the mission.

Tab. 9-6: Number of Pictures vs. Time of transmission

# of pictures/ day	Data transmission time(min)
14	132.24
10	94.46
5	47.23
2	18.89

On working out the same with the number of pictures taken/day, the table above shows, the optimum number of pictures per day will be 2, with 19 minutes to transfer (well under the limits). The result supports the power and communication system scenario in the previous chapter.

The camera data rate is calculated to be 116 bps and the telemetry data rate to be 4 bps. Using the formula (9-4), the **transmission data rate** is calculated to be **1164 bps**, which is rounded as 1200bps. This supports the link calculation in the previous chapter.

Whereas, the volatile memory required to store the raw data from the Camera will be **3Mbytes/picture**. Taking in to consideration, the

The uncompressed data are stored for a very short period of time, hence that will not disturb the assumptions. Any value above the requirement is advisable. The table below lists the memory size with respect to the number of orbit data it can store.

Tab. 9-7: Memory size vs. Total number of orbit data storage

Memory size(Mbytes)	# of orbit (information)
5	19.23
10	38.46
25	96.15
50	192.30
100	384.615
150	576.923
200	769.230

The minimum volatile memory capacity can be chooses based on the worst case conditions. A Memory space of **5 Mbytes** will hold data of 20 orbits of compressed payload data and telemetry data. Based on the COTS components available in the market, the memory storage unit can be chosen.

Hence a minimum volatile memory of **7 Mbytes** is required for the MOVE mission (without compressing the Camera data). As the memory storage units are of less mass and size, the satellite can carry even a 1GB memory stick comfortably. The mission does not require so much of memory space as well the camera operation will not be so frequent since the power consumption has to be taken in to consideration. The values above will the worst case scenario.

9.2 Thermal Budget

The thermal budgeting will give an idea on choosing the electronics components and adding thermal system to the satellite based on the results obtained. This can be used as a thermal dimensioning model for satellite modelling (v) Sys.

This section provided a brief estimation of the thermal gradients during the first 8 orbits. The following table shows a set of parameters used for the initial estimation of the thermal gradient. There are other sophisticated thermal simulation software's used for the MOVE mission. This thesis work doesn't carry detailed analysis on the Thermal budgets. The basic calculations are presented below.

Tab. 9-8 Thermal Budget input parameters

S.No.	Parameters	Values	Unit
1	Altitude	625	Km
2	Mass	1.005	Kg
3	Heat Generated	2.7	W
	Dimension of the Cube	.1*.1*.1	m ³
4	Absorptivity of Solar cell	0.85	
5	Emmisivity of Solar cell	0.9	

6	Absorptivity of Aluminium Structure	0.2	
7	Emmisivity of Aluminium Structure	0.05	
8	Sunlight Period	3716.8565	s
9	Eclipse Period	2131.831	s
10	Solar constant	1353	
11	Stefen Boltzmann constant	5.67 E-8	W/m ² K ⁴
12	Absorbtivity Area	0.0173205	m ²
13	Emissivity Area	0.06	m ²
14	Proportions of Solar cell area	60.3	%
15	Proportions of Aluminium Surface	29.7	%
16	Equivalent Temperature T_{eq} in sunlight	303.73303	K
17	Equivalent Temperature T_{eq} in Eclipse	194.23899	K
18	Constant k sunlight	4136.3754	s
19	Constant k eclipse	15815.64	s
20	Starting Temperature	283.5	K

A spacecraft in earth orbit is subjected to heat generating radiation from three different sources: Direct solar radiation primarily in the visible spectrum; solar radiation reflected by the Earth's surface in the visible spectrum; and thermal radiation from the earth in the infrared spectrum. The albedo radiations originate from the daytime side of the earth and depend on the orbital height as well as on the degree of light diffusion occurring over land oceans and the atmosphere.

The equilibrium temperature during the sunlight is given by the below equation. Where α is the average absorbtivity constant, A_a and A_e is the area of heat absorption and area of heat emitted, q_s is the internal heat generated and S is the solar constant. σ is the stefen Boltzmann Constant. (Berlin, 2005)

$$T_{equ} = \left(\frac{\alpha A_a S + q_s}{\epsilon A_e \sigma} \right)^{\frac{1}{4}} \quad (9-5)$$

The below equation is used to calculate the Equilibrium temperature at Eclipse. Since MOVE mission is a Low earth orbit mission, the earth's Albedo needs to be taken in consideration. Hence the solar constant is given by $S_a = 0.25S$. (Berlin, 2005).

$$T_{equ} = \left(\frac{\alpha A_a S_a + q_s}{\epsilon A_e \sigma} \right)^{\frac{1}{4}} \quad (9-6)$$

The radiative heating can be given by the below Equation, where k is the time constant and C is the integration constant and the t is the current cycle or orbit time. T is the initial temperature.

$$\frac{T}{T_{equ}} = 1 - e^{-0.4(\frac{t}{k} + C)} \quad (9-7)$$



The time constant k is given by the thermal gradients equation, where c is the specific heat of the element m is the mass of the satellite.

$$k = \frac{mc}{4\varepsilon A_{\varepsilon} T_{equ}^3 \sigma} \quad (9-8)$$

$$C = 2.5 \ln \left(\frac{T_{equ}}{T_{equ} - T_o} \right) \quad (9-9)$$

The below equation is for radiative cooling,

$$\frac{T}{T_{equ}} = 1 + 1.3 e^{-2(\frac{t}{k} + C)} \quad (9-10)$$

$$C = 0.1312 + 0.5 \ln \left(\frac{T_{equ}}{T_o - T_{eq}} \right) \quad (9-11)$$

Tab. 9-9 Thermal Iteration table

TIME	Orbit	Cycle		unit	Constant C		Temperature in Deg	Unit
2131.831	1	Cooling To Eclipse	262.40485	K	C Eclipse	0.51996226	-10.59514931	°C
5848.6875	1	Heating To Sunlight	280.25749	K	C sunlight	4.98651159	7.257493128	°C
7980.5185	2	Cooling T1 Eclipse	225.59273	K	C Eclipse	0.53846343	-47.40727435	°C
11697.375	2	Heating T1 Sunlight	278.52066	K	C sunlight	3.39410795	5.520664154	°C
15414.232	3	Cooling T2 Eclipse	206.23879	K	C Eclipse	0.54866242	-66.76121483	°C
17546.063	3	Heating T2 Sunlight	285.86459	K	C sunlight	2.84088951	12.86458571	°C
21262.919	4	Cooling T3 Eclipse	200.46557	K	C Eclipse	0.50688931	-72.5344275	°C
23394.75	4	Heating T3 Sunlight	292.98224	K	C sunlight	2.69706714	19.98223595	°C
27111.607	5	Cooling T4 Eclipse	197.44181	K	C Eclipse	0.46948314	-75.55818983	°C
29243.438	5	Heating T4 Sunlight	297.44747	K	C sunlight	2.62491617	24.44747261	°C
32960.294	6	Cooling T5 Eclipse	195.83683	K	C Eclipse	0.44736912	-77.1631722	°C
35092.125	6	Heating T5 Sunlight	300.10875	K	C sunlight	2.58744869	27.10874783	°C
38808.982	7	Cooling T6 Eclipse	195.0213	K	C Eclipse	0.43463983	-77.97869583	°C
40940.813	7	Heating T6 Sunlight	301.65878	K	C sunlight	2.56862372	28.65877648	°C
44657.669	8	Cooling T7 Eclipse	194.61786	K	C Eclipse	0.42737245	-78.38214438	°C
46789.5	8	Heating T7 Sunlight	302.55042	K	C sunlight	2.55936295	29.55042289	°C

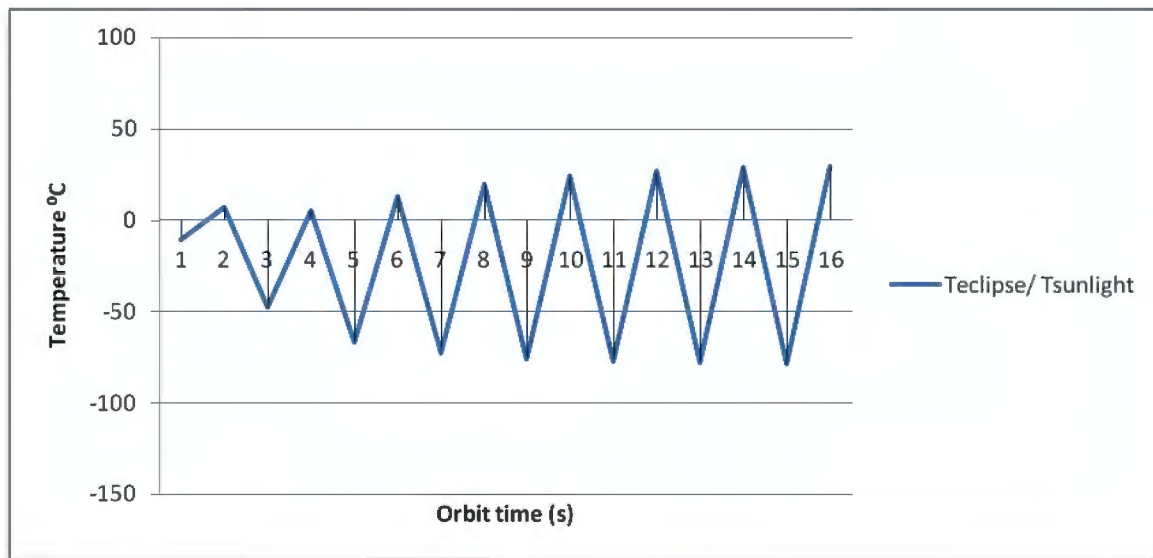


Fig. 9-3 Thermal Analysis

The above output shows, the temperature gradient is tending towards negative temperature. The active Heating elements inside the MOVE satellite are not taken into consideration. The critical components in the MOVE satellite are well protected to withstand the thermal gradients. The most important component to be taken into consideration is the battery, but the battery is well protected by internal heaters called the kapton heater and controlled actively.

9.3 Conclusion

The data budget shows a Telemetry data packet of 27Kbytes/day and the payload data of 90Kbytes. The data management budgets shows that the minimum memory required to store the data generated per day is calculated to be 260 Kbytes, taking raw payload camera data the minimum memory required to will be 5 Mbytes. Since the memory devices are of low mass and small dimension, it does not have any physical dimensions. A different scenario for choosing the volatile memory has been discussed in this chapter. The thermal calculation show the satellite cools down heavily as it progresses, but since the satellite uses internal heating elements and convertors to heat up the satellite and maintain in the equilibrium margin. An array of thermal sensors will be used to monitor the thermal deviation, the temperature sensor is selected based on the expected temperature variations, and there are COTS components for temperature measurement available directly in the market for using in space missions. A detailed thermal analysis done by another MOVE team member in software called Thermal Desktop shows, the values are well under control. The following chapter will deal with the dynamic analysis of the antenna, payload and ground tracking.

10 Development of Open SimKit

10.1 Introduction

OpenSimKit is a dynamic simulation software, developed as an outcome to upgrade the (v)-Sys static modelling software. Even though, they are completely a different kind of software, the basic concept of software structure remains the same. The (v) - Sys is built as common software for developed for any system engineering application, Whereas OpenSimKit is dedicatedly created for Space Systems Engineering. The basic architecture of the software remains the same as (v)-Sys. Few of the functionalities used in (v)-Sys are reused in OpenSimKit. In this part of the thesis work, Based on the experience of the simulation software used in previous sections for MOVE mission, will be taken as an input to future create this software OpenSimKit. The following sections explain the reference coordinate system, attitude control dynamic equation and various external torque equations used for the attitude control modelling.

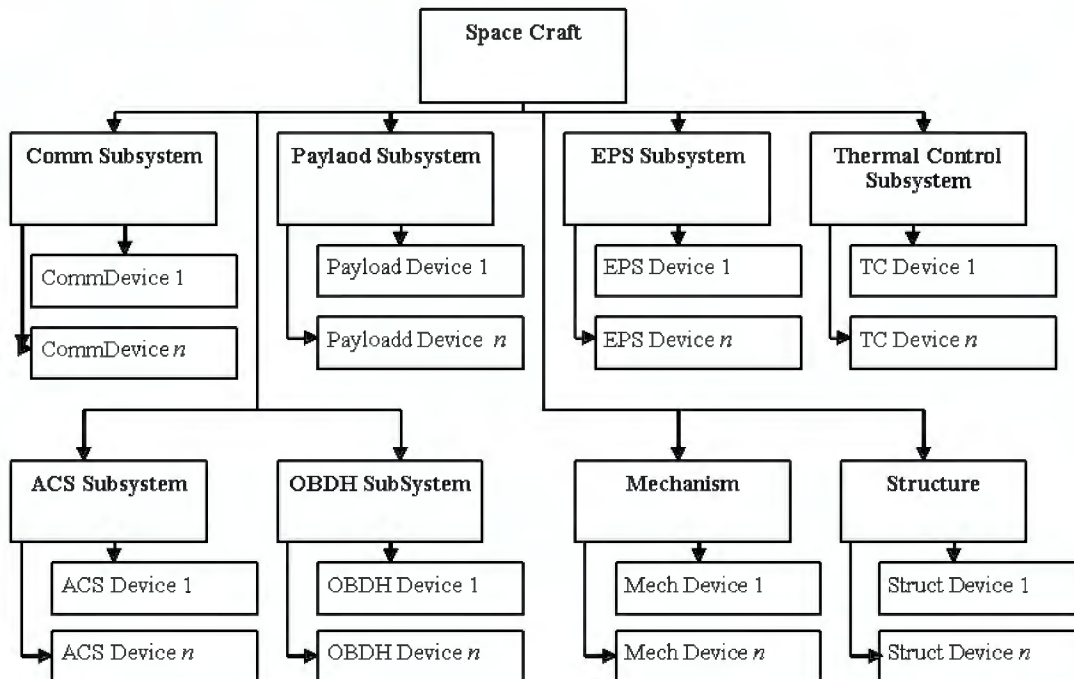


Fig. 10-1 Spacecraft model Architecture

The above layout shows the generic spacecraft layout in OpenSimKit, the spacecraft with different subsystems and its elements in a hierarchical order.

10.2 Reference Coordinates

One of the first requirements for describing an orbit is to define a suitable inertial reference frame. The attitude describes the orientation of one reference frame to another one, such as a spacecraft body-fixed frame with respect to an Earth-fixed frame. An orbit frame is also required. A set of reference frames and the methods for representing the orientation of these frames with respect to one another are defined. A reference frame is a set of three orthogonal vectors in space that are used to describe a set of coordinates. To define a frame, one of the vector directions must be specified, a preferred, or desired, direction for a second vector is

chosen, and the third direction is determined by right-handed orthogonality. There are numerous reference frames to be used describing spacecraft attitude. These frames can be highly dependent on the mission scenario, operating characteristics, or project standards. Most attitude simulations and analyses are done with respect to spacecraft-fixed coordinates (origin moving with the spacecraft), but may include non-spacecraft reference frames for further analysis.

10.2.1 Earth Centred Inertial Frame

The earth centred Inertial frame is the most used reference frame in the Navigation system application. This frame is a fixed frame in space. It is a non accelerated frame in which the Newton's law are valid. The origin of the frame is located at the centre of the Earth, where the Z-axis points towards the North Pole, X-axis towards the Vernal Equinox, the point where the Ecliptic or the plane of the Earth's orbit about the sun crosses the equator going from south to north and the Y-axis completed the right handed Cartesian coordinate system. In general the X, Y, Z are represented as I, J, K coordinates.

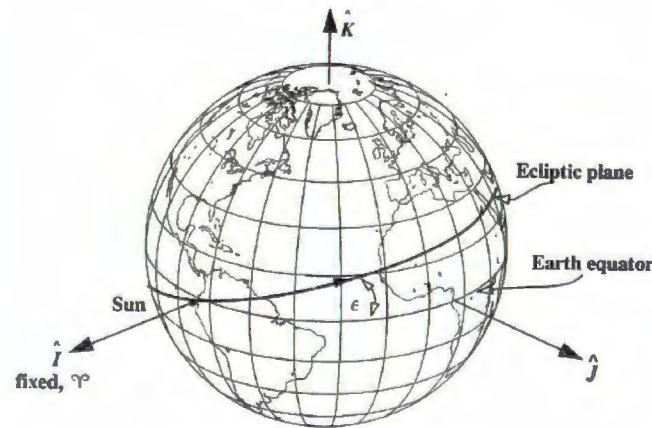


Fig. 10-2: ECI Coordinate System

10.2.2 Earth Centred Earth Fixed Coordinate system

The ECEF coordinate system is a three-dimensional Cartesian coordinate system. Its origin is at the centre of Earth's mass, its X-and Y-axes coincide with the plane of zero latitude, and the Z-axis coincides with the Earth's rotational axis. The X-axis also passes through the point of zero longitude (the prime meridian). This frame representation is essential since the IGRF data's for the magnetic field mode uses this ECEF coordinate system. The X and Y axis rotate with the earth relative to the ECI frame about the Z-axis.

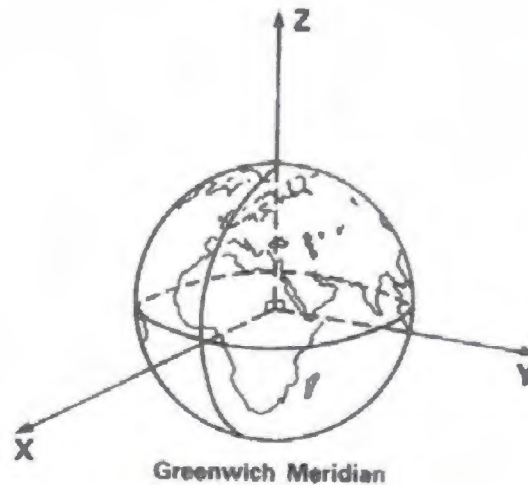


Fig. 10-3: ECEF coordinate system

10.2.3 Body Coordinate system

The Coordinate frame is fixed with the satellite and its origin is placed at the center mass of the satellite. The orientations of the satellite are described with respect or relative to the orbital frame, while angular velocities are expressed in the body frame. The axes are then enduringly described within the spacecraft as specified by the spacecraft engineers. Body frames are useful for linking objects on a spacecraft comparative to one another, or for defining how a spacecraft is oriented with respect to an external frame such as the orbital or inertial frames.

10.3 Orbit Propagator

In Open SimKit, the attitude propagator generates the position of the satellite with respect to with respect to time, An orbit can be completely described with five constants and a time varying quantity. These elements are known as Classical Orbital Elements (COE). The elements are similar to that discussed under the chapter 9.

Orbit generator provides a sequence of satellite position in the Orbit frame, ECI frame and ECEF frame and for a given orbit. For every position generated, a set of transformations between different reference frames are also computed. If all the orbit parameters are known at any given time, then estimation of future satellite position can be done with following sequence of computations.

10.4 Attitude Model

Part of designing a satellite means to assemble hardware components. During the design process the position of those components will shift causing the physical properties to change. Therefore in order to determine the position of a component one has to use coordinates, and to do on a satellite level, the *global coordinate system* is introduced. Every component has its own local coordinate system in which its geometry properties are expressed, e.g. the position of its centre of gravity. In order to merge the information expressed in all local coordinate systems the systems have to be unified into the global coordinate system. This implies that the orientation of a component's coordinate system within the satellite coordinate system must

be known. With knowledge of this orientation, certain *translations* and *rotations* can be performed. It is irrelevant whether a translation or a rotation is performed first.

10.4.1 Direction Cosine Matrix

The basic three-axis transformation is based on the direction cosine matrix. Let us take the unit vectors **1**, **2**, and **3**, defining an orthogonal, right-handed triad. This triad is chosen as the reference frame. Next, a generic orthogonal triad is defined by the unit vectors **u**, **v**, and **w**. We define the direction cosine matrix **[A]** as follows:

$$[A] = \begin{bmatrix} u_1 & u_2 & u_3 \\ v_1 & v_2 & v_3 \\ w_1 & w_2 & w_3 \end{bmatrix} \quad (10-1)$$

In this matrix, u_1, u_2, u_3 are the components of the unit vector **u** along the three axis **1**, **2**, **3** of the reference orthogonal system, and in a similar way are defined the other six components. Each of these elements is the cosine of the angle between a reference axis and an axis of the generic triad: u_1 , for example, is the cosine of the angle between **u** and the reference axis **1**.

The product of two proper real orthogonal matrices $[A] = [A2] [A1]$ is the result of two successive rotations, first by $[A1]$ and then by $[A2]$. A chain of successive rotations is common in attitude transformations.

10.4.2 Euler Angle Rotation

The Euler angle rotation is defined as successive angular rotations about the three orthogonal axes. The angles involved in the rotations are the Euler angles. The direction cosine matrix of a rotation about the axis 3 by a positive angle ψ is denoted by $A3(\psi)$; rotations about the axis 2, 1 by positive angles θ, ϕ are denoted by $A2(\theta), A1(\phi)$; in explicit form,

$$\begin{aligned} A_1(\phi) &= \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\phi & \sin\phi \\ 0 & -\sin\phi & \cos\phi \end{bmatrix} \\ A_2(\theta) &= \begin{bmatrix} \cos\theta & 0 & -\sin\theta \\ 0 & 1 & 0 \\ \sin\theta & 0 & \cos\theta \end{bmatrix} \\ A_3(\psi) &= \begin{bmatrix} \cos\psi & \sin\psi & 0 \\ -\sin\psi & \cos\psi & 0 \\ 0 & 0 & 1 \end{bmatrix} \end{aligned} \quad (10-2)$$

For attitude representation, it is common to define the Euler yaw, pitch and roll angles ψ, θ , and ϕ as successive rotations about the z, y, and x body axes. Because the matrix multiplication is not commutative, the order of multiplication must be defined.

In OpenSimKit, the orientation of the satellite is given by Quaternion's, since quaternion's eradicated the singularity problem(Marcel.J.Sidi, 2001-02).Using the Euler angles, the orientations can be expressed in terms of quaternion's(q_1, q_2, q_3, q_4). The orientation of the spacecraft varies with respect to time. The change is orientation is given by,



$$\frac{d}{dt}q = \frac{1}{2}[\dot{\Omega}]q \quad (10-3)$$

$$[\dot{\Omega}] = \begin{bmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ \omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{bmatrix} \quad (10-4)$$

10.4.3 Dynamic Equation of Motion

The dynamic equation of motion is governed by Euler Equation of motion. An undisturbed spinning spacecraft will maintain a constant spin axis direction in inertial space, no matter where the satellite is along the orbit. In real life, a spin stabilized satellite is disturbed all the time, either intentionally to achieve a reorientation of the spin axis or involuntarily by internal or external forces. The spin rate ω will change if a torque is applied along the spin axis.

$$T = \frac{dH}{dt} = I \frac{d\omega}{dt} \quad (10-5)$$

In this case the orientation of the spin axis does not change. When the torque applied perpendicular to the spin axis will change its orientation but not its magnitude (Berlin, 2005).

The torque vector has three components T_x, T_y, T_z , same with the resulting spin vector ω namely $\omega_x, \omega_y, \omega_z$. This leads us to the intuitively obvious but nonetheless important conclusion that an arbitrary torque will cause a rotational movement not just around the spin axis, but around all three axis.

The Euler's equation of motion and is a close cousin of the Coriolis theorem. The equation is given as,

$$T = \frac{dH}{dt} + \omega \times H \quad (10-6)$$

Given that H is the angular momentum, with the expression

$$H = I\omega \quad (10-7)$$

The differential of the above equation will be,

$$\frac{dH}{dt} = [I]\dot{\omega} \quad (10-8)$$

When adding a moving body like momentum wheels in the space craft, it changes the above equation as

$$\frac{dH}{dt} = [I]\frac{d\omega}{dt} + \frac{d[I]}{dt}\omega \quad (10-9)$$

Hence applying equation 10-7 in 10-4,

$$T = [I]\frac{d\omega}{dt} + \frac{d[I]}{dt}\omega + \omega \times H \quad (10-10)$$

Taking $\frac{d\omega}{dt}$ to the left hand side,

$$\frac{d\omega}{dt} = [I]^{-1} \left([T] - \left(\frac{d[I]}{dt} \cdot \omega \right) - (\omega \times H) \right) \quad (10-11)$$

The above equation is the derived equation for the angular acceleration.

T will be calculated as sum of the following disturbance torques:



- Earth gravity gradient torque.
- Aerodynamic Torque.
- Solar-radiation pressure torque.
- Magnetic disturbance torque.

10.5 Disturbance Torques

10.5.1 Residual Magnetic field Torque

The attitude propagator requires Earth's reference magnetic field intensity information to compare the magnetic field intensity measured by the Magnetometer. The reference magnetic field values should either be known priori or estimated. Due to constraints of storage space on-board the satellite computer, storing large amount data for expected magnetic field values is infeasible. Instead, the reference value is estimated based on the satellite positions generated by the orbit generator. To calculate the earth's magnetic field, OpenSimKit uses the International Geomagnetic Reference Field (IGRF) 2005 data. The g and h values are mapped from the IGRF data to obtain the magnetic field value B_{ijk} in the inertial coordinates. The IGRF is represented in the spherical coordinate system. Hence the satellite orientations have to be converted to spherical coordinates and then back to the inertial coordinates. As a part of the thesis work the pseudo code for the residual magnetic field is implemented. The reference for the mathematical model was deduced from this reference (Pippia, 2007). This reference work deals with the development of a spacecraft dynamics propagator using MATLAB, which includes the calculation of the earth's magnetic field.

The instantaneous magnetic disturbance torque is given by,

$$T_{mag} = \bar{M} \times \bar{B} \quad (10-12)$$

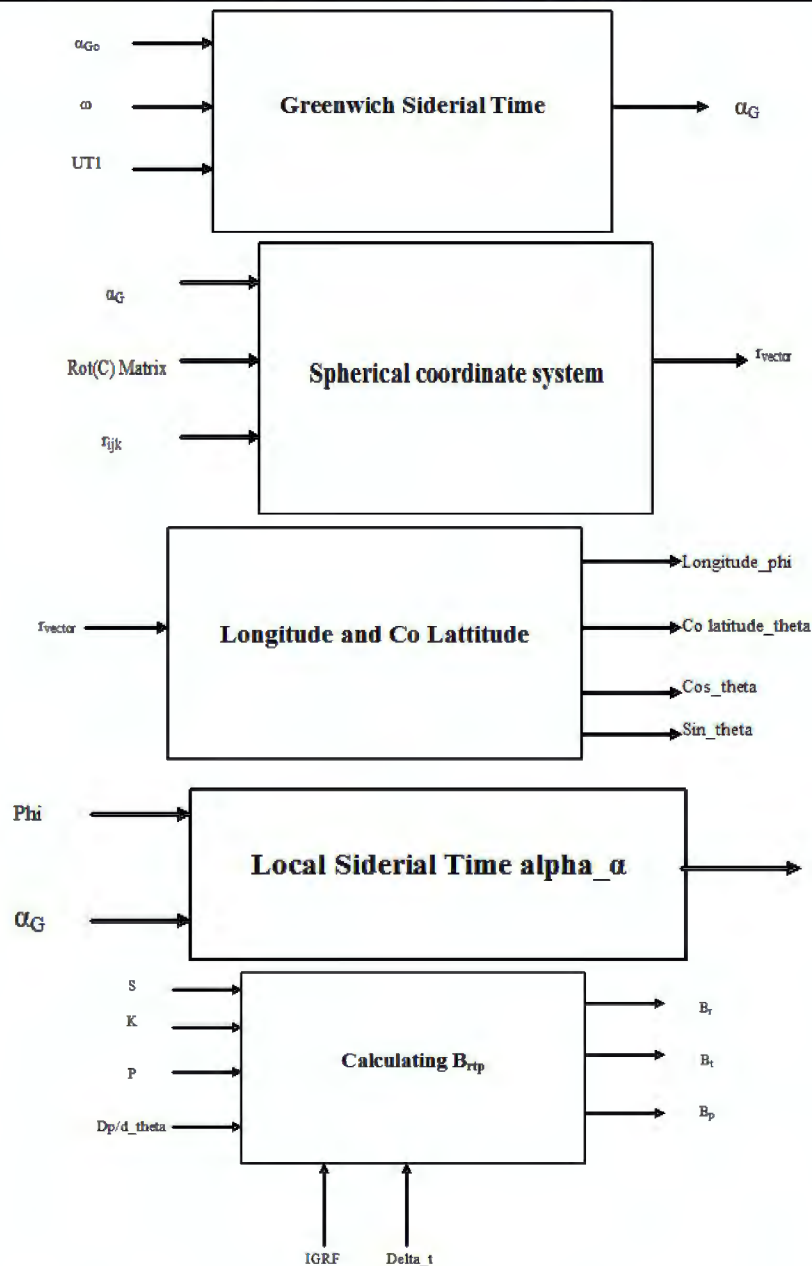
Where $\bar{B}$ is the geocentric magnetic flux density and $\bar{M}$ is the sum of the individual magnetic moments caused by the induced and the permanent magnets present in the spacecraft.

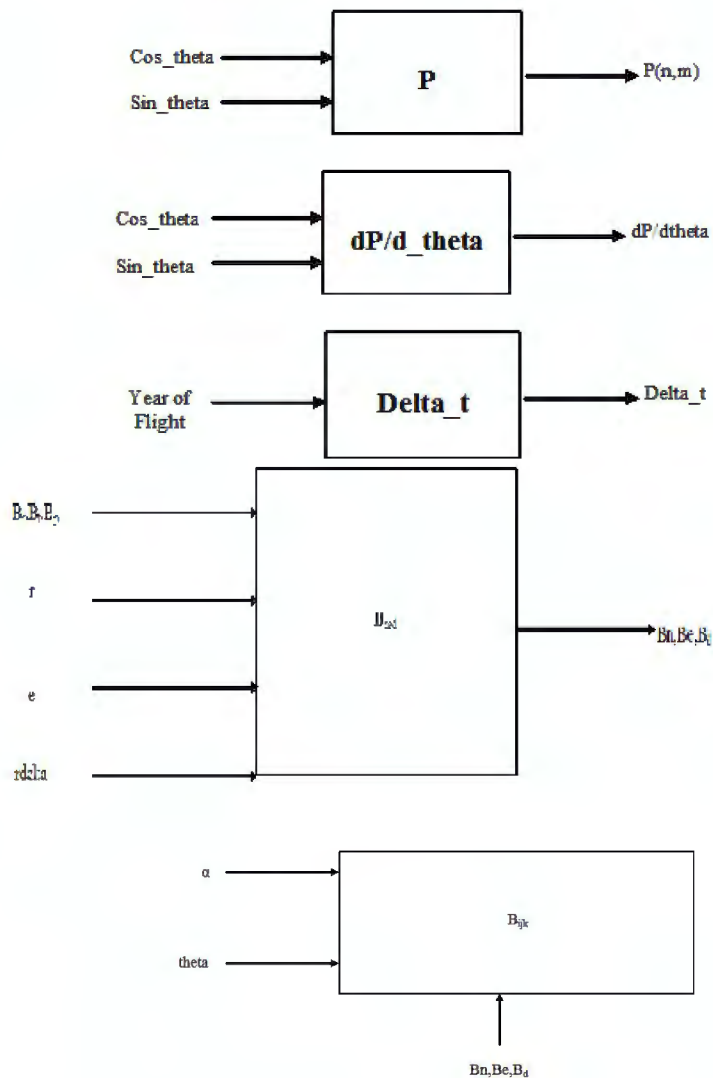
With respect to the MOVE static model the magnetic moment of the permanent magnet is calculated to be 0.023 Am^2 . Based on the earth's magnetic flux, the T_{mag} will vary.

The blocks below represent the conversion of the geocentric flux from the Spherical coordinates to the ECI coordinates.

r_{ijk}	position of satellite in IJK [km]
ω	Angular velocity of the earth's rotation
UT1	Universal Time
α_G	Greenwich sidereal time
yr	year of flight [decimal format]
α_{Go}	Greenwich sidereal time at midnight of epoch
a	Earth's radius [km]
r	position of satellite in Earth Fixed Greenwich Cartesian Coordinates [km]
Phi	longitude [rad]
Theta	co-latitude (pi-latitude) [rad]
gh	Gaussian coefficients (g,h,g',h') [nT,nT/s]

Deltat	Amount of time from 2005 [years, decimal format]
B_r, B_t, B_p	Magnetic field in Earth Fixed Greenwich Spherical Coordinates [nT]
B_n, B_e, B_d	Magnetic field in north, east, down components [nT]
B_{ijk}	magnetic field in Inertial Reference Frame [nT]
α	local sidereal time
e	Earth eccentricity
P	Schmidt Function
S	Recursive Function
Rdelta	position vector magnitude on equatorial plane [km]





The mathematical equation for calculating the magnetic field in the ECI coordinates are referred from the master thesis of (Pippia, 2007). The B_{ijk} will be used for the calculation of the T_{mag} .

10.5.2 Aerodynamic Torque

The interaction of the upper atmosphere with a satellite surface produces a force directly opposite to the velocity of the satellite, and a torque about the centre of mass. The force due to the impact of atmospheric molecules on the spacecraft surface can be modelled as an elastic impact without reflection. The incident particle energy is usually completely absorbed. The particle escapes after reaching thermal equilibrium with the surface with a thermal velocity equal to that of the surface molecules. Because this velocity is substantially less than that of the incident molecules, the impact can be modelled as if the incident particles lose their entire energy on collision.

$$A_{drag} = \frac{1}{M} \sum dF_i \quad (10-13)$$



Where dF_i is the force acting on the i th surface element dA_i with outward normal n_i , and it is given by the well known expression

$$dF_i = \frac{1}{2} C_d \rho V_r^2 (n_i \cdot V_r) V_r dA_i \quad (10-14)$$

The integral is over the spacecraft surface for which $n_i \cdot V_r > 0$. V_r is the satellite velocity with respect to the atmosphere. The drag coefficient C_d is a dimensionless quantity that describes the interaction of the atmosphere with the surface material. This coefficient depends on the interaction of the atmospheric constituents with the satellite surface; in particular on Knudsen number and scattering mechanism. The pseudo code for the above equation is built as a part this work for OpenSimKit.

The aerodynamic torque N_{Aero} acting on the spacecraft due to the forces dF_i defined by

$$N_{Aero} = \sum r_i \times dF_i \quad (10-15)$$

Where r is the vector from the spacecraft's centre of mass to the i th surface element dA .

10.6 Future Work with OpenSimKit

The initial plan for the thesis was to develop a dynamic model of the MOVE in OpenSimKit. Once on creating all the functionalities required for the satellite mission simulation. The Subsystems and systems of the MOVE CubeSat can be modelled to get more realistic values and create a more reasonable scenario to design and verify the mission. The next level of implementation will be, to develop a pseudo code for the solar torque to complete the attitude dynamics. Later, to implement the generic dynamic satellite model for power system and communication system. To further add to this thesis work the above models will be researched and developed. Based on the comparison and suggestions given the following chapter will help to further develop the current thesis work and as well the OpenSimKit.

11 Comparison between Modelling approaches

11.1 (v)Sys

The (v)-Sys carries the major proportion of the thesis work. The (v)-Sys works on an object oriented approach, which looks like a UML Modelling. Before using the software, the user needs to understand the system engineering concept to model any system. As a name suggests, the software uses a systems engineering approach from an element level to the complete system level. The author of this thesis work does not have a background experience with any of the systems engineering software other than MATLAB simulink and LABVIEW, even though the above mentioned software looks completely different from (v)-Sys. On a first look, the software is easy to learn and understand the exact usage of the software. Once on understanding the order relations and flow relation, the user can directly jump into the software and start using it. The ambience of the software looks comfortable and the placements of the required icons are well visible.

+VE:

- The Software uses a Backend as MS Excel; it is well know office software among engineers.
- The GUI of the software is as common as any other technical software.
- Does not require a manual to learn or test the software.
- The data types for writing the model functions are open to users.
- The functionality of the models can be shared among the class or the projects.
- Instantiation of the base class can be done, to reduce the work of the user.
- The manipulations are made using the normal Excel programming Language.
- The Excel also support Visual Basic program as a Macro to support integrative process, and manipulations which are not possible in Excel.
- It is generic program for any engineering application or even organizational application.

-VE:

- Currently the system can be used only for Static modelling; dynamic aspects need to be applied on static models.
- The program hangs or shuts down without reason.
- While adding input to a model block, it model needs to be refreshed (check Function), which consumes time.
- Every time the systems model needs to be refreshed, when a change is made in the element level.

Rating:



11.2 SIMVIS

The SIMVIS is the dynamic modelling software developed by VEGA GmbH for space application. The software uses C++ and FORTRAN as the base language. The SIMVIS is composed of three sections, SIMVIS Designer, SIMSAT and OpenGL API. The initial static conditions are given as an input through Excel sheets; consist of the subsystem parameter entry and visual Basics Macros, which translates the values in to the designer. The SIMVIS



designers consist of the wizards. Space mission Wizard, Spacecraft Subsystem wizard etc. The wizards read the input valued from the Excel sheet to initialize the simulation. The designer consists of different settings, mission setting, Assembly setting and scheduler setting etc. The model functionalities are inbuilt in the software. The functionalities use C++ codes. New function can be created (never worked when tried for this thesis work).

+VE:

- The software is used for dynamic simulation.
- Contains inbuilt satellite models (to help the user to understand).
- Can be used for different satellite application and modelling.
- The functional codes are transparent to the user.

-VE:

- The software lack user friendliness.
- The software is so moody; it works only with system which it likes. It works with the system, which has different configuration to the required.
- The layout of the software is complex. Setting the simulation using an Excel sheet, then to a designer and then to a simulator, finally to the visualizer.
- All the three elements of the software work with different language. Visual Basics, C++ and XML. The user should have a background of the mentioned languages to use this software.
- The software user requires training and consultation while using the software. Else the user is left in a dark room without a torch light.
- The interface between the software elements is poor. Every time, an exception message is generated. And every time the user has to verify, the call of catalogues and libraries.
- The man hour spend on developing a dynamic model is too high, where time is directly proportional to money.
- After using the software for more than month, still not able to set the dynamic aspects, like changing the spacecraft modes. For example, while creating a scenario, the spacecraft modes need to be considered and designed. There is no link between the simulation time and the spacecraft modes. Hence dynamic analysis of power system is impossible, with this experience. Though, the point is, this software is used in ESA for Modelling of satellites.
- This shows without an introductory course on this software, it is highly impossible to set up a model for dynamic simulation.
- There is no forum in the web to at least take help or share knowledge.
- The OpenGL needs a XML programming knowledge to understand and write or to debug.
- On the whole the software is not professional software.
- Does not have a 2D visualization.

Rating



11.3 STK

The Satellite Tool Kit is the static and dynamic modelling software used for Aerospace, Defence, Space Automotive and Navel applications. The Software is developed by Analytical Graphics, Inc.AGI. At the institute of Astronautics, the STK is being used for the satellite



modelling application. The Satellite tool Kit software is currently the most predominantly used software by all the institutes and satellite builders. In this thesis work, STK is used as a verification tool to verify the results from SIMVIS. It is mainly used for the visualization of the satellite elements in orbit. The STK contains all the basic functions required to do the verification process. To write extra functions, MATLAB codes can be imported if required.

+VE:

- It is well developed software; it is of no comparison with SIMVIS.
- It is user friendly software. Takes less time to understand the basic functionalities and setting of the software.
- Unstablized satellite can be modelled with inbuilt functions.
- The 3D visualization and 2D visualization gives the user lot of space to play, think and analyse with the results.
- The user is not required to wait for the whole simulation time like SIMVIS. On setting the initial conditions, the end results can be visualized immediately.
- The colour schemes and the ambience of the software motivate the user to further explore with the software.

-VE:

- The software does not have an auto save function, the settings are lost if the user forgets to save.

Rating:



11.4 OpenSimKit

By keeping all the +ve and -ve about the different software's, the OpenSimKit has to be developed in order to have good and comfortable product in market. The OpenSimKit is expected to a GUI like STK and with background much more easier to rewrite or modify according to the requirements of the user. The OpenSimKit should have all the simulation support independently rather than like SIMVIS. The software should have both 2D visualization and 3D visualization. The satellite modes and simulation time should be linked to give more realistic results. Modelling MOVE mission in OpenSimKit, will give more realistic verification models. This could give a better idea about the satellite operation in space.

12 CONCLUSION

This thesis work gives a detailed explanation about the first MOVE CubeSat mission. The static modelling and the dynamic aspects assumed for the static modelling approach. The power budgeting, mass budgeting, data budgeting and link budgeting were implemented in (v)-Sys and verified with the scenario. The CubeSat subsystems were dimensioned and modelled using the (v)-Sys software in a systems engineering approach. After the initial dimensioning and verification of the system elements the dynamic aspects needed to be established using visualization software. The important elements of the CubeSat are the camera, antenna and sensors used were visualized using the ESA's SIMVIS and verified using the Satellite tool kit STK. Various aspects like time of coverage with the target and ground station and the antenna contact time with the ground station are verified and analysed. Out of the visualization results the dynamic aspects were implemented in the static model of (v) - Sys. The mass budget results show that the MOVE CubeSat is well below the 1kg limits including the deployable panels. This needs to be verified once again after the integration of the system. A mass allowance of $\pm 10\%$ is expected. The power budget calculations were made based on the satellite modes and critical elements of the satellite. After studying on different scenario combinations, the total energy consumption was calculated to be 14.1Wh. The power generation calculation shows that the power generated by the solar panels are of an average of 2.33W and the total energy generated during an orbit will be 33Wh, which is significantly more than the consumption. In the future CubeSat mission on inclusion of the payload cells for power generation will increase the average power generation to more than 3W. The link budget scenario was established for the downlink as well the uplink. The dynamic variation of the satellite antenna gain were taken into consideration and verified to have the communication system to have a link margin more than the 3dB margin. By taking concern of the all the critical subsystem budgets and analysis, the mission is verified and necessary suggestion were made to improve the system design and performance. The OpenSimKit software is currently under development process and the pseudo codes for few of the important functionalities are being developed and coded with C++.As a future prospect to this work, on completion of the OpenSimKit software, the MOVE mission can be implemented and verified with the dynamic aspects.

12.1 Lessons Learnt

The lessons learnt out of this thesis work were, it has given a detailed overlook to the satellite subsystems and various other functionalities supporting the total system. It has given an idea on how to approach systems engineering problems from a systems engineer's point of view. This work has given a detailed view on the different CubeSat missions conducted in different part of the world. Doing the systems engineering is much easy with a small scale satellite. Even though the scale of the satellite is small comparatively, it has given the generic idea of dimensioning a satellite, irrespective of the size of the satellite or the application. Also learnt to work in a group and coordinate with them to attain the ultimate goal. This opportunity has given a personal improvement by attitude, behaviour and patience.

A Bibliography

A.1 Reference

- Tribble, C. Alan. . *Space Environment and Implication for space design*. Princeton University press (2003).
- Assmann, K. *Orbit Report Phase C/D*. (2007) LRT Muenchen.
- Balanis, C. A. *Antenna Theory Analysis and Design*. (2003)
- Berlin, P. *Satellite Platform Design*. Netherlands. (2005)
- BharatRakshak ISRO. (kein Datum). Von Bharat Rakshak: www.BharatRakshak.com abgerufen
- Cpt. Anthony, R. *Microdot- a four bit Microcontroller designed for distributed low end computing in satellite*. (2003)
- Delft, T. U. (kein Datum). *DelfiC3*. Von DelfiC3 project: <http://www.delfic3.nl/> abgerufen
- Granite Island group. (2005). Abgerufen am 2008 von Technical Surveillance counter Measures: www.tscm.com
- Hager, P. *Development of a Generic Model*. LRT Muenchen: Semester Thesis. (2007)
- Harder, J. *Realisierung eines Testenders für die LRT Bodenstation*. LRT München: Semesterarbeit. (2007)
- Hales, H. Jan, M. P. Two Axis MOEMS Sun-Sensor for Pico satellite. *16th Annual AIAA/USU Conference on Small Satellites* (S. 1-12). Technical University of Denmark (DTU): Mikroelektronik Centret (MIC). (2007)
- Sweeting, M. N. *Cost Effective Microsatellite for Low Earth Orbit Communications*. Guildford, Surrey: Center for Satellite Engineering Research, University of Surrey. (1994)
- M. N. Sweeting. Uosat Microsatellite Missions. *Electronics and communication Engineering Journal*, 141-150. (1992).
- Marcel, J. Sidi. *Spacecraft Dynamics and Control*. Cambridge, UK: Cambridge University Press. (2001-02)
- Parasher, A. Dartmouth Nanosatellites as Orbital Research Platform . *Dartmouth Undergraduate Journal of Science*, 6-10. (2003)
- Peter Fortescue, J. P. *Spacecraft System Engineering*. Wiley. (2002)
- Pippia, G. *A Propagator For Spacecraft/Orbit Coupled Dynamics*. LRT, Technical university of München: Master Thesis. (2007).
- Römhild, F. *Mission Report*. LRT München: MOVE Mission Report. (2007)
- technology, T. I. (kein Datum). *Laboratory for space system*. Von Tokyo Institute : <http://lss.mes.titech.ac.jp> abgerufen
- Toronto, U. o. (kein Datum). *Space Flight Laboratory*. Von CAN-X2: <http://www.utias-sfl.net/nanosatellites/CanX2/> abgerufen
- Twigg, J. P.-S.. *Cubesat design specifications document*. . Californian Polytechnic State University. (August 2003)
- University, A. *AAUSAT II*. Von Aalborg University: www.aasatii.space.aau.dk abgerufen, (2006)
- University, A. *AubieSat-I*. Auburn: Auburn University student space program. (2007)
- Wertz, J. *Space Mission Analysis and Design*. Wiley. (1999).



Wiesel, W. E. *Space flight dynamics*. (1997).

Zeigler, F. *Modellierung eines generischen Satelliten Energieversorgungs systems*. LRT, TUM, München: DiplomaArbiet. (2008)