

# **DEVELOPMEN OF 2U NANO SATELLITE WITH ATMOSPHERIC SENSING**

Submitted in partial fulfilment of the requirements for the award of

Bachelor of Engineering degree in

Aeronautical Engineering

By

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## **SATHYABAMA**

**INSTITUTE OF SCIENCE AND TECHNOLOGY**

**(DEEMED TO BE UNIVERSITY)**

**Accredited with grade “A” by NAAC**

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**APRIL 2021**



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**DEPARTMENT OF AERONAUTICAL ENGINEERING**

**BONAFIDE CERTIFICATE**

This is to certify that this Project Report is the bonafide work of CH .  
MANIKUMAR(Reg. No. 38260008) and K .S KARTHIK (Reg. No. 38260021)  
who carried out the project entitled **DEVELOPMEN OF 2U NANO SATELLITE WITH  
ATMOSPHERIC SENSING** under our supervision from September 2019 to April  
2020.

**Internal Guide**

**Dr. VENKATESH S . M.E.,**

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**Internal Examiner**

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## DECLARATION

We ABCDEF GHIJK (Reg.No. 3615xxx) hereby declare that the Project Report entitled "**DEVELOPMEN OF 2U NANO SATELLITE WITH ATMOSPHERIC SENSING** " done by us under the guidance of **Dr .VENKATESH, M.E., Ph.D.**, is submitted in partial fulfilment of the requirements for the award of Bachelor of Engineering degree in Aeronautical Engineering.

DATE: .04.2021

SIGNATURE OF THE CANDIDATE

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I would like to express my sincere and deep sense of gratitude to my Project Guide **Dr . VENKATESH S., M.E**, for his valuable guidance, suggestions and constant encouragement paved way for the successful completion of my project work.

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## 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.

The first Nano Satellite will verify the satellite platform and new highly efficient solar cells from industry. It is a CubeSat in single-configuration (10x10x12 cm<sup>3</sup>) which features deployable solar arrays. 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 Nano Satellite, 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 major aim of this project is to fabricate the NanoSat module for measuring the atmospheric parameters such as humidity, temperature, altitude and speed of the object.

## Acknowledgement

This report is the final thesis of my project work which I carried out as a part of the Master of Engineering course in Aeronautical Engineering at Nehru Institute of Engineering & Technology, Coimbatore. I have performed the necessary practical work of which a detailed description of the activities is provided in this thesis report. First of all I would like to thank god almighty for his blessings on me. Then I would like to thank **Dr.P.Krishna Kumar**, CEO & Secretary, Nehru Group of Institutions for his kind support by providing us the opportunity to learn and utilize the necessary facilities in our institution.

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speed , distance, sat count

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# **CHAPTER 1**

## **INTRODUCTION**

### **1.1 NANO SATELLITES**

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. 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.

The first satellite First Nano Satellite will verify the satellite platform and new highly efficient solar cells from industry. It is a CubeSat in single-configuration (10x10x10 cm<sup>3</sup>) which features deployable solar arrays. This design improves the performance of such small CubeSats in several ways. 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.

## 1.2 EVOLUTION

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. 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:

| 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       |

**Table 1: Satellite categorization with respect to mass**

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.

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. There has been a common belief 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 niche 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.

### **1.3 MAJOR AREAS OF WORK**

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. 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.

## **CHAPTER 2**

### **LITERATURE REVIEW**

- 1 N. Adam, V. Atluri, S. Yu and Y. Yesha - 10th NASA Goddard Conference on Mass Storage Systems and Technologies 19th IEEE Symposium on Mass Storage Systems, April 15–18, College Park, MA (2002).** This journal Efficient Storage and Management of Environmental Information proposes a suitable architecture for data collection and monitoring of environmental parameters and that allows specification of the queries and their visual presentation.

- 2 **Karthik Krishnamurthy, Suraj Thapa, Lokesh Kothari and Arun Prakash, (2015)** in this work 'Arduino-Based Weather Monitoring System' explains Arduino is an open-source platform that enables us to quickly build electronics projects. It consists of both a physical Programmable Circuit Board (PCB) and a piece of software (an Integrated Development Environment (IDE)) that works on all known operating systems. We use the Arduino to develop a weather monitoring system based on temperature and humidity variables obtained from a DHT11 sensor. The system, when tested, was able to report if weather is Hot, Normal, or Cold based on the exact temperature and relative humidity within a 20meter area.
- 3 **Amy Hutputanasin, Arash Mehrparvar, Wenschel Lan I, (2014)** in the work CubeSat Design Specification Rev. 13 The CubeSat Program, Cal Poly SLO, that closely meet the requirements for reliability, security, low power and low cost and standard -based, design data which will address the unique needs of standardised dimensional data for 1U, 2U & 3U cubesats.
- 4 **Thu Vu Trong, Tri Dinh Quoc, Anh Nguyen Tuan (2009) work Design and manufacture of a nanosatellite for space technology education and potential application** this paper will describe the experience in developing a simple ground receiving station and the basic design of F-1 nanosatellite and gives an insight regarding the design, manufacturing, components and working of a Nano Satellite and its ground control station.
- 5 **H.R.Chiranjeeve<sup>1</sup>, K.Kalaichelvan<sup>2</sup>, A.Rajadurai<sup>3</sup> work Design and vibration analysis of a 2u-cubesat structure using aa-6061 for aunsat – II** provides the ideas for design and analysis of modular 2U Cubesat structure based upon the Cubesat standard for the exploration of the lower thermosphere were carried out in this project. The structure composed of many sub components and multiple mounting configurations, giving maximum flexibility in the design process. The CAD model of the various parts and their assembly were made using SOLIDWORKS 2014 cad software and subjected to FE analysis using ANSYS 14 in a static load of 9g and 13g.

- 6 **Mohamed Ibrahim, Student Member, IEEE, and Moustafa Youssef, Senior Member, IEEE work CellSense: An Accurate Energy-Efficient GSM Positioning System** evaluates the effect of changing the different system parameters on the accuracy-complexity tradeoff and how the cell towers density and fingerprint density affect the system performance.
- 7 **Z. Machacek and V. Srovnal , In 7th WSEAS International Conference (2005) work Automated system for data measuring and analyses from embedded systems** presents the design of an automated system for signal processing in embedded systems. The design is worked out for an automatic predictive diagnostic system use. This automated system allows communication with embedded systems, an embedded systems control, data collection from sensors, various signal analyses, and react to limits transcendence.

## **CHAPTER 3**

### **METHODOLOGY**

#### **3.1 INTRODUCTION**

This chapter will discuss about the flow of this Final Year Project from the beginning until finish this project. The flow chart is very important to illustrate the sequence of operations to finish the work. The flowchart is generally drawn in the early stages. It will guide to finish the works. Meanwhile, the Gantt chart shows how the project is planned and seen instantly whether the project is behind or

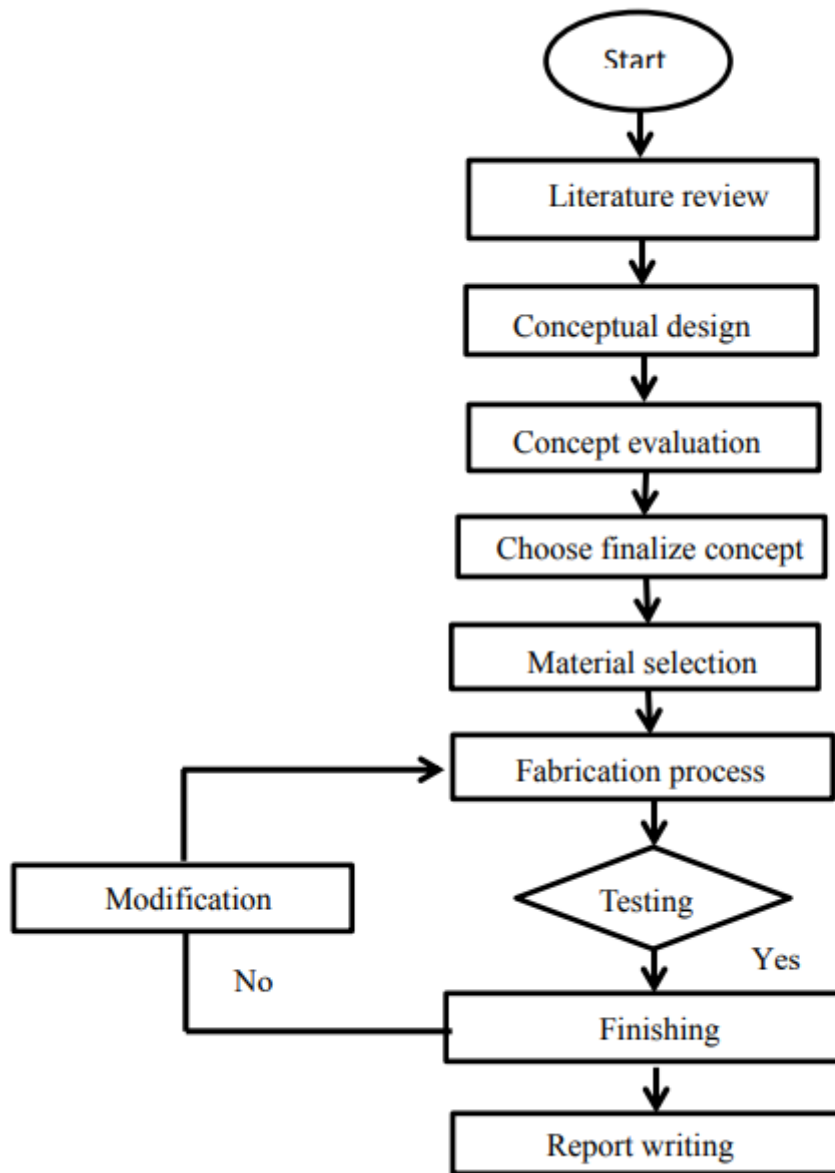
ahead of the schedule. The function of the Gantt chart is to guide towards the direction of the project plan. So, these two charts are very important to guide us to finish up the projects.

Moreover, in this chapter also will discuss about the concept generation. There were four design of product. Then, list the advantage and disadvantages of each of the product design. After that, proceed to finalize concept. This finalize concept is to choose the good one from the four design. The best product will proceed to another step, which is fabrication process. Before that, the material selection is needed to do. This is to select the suitable material that being to use. The material selection is also followed by the function of the product.

### **3.2 FLOWCHART OF THE PROJECT**

Flowchart is very important to illustrate the sequence of operations to finish the work. It use symbol to represent of a process. Each of steps in the process is represented by different symbol and contains a short description of the process step. The top of the flowchart will begin with start.





Then it followed by literature review. On this step it will discuss about the currently method use. That is collecting by using hand, by scoop and basket and rake. Apart from that, there is list of advantage and disadvantage of each method. By doing so, know that which one is the efficient and which one is take short time in collecting.

### **3.3 CONCEPTUAL DESIGN**

Next is conceptual design. 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.

## **CHAPTER 4**

### **CONCEPTUAL EVALUATION**

#### **4.1 NANO SATELLITE**

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 (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.

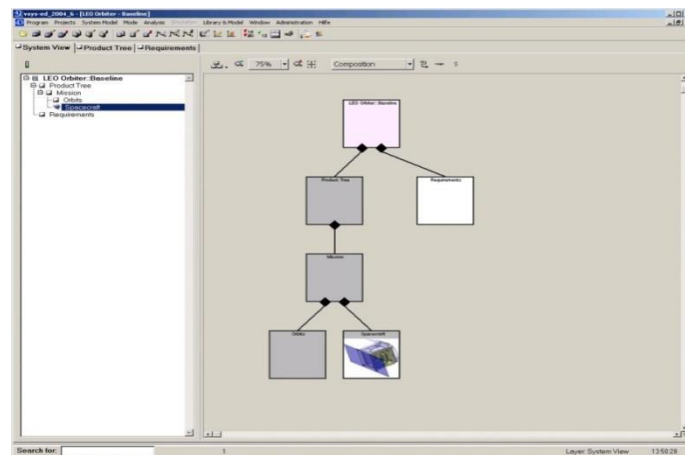
## **4.2 INTRODUCTION TO (V)-SYS**

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.

## **4.3 (V)-SYS LAYOUT**

Based on top level requirements and derived functional requirements for the Nano Satellite 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.



**Figure 1: General (v) Sys Layout**

## 4.4 MODELS

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.

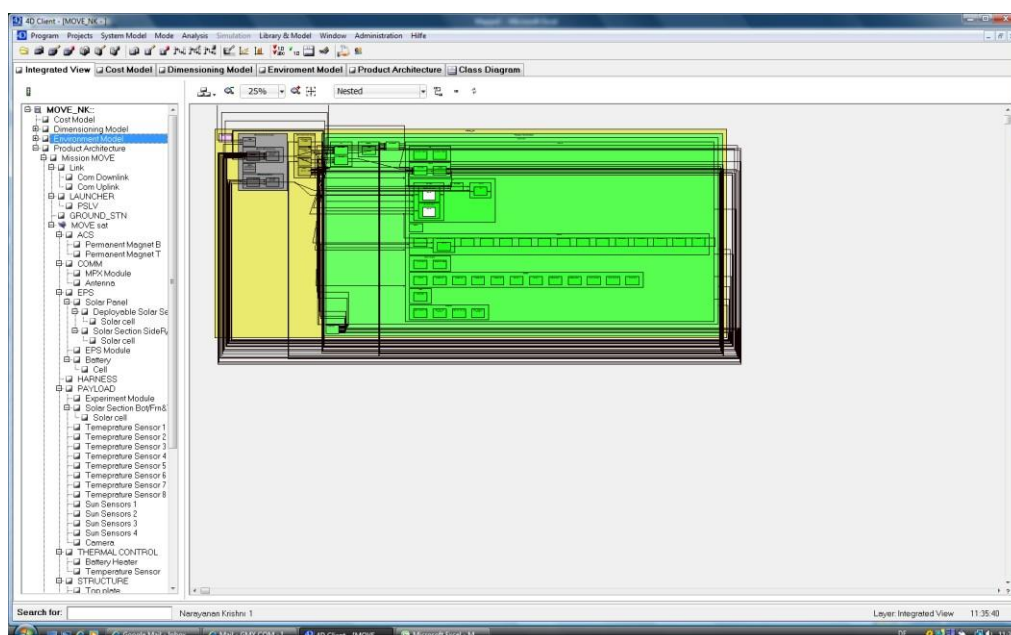
Environment model consists of all kind of environmental information and values which will interact with our system.

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

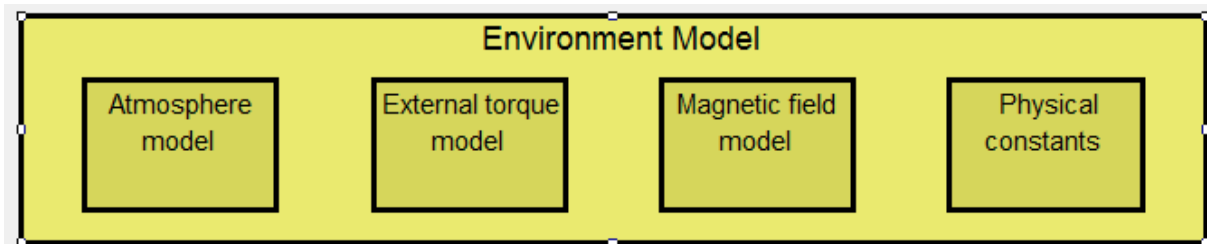
(v)-Sys provides a feature to automatically add up these critical values. If the property of concern is implemented with SUM as inheriting action (there are, of course, also other mathematical functions like MAX or MIN available) and the mass property has correctly been assigned to all system elements,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



**Figure 2: (v)-Sys Illustration of Nano SatelliteModel**

## **4.5 ENVIRONMENTAL MODEL**

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



**Figure 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)

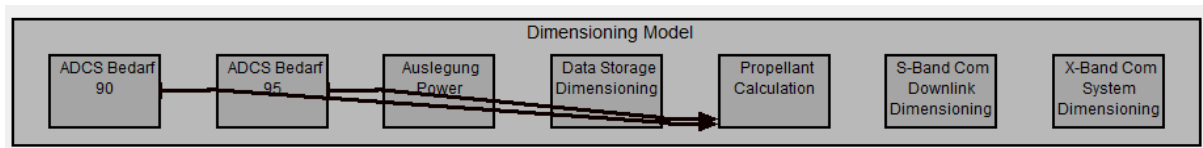
The physical constants model consists of all the commonly used physical constants for the generic satellite model calculations, for example, Boltzmann constant, solar intensity, obliquity of the ecliptic, gravity constant of the Earth, Stephan Boltzmann constant etc.

## **4.6 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,



**Figure 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.

## 4.7 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 systems 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).

## **4.8 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.

$$M_s = \sum_{n=i} M_n$$

## **4.9 COMMON PHYSICAL PROPERTIES**

### **4.9.1 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 Nano Satellitements 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, where 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.

#### 4.9.2 TRANSLATION

A translation is a Nano Satellitement 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.

$$X_{global} = X_{local} + \Delta_x$$

$$Y_{global} = Y_{local} + \Delta_y$$

### 4.9.3 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

$$I = \sum r_i^2 m_i$$

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  $[\text{kgm}^2]$ .

$$[I] = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix}$$

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

$$I' = I + M \begin{Bmatrix} \Delta y^2 + \Delta z^2 & -\Delta y \Delta x & -\Delta z \Delta x \\ -\Delta x \Delta y & \Delta x^2 + \Delta z^2 & -\Delta z \Delta y \\ -\Delta x \Delta z & -\Delta y \Delta z & \Delta x^2 + \Delta y^2 \end{Bmatrix}$$

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

$$I_{ij,rot,global} = \sum_{k,m} A_{ik} A_{jm} I_{km,local}$$

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.

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

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.

#### 4.9.4 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. 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:

$$X_{cog,global} = X_{cog,local,rot} + \Delta_x$$

$$Y_{cog,global} = Y_{cog,local,rot} + \Delta_y$$

$$Z_{cog,global} = Z_{cog,local,rot} + \Delta_z$$

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.

$$X = \frac{\sum_{i=0}^n (m_i \cdot X_i)}{\sum_{i=0}^n m_i}$$

$$Y = \frac{\sum_{i=0}^n (m_i \cdot Y_i)}{\sum_{i=0}^n m_i}$$

$$Z = \frac{\sum_{i=0}^n (m_i \cdot Z_i)}{\sum_{i=0}^n m_i}$$

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.

#### 4.9.5 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  $\epsilon$  and,  $\epsilon$  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  $\theta$  depends on the location of the component within the satellite and therefore is a local, component own, property. Moreover  $\theta$  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  $\theta$  is zero.

#### 4.10 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.

#### 4.11 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  $G_r$ .

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

$c$  is the velocity of light,  $f$  is the frequency of transmission and  $d$  is the maximum distance between ground station and the satellite. Using this Path Loss, the relationship between the received and transmitted powered may now be simply expressed with:

$$\frac{P_r}{P_t} = \frac{G_t G_r}{L_p}$$

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).

$$P_r = P_t + G_t + G_r - L_p$$

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

$$L_p = 32.34 + 20 \log \frac{d}{1 \text{ km}} + 20 \log \frac{f}{1 \text{ MHz}}$$

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

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

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).

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

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

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

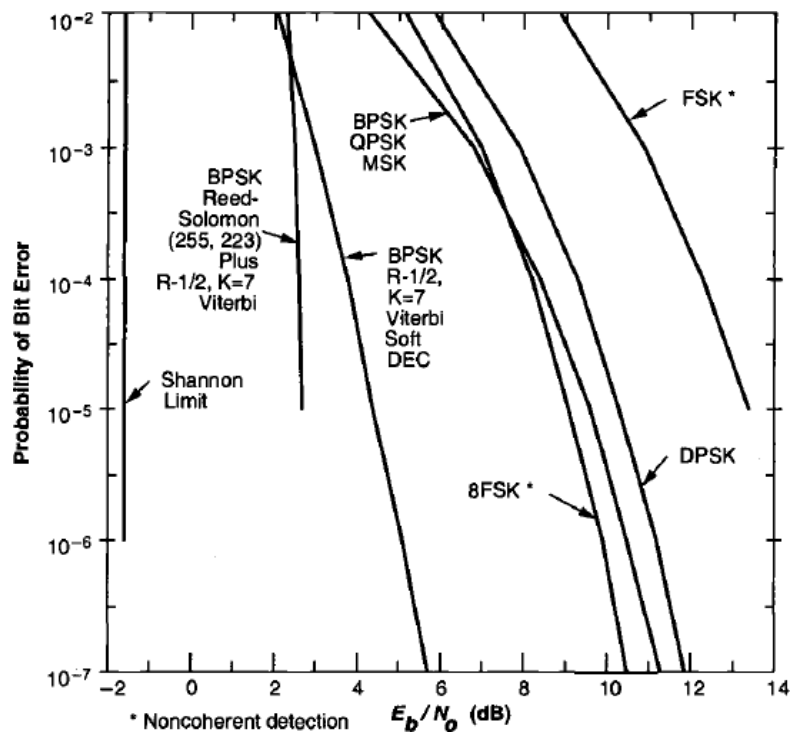
$EIRP = P_t + G_t$  is called the “Equivalent Isotropic Radiated Power” or the power required by the transmitter output stage if the antenna radiated equally in all directions (isotropic). The advantage of using EIRP is that it is possible to trade antenna gain for transmitter output power for a given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

$$\frac{G}{T} = G_r - 10 * \log \frac{T}{1 \text{ K}} [\text{dB/K}]$$

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  $E_b/N_0$  is,

$$\frac{E_b}{N_0} = \ln \left( \frac{1}{2BER} \right)$$

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  $E_b/N_0$  (energy bit to Noise spectrum) is based on a graph,



**Figure 5: BER vs  $E_b/N_0$**

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

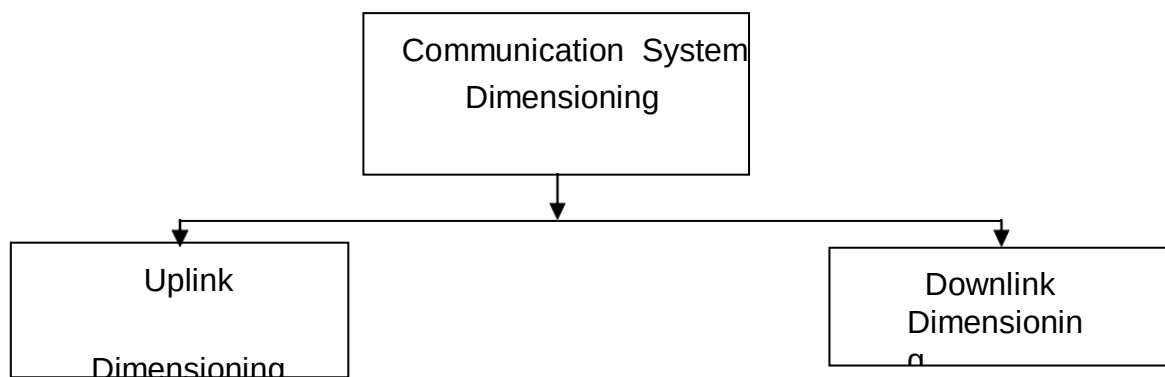
$$\left( \frac{C}{N} \right)_{req} = \frac{P}{B} \ln \left( \frac{1}{2BER} \right)$$

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

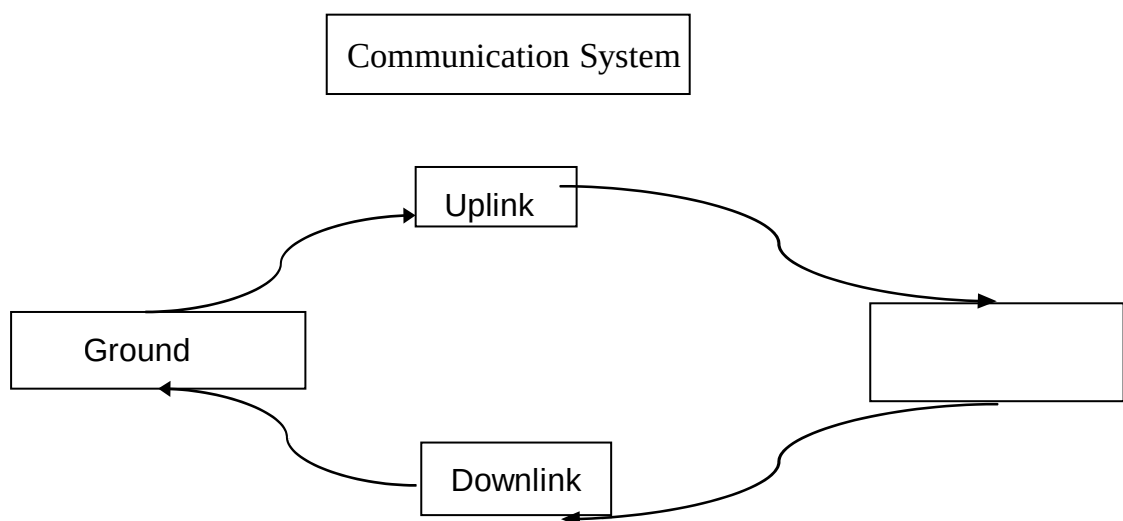
communication dimensioning model are the required data rate, type of modulation, antenna specification and the physical constants used for the link dimensioning.

The outcome of the dimension model are the specified minimum ratio of Energy per bit to noise ratio ( $E_b/N_0$ ), 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.



**Figure 6: Communication Dimensioning**



**Figure 7: Generic Communication Model for Satellit**

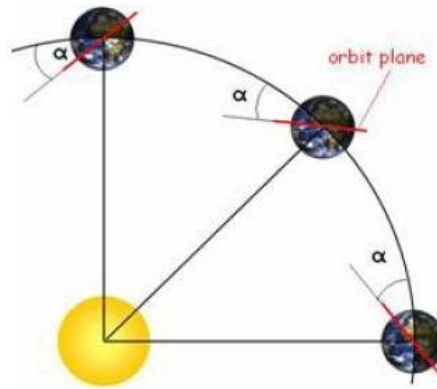
## **CHAPTER 5**

### **ORBITS & PHASES**

#### **5 NANO SATELLITE ORBIT**

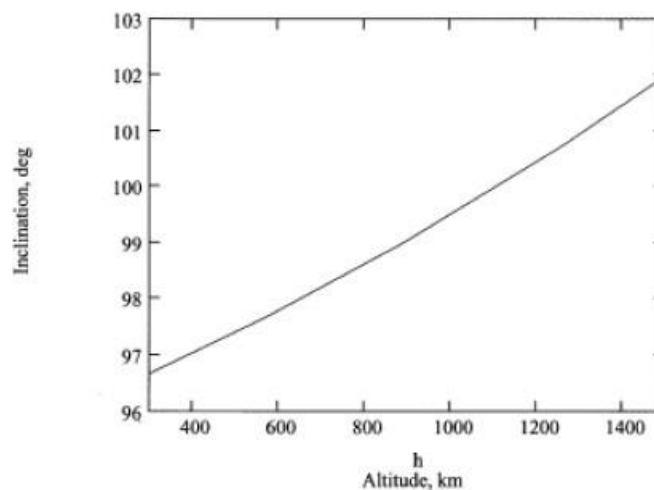
For the Nano Satellit Emission different orbits were chosen. The possibilities were sun synchronous Earth orbit where the former CubeSat missions were launched with an altitude of 600 to 850 km and a launch from the ISS. For viewing all parts of the Earth's Surface at close quarters it is necessary to 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 Nano Satellites 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.



**Figure 8: Angle  $\alpha$**

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.



**Figure 9: 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  $\alpha$  and the altitude.

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  $20^\circ < \alpha < 80^\circ$  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  $97.871^\circ$ . 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.

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.

## **5.2 MISSION PHASES**

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

### **5.2.1 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.

### **5.2.2 COMMISSIONING PHASE**

During satellite commissioning the on-board systems of the Nano 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 Nano Satellite 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.

### **5.2.3 EXPERIMENT PHASE**

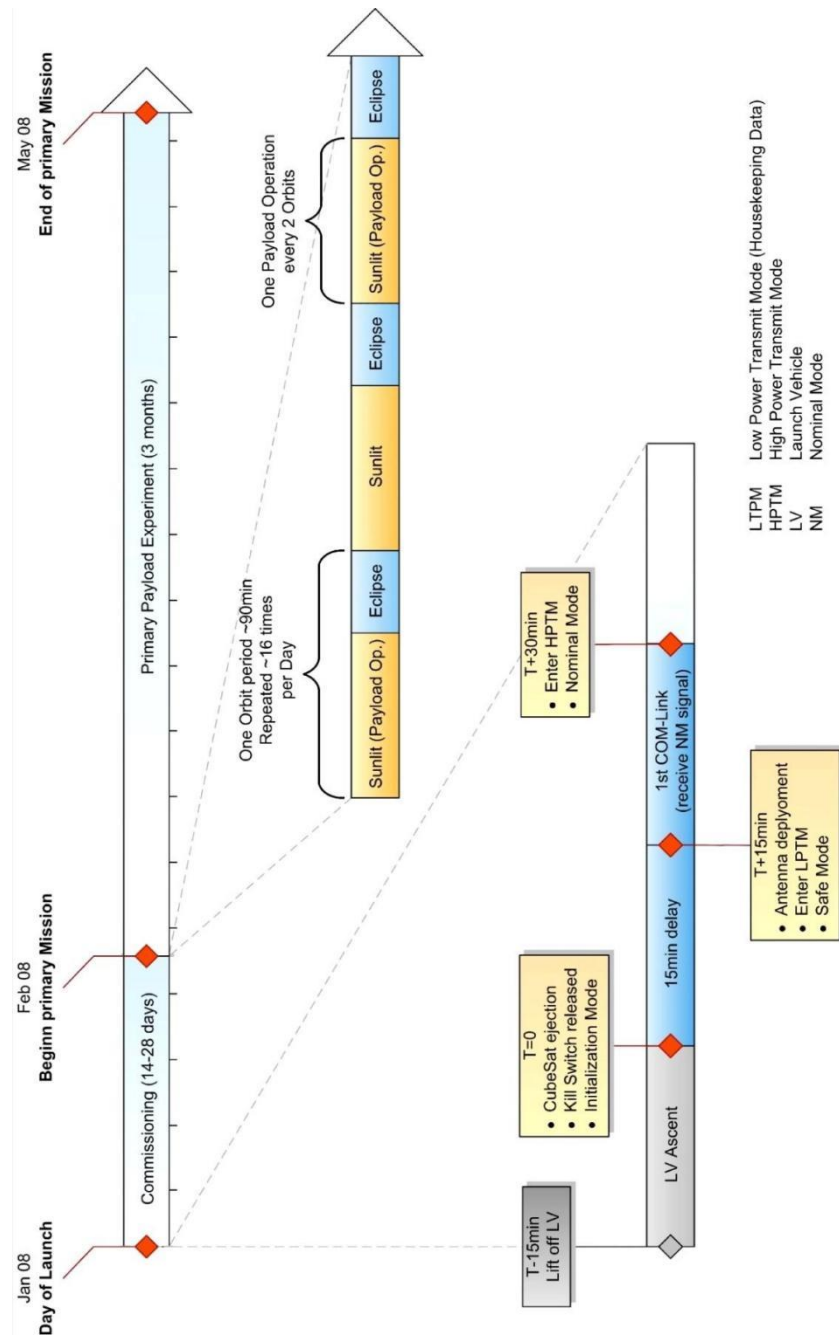
In this phase Nano Satellite's primary payload experiment is being conducted. It is important to define this separate stage for the payload operation, since it is very power consuming and only active for a short period of time. Every two orbits one test cycle is carried out for the batteries. Should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

- 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

#### **5.2.4 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.



**Figure 10: Initial Scenario design for Nano Satellite Mission**

### 5.3 SATELLITE MODES

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

### **5.2.2 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. 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.

### **5.2.3 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 Nano Satellites 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.

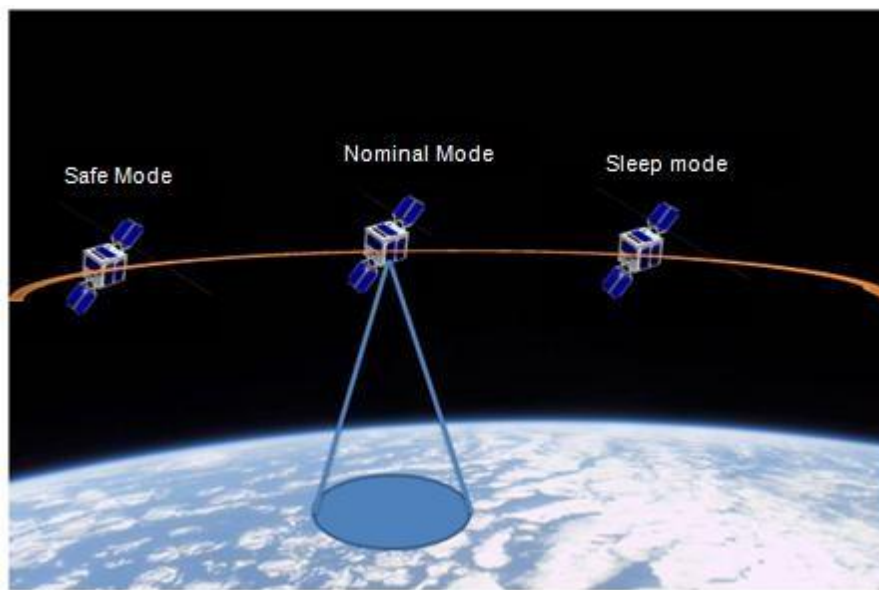
### 5.3.3 NOMINAL MODE

The main operational mode Nano Satellite 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. In this mode the CubeSats activities can be classified into different phases It is vital to define several subsystem activation limits within the Nominal Mode, since the survival of Nano Satellite 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 Nano Satellite 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.

| Modes               | EPS | COM TX | COMM RX | BEACON | OBDH | PL |
|---------------------|-----|--------|---------|--------|------|----|
| Initialization Mode |     |        |         |        |      |    |

|              |  |  |  |  |  |  |
|--------------|--|--|--|--|--|--|
| Safe Mode    |  |  |  |  |  |  |
| Sleep Mode   |  |  |  |  |  |  |
| Nominal Mode |  |  |  |  |  |  |

**Table 2: Modes and Subsystem activity**



**Figure 11: Satellite Model illustration**

## 5.4 GROUND STATION

The Ground station being used for the Nano Satellite 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 Nano Satellit Emission and future missions like Bayern SAT. The latitude and the longitude were referred from the Google map as 49.0108N and 11.081E respectively.



**Figure 12: LRT Ground station, Muenchen**

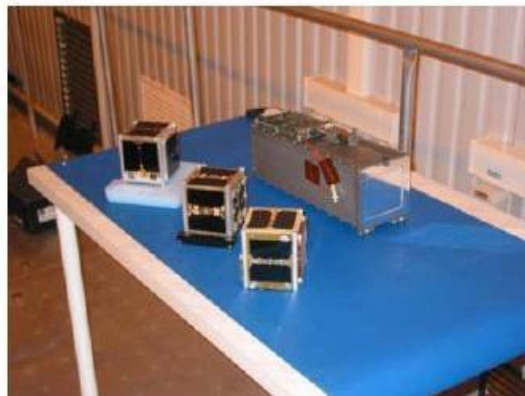
## **5.5 LAUNCHER**

The Nano 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 850 km. The Nano satellite will be flying at the altitude of 625 km and inclination of  $97.871^\circ$ .



**Figure 13: Indian Launcher PSLV (BharatRakshak ISRO)**

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



**Figure 14: PPODs**

## 5.6 ENVIRONMENTAL MODEL

### 5.6.1 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.

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

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 restricted to low earth orbits, particularly for the Nano Satellite 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 Nano Satellite orbit, the orbital radius is calculated to be 7003 km. For an altitude of 625 km, the atmospheric density is calculated to be  $1.0957 \times 10^{-14} \text{ kg/m}^3$ .

### 5.6.2 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

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

Where B is the local Magnetic field intensity,  $\lambda$  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  $7.96 \times 10^{-5} \text{ tesla}$ . 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.

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

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 Bmax is calculated using the below formulae, with the same assumption.

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

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

## 5.7 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 Nano Satellite modelling these are the frequently used constants.

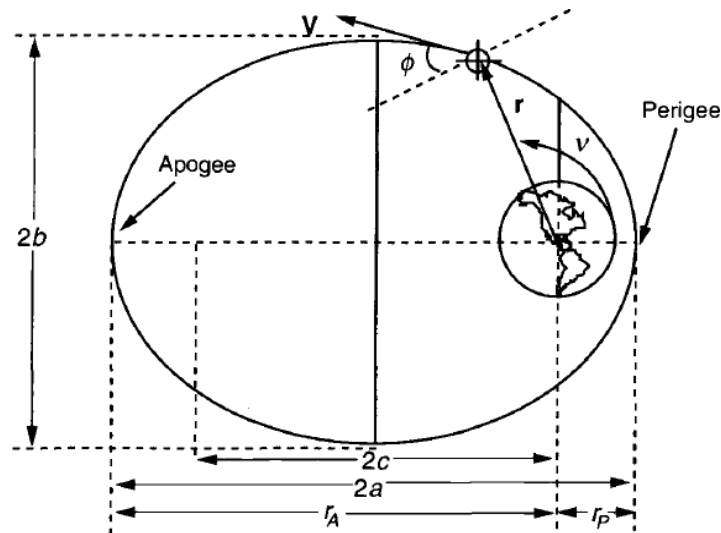
| Parameter                              |            |                     |
|----------------------------------------|------------|---------------------|
| Property                               | Value      | Dimension           |
| Mean Solar Intensity at Earth Distance | 1,3670E+03 | [W/m <sup>2</sup> ] |
| 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 <sup>3</sup> /s <sup>2</sup> ]  |
| Standard Acceleration of Free Fall on Earth | 9,8067E+00 | [m / s <sup>2</sup> ]              |
| Stefan Boltzmann Constant                   | 5,6704E-08 | [W/m <sup>2</sup> K <sup>4</sup> ] |

**Table 3: Physical Constants**

## 5.8 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.



**Figure 15:Orbital Elements**

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  $97.871^\circ$  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.

| 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 |

**Table 4: Orbital calculation**

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.

## 5.9 ELECTRICAL SYSTEM DIMENSIONING

The electrical power system consists of three subsystems, power generator, power storage and power distribution unit. 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.

## 5.10 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 Nano 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.

| 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 |
| Efficiency Electrical Path Battery 2 Loads | 90%     |

**Table 5 Input parameters for dimensioning of battery**

The chiefly used battery package for EDUSAT Missionare 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.

| 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       |

**Table 6: Battery dimensioning calculation results**

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.

### 5.10.1 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 Nano Satellite, where the power consumption of the Nano Satellite 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.

| Input                                   | Values   |
|-----------------------------------------|----------|
| Mean Power Need Eclipse                 | 1.28 W   |
| Mean Power Need Sunlight                | 2.41 W   |
| 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 |
| Efficiency Electrical Path SA 2 Battery | 260%     |

|                                       |     |
|---------------------------------------|-----|
| Loads                                 |     |
| Efficiency Electrical Path SA 2 Loads | 80% |

**Table 7: Solar cell Input parameter**

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

| 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    |

**Table 8: Solar cell dimensioning results**

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.

### 5.10.2 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 1134 bps and the required minimum volatile mass memory will be 5MB.

### 5.10.3 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 di- pole (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  $5^{\circ}$  to  $\pm 15^{\circ}\text{C}$ . 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.

#### **5.10.4 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.

| 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     |

**Table 9: Uplink Dimensioning**

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.

#### 5.10.5 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 |
|---|---------------------------------------|------|

**Table 10: Downlink dimensioning**

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 PEIR, spec is calculated to be 0.017W and the specified minimum transmitter power to be 0.17W.

#### **5.10.6 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.

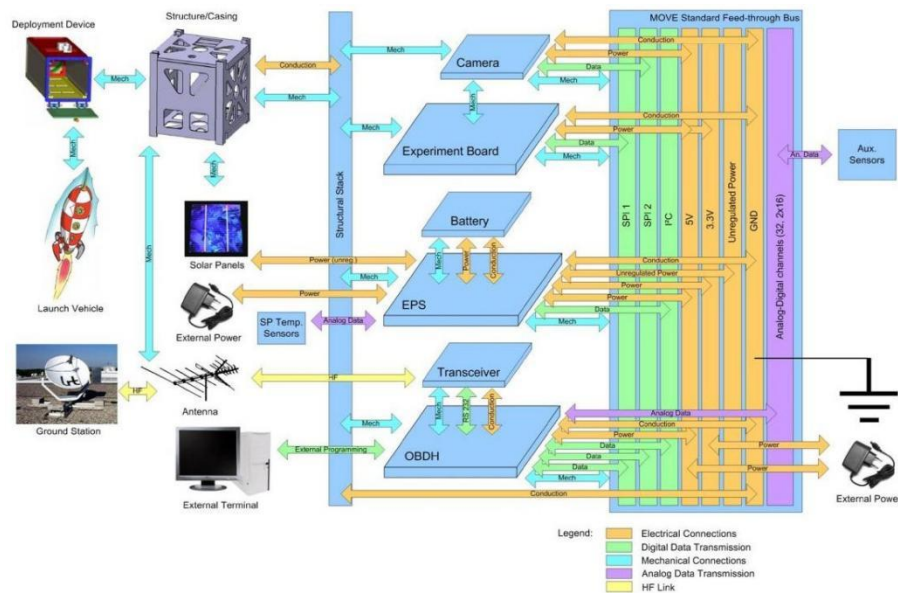
The orbit radius  $R_{apo}$  [km], taken from the class Orbit, the Magnetic Field Model calculates the minimum Earth's magnetic field strength  $B_{min}$  in tesla. Therefore the class Passive Magnetic Control Dimensioning needs to be connected with  $B_{min}$  of class Magnetic Field Model. The external torques from the object Nano 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

$$T_{mag,max} = D_{mag} \cdot B_{max}$$

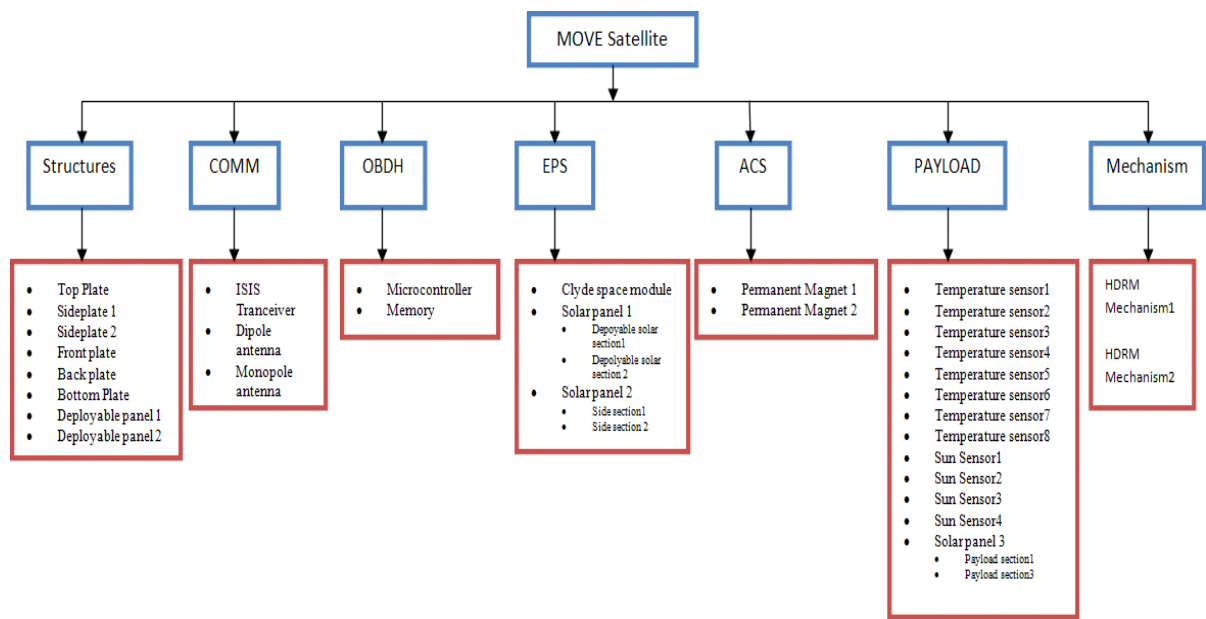
Because it is always the aim in this work to get the biggest impact of a certain effect, the maximum torque  $T_{mag,max}$  is calculated. This consideration requires the maximum Earth magnetic field strength  $B_{max}$  in the satellite orbit. The initial values to dimension the Magnets were, the force 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  $0.023 \text{ Am}^2$ .

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 Nano Satellitesatellite.



**Figure 16 :Product functional layout**



**Figure 17: EPS Module Block Diagram Layout**

## **CHAPTER 6**

### **NANOSATELLITE DESIGN**

#### **6.1 DESIGN CONSTRAINTS**

An airborne satellite must comply with hard constraints related to the severe space environment and the inability to repair the system in case of failure. Therefore, the design and the assembly of the device must abide by tighter rules than usual good and safe design criteria applied for any electronic system. This is particularly true and require the adoption of design techniques able to guarantee system operation even in the presence of limited faults at the device level. Other specific characteristics of a space application, although not directly related to failures of the system, further constrain the possible design solutions. These include the need to autonomously produce power, the limited visibility of the satellite from a ground station and the distance from it, the length of the mission, and so on.

##### **6.1.1 RADIATIONS**

The planned orbit (polar, at 600km of altitude) is below the Van Allen belts, albeit close to them. At this height, only a limited amount of radiation is present, compared to the belts and the outer space. This radiation might be in the form of high energy particles (protons, neutrons, alpha and beta particles) or ionizing electromagnetic rays from ultraviolet to X-rays, that lead to a Total Ionizing Dose (TID) of approximately 1krad/year. Due to the low orbit, and the short assumed lifetime of the mission (3 months), total dose effects have not been considered. However, singleevent effects (SEE) such as latch-up occurring in CMOS devices, and state upsets in memories and/or registers of digital. Temperature simulation of electronic boards circuits, might indeed

induce wrong behaviors or even permanent faults. Thus the satellite circuits have been protected at the logical and system level against these events.

### **6.1.2 TEMPERATURE RANGES**

While the satellite cycles through its orbit, it alternates from broad daylight to being deep in Earth shadow. In these conditions, temperature may vary considerably. However, the orbital period is fast enough not to allow too much heat to build up or be released into space, preventing overheating or frosting of the satellite. Thermal simulations allowed us to predict the actual temperature ranges for the outside and the inside faces of the aluminum plates that constitute the external structure of the satellite, and for the internal electronic boards. The predicted outside temperature range with active electronics is  $[5, +50]$  °C; the parts subject to this range are external ones, such as solar panels and antennas. The temperature range inside the satellite is  $[+20, +70]$  °C where the different curves represent the temperature of each board; all electronic circuits must comply with this range, which is compatible with standard commercial devices.

### **6.1.3 VACCUM**

Vacuum is not a problem for sealed electronic components, but reduces the power dissipation capability due to missing convection, leaving only conduction and radiation to the outside. This problem is related to the temperature ranges outlined above. The board that dissipates more heat is the one responsible of data transmission, as it hosts the power amplifiers; we successfully tested it in a thermal vacuum chamber, with a temperature range of  $[-20, +50]$  °C (we not could reach the predicted maximum temperature of  $+70$ °C due to limitations of the test equipment) and a pressure of 10Pa.

### **6.1.4 VIBRATIONS**

Forces and vibrations applied to the satellite during the launch are very high, and might cause physical damages, as well as disconnections of electronic devices and disengage of electrical connectors based on mechanical contacts. A careful choice of packages (i.e., no BGA devices were allowed in the design, since they are more sensitive to vibrations), mounting technologies, and overall structure is therefore mandatory. The satellite itself is fixed to the cylindrical launcher adapter with two machined tips on the bottom plate, which are released to place the satellite into orbit. PCBs have small size (about  $12 \times 8 \text{ cm}^2$ ), and are mechanically blocked at the four edges, therefore vibrations are kept within acceptable limits. More bulky components are secured to the PCB. A more critical point is represented by connectors. Direct board-to-board connectors are kept in place by the mechanical fixture of boards.

## **6.2 DESIGN SOLUTIONS**

Most of the efforts for components in a satellite are aimed at protecting the circuits from fatal events. Techniques to achieve this goal can be classified as either physical or logical. The former includes shielding the sensitive parts and choosing devices that are less prone to errors due to radiation at a comparable price tag. The latter, while allowing events to take place, mitigates or completely eliminates their effects by acting at the system level. Examples of such techniques include error correction (i.e., in memories), redundancy at several abstraction levels, and watchdog timers to reset misbehaving devices or boards. We applied numerous such techniques in the design of the satellite as described in the following.

### **6.2.1 SINGLE EVENT LATCH-UP (SEL)**

Single event latch-up or simply latch-up (LU), occurs when a parasitic SCR made by the couple of complementary MOS devices is turned on by high input voltages (this is the usual LU in ICs, caused for instance by input over-voltages) or by high energy particles which induce a small current (this is the case for a space device). The effect is a high, self-sustaining current flow, which can bring a high power dissipation and, in turn, device disruption.

### 6.2.2 SINGLE EVENT UPSET (SEU)

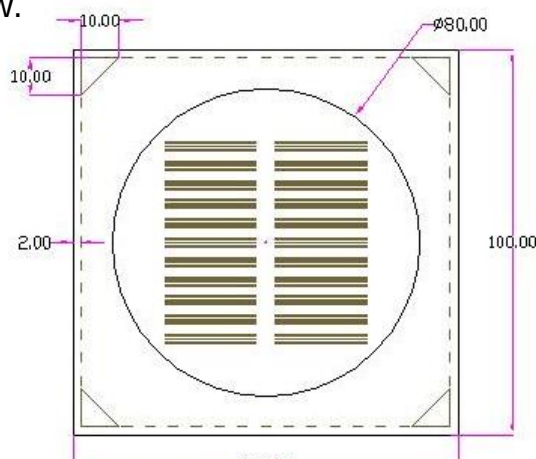
Picpot contains several digital circuits, including 5 processors, different kind of memories and programmable logic Devices. When a high-energy particle hits a circuit, it may Cause a transient change in voltage levels. While this is Usually not considered a problem with analog circuits, it might Adversely affect digital circuits which typically involve high Speed signals with steep edges, and especially memories that Rely on tiny voltages to carry their information. If the final Effect results only in a glitch (single event transient, set), Then it can safely be ignored; however, if the event is latched, Or directly upsets a bit (or multiple bits) in a memory or A register, it will probably lead to incorrect behaviors (soft Errors). In extreme cases, such as when a configuration bit of A programmable logic device turns an input into an output, it Can even cause severe damages.

### 6.2.3 SHIELDING

In a satellite two kinds of problems have to be mitigated: radio-frequency interferences and high-energy particles. Both can be reduced using appropriate shielding techniques. The outer structure is based on six aluminum alloy faces, electrically connected together, using screws which are less than  $\lambda/4$  apart for the highest frequency used on PiCPoT (2.4GHz). On the other hand we also protect the wires connecting internal and external parts.

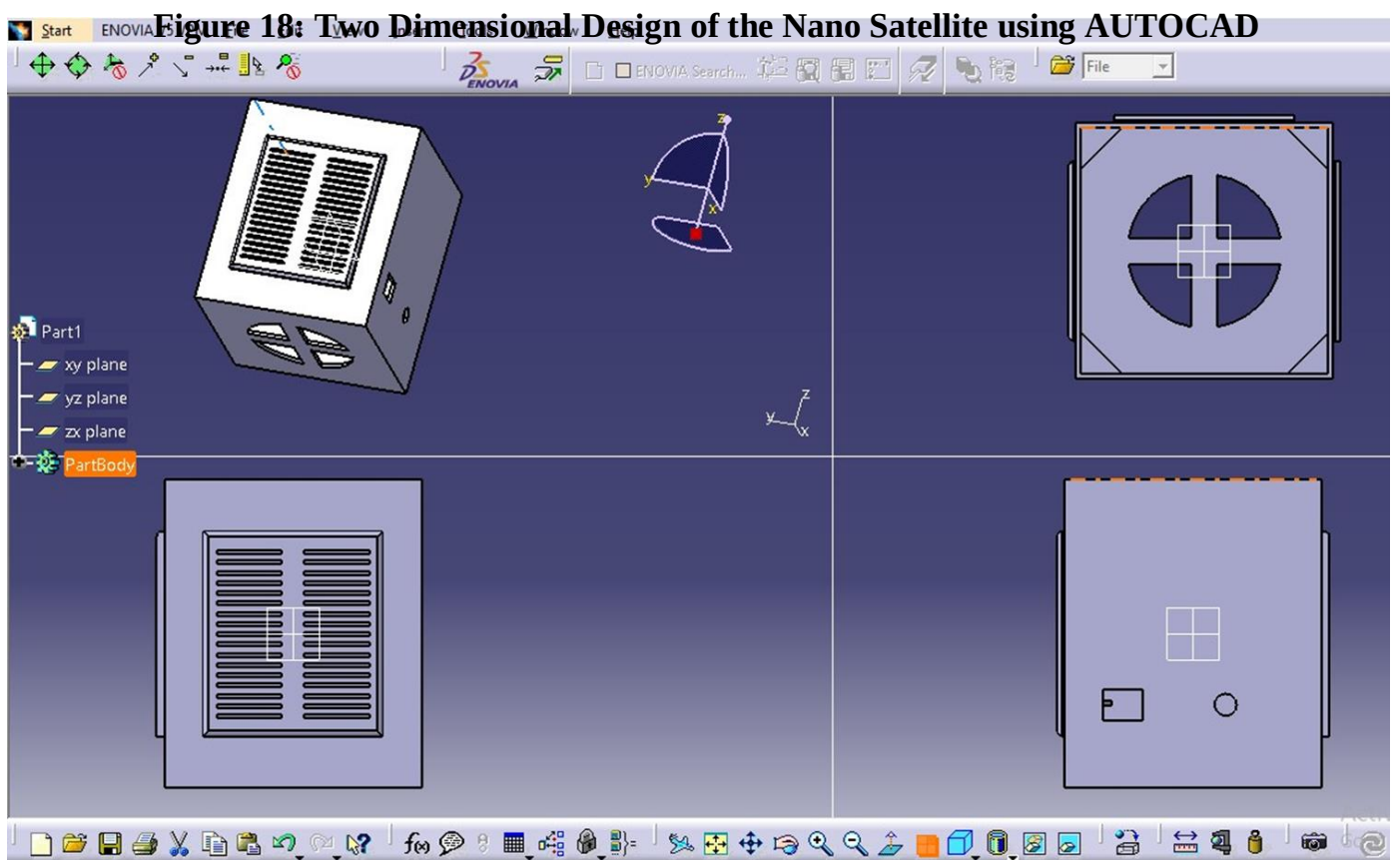
### 6.3 PROPOSED DESIGN

Considering the constraints and solutions a final design is created using Modelling software AUTOCAD and CATIA v5 as shown in the figures below.



## NANO SATELLITE

All Dimensions are in mm



**Figure 19: Three Dimensional Design of the Nano Satellite using CATIA**

## **CHAPTER 7**

### **MATERIAL SELECTION**

#### **7.1 MATERIALS USED**

Materials used in the structure must feature the same coefficient of thermal expansion as the deployer to prevent jamming. Specifically, allowed materials are four aluminum alloys: 7075, 6061, 5005, and 5052. Aluminum used on the structure which contacts the P-POD must be anodized to prevent

cold welding, and other materials may be used for the structure if a waiver is obtained. Beyond cold welding, further consideration is put into material selection as not all materials can be used in vacuums. Structures often feature soft dampers at each end, typically made of rubber, to lessen the effects of impacting other CubeSats in the P-POD.

#### **7.1.1 ALUMINIUM 6061**

6061 (UNS designation A96061) is a precipitation-hardened aluminum alloy, containing magnesium and silicon as its major alloying elements. Originally called "Alloy 61S", it was developed in 1935.[2] It has good mechanical properties, exhibits good weldability, and is very commonly extruded (second in popularity only to 6063). It is one of the most common alloys of aluminum for general-purpose use. It is commonly available in pre-tempered grades such as 6061-O (annealed), tempered grades such as 6061-T6 (solutionized and artificially aged) and 6061-T651 (solutionized, stress-relieved stretched and artificially aged).

#### **7.1.2 ALUMINIUM 7075**

7075 aluminium alloy (AA7075) is an aluminium alloy, with zinc as the primary alloying element. It has excellent mechanical properties, and exhibits good ductility, high strength, toughness and good resistance to fatigue. It is more susceptible to embrittlement than many other aluminium alloys because of microsegregation, but has significantly better corrosion resistance than the 2000 alloys. It is one of the most commonly used aluminium alloy for highly stressed structural applications, and has been extensively utilized in aircraft structural parts.

7075 aluminium alloy's composition roughly includes 5.6–6.1% zinc, 2.1–2.5% magnesium, 1.2–1.6% copper, and less than a half percent of silicon, iron, manganese, titanium, chromium, and other metals. It is produced in many tempers, some of which are 7075-O, 7075-T6, 7075-T651. The first 7075 was developed in secret by a Japanese company, Sumitomo Metal, in 193 but introduced by Alcoa in 1943 and was standardized for aerospace use

in 1945.7075 was eventually used for airframe production in the Imperial Japanese Navy.

### **7.1.3 HTM143/M55J (6K) CYANATE-ESTER**

Cyanate esters are chemical substances in which the hydrogen atom of the phenolic OH group is substituted by a cyanide group. The resulting product with an -OCN group is named a cyanate ester. Cyanate esters based on a bisphenol or novolac derivative are used in the production of resins.

The chemistry of the cure reaction is a trimerization of three CN groups to a triazine ring. When the monomer contains two cyanate groups the resulting structure is a 3D polymer network. Thermoset polymer matrix properties can be fine tuned by the choice of substituents in the bisphenolic compound. Bisphenol A and novolac based cyanate esters are the major products; bisphenol F and bisphenol E are also used. The aromatic ring of the bisphenol can be substituted with an allylic group for improved toughness of the material. Cyanate esters can also be mixed with bismaleimides to form BT-resins or with epoxy resins to optimize the end use properties.

### **7.1.4 UNIDIRECTIONAL CARBON FIBER FABRICS**

Unidirectional carbon fiber fabric is a type of carbon reinforcement that is non-woven and features all fibers running in a single, parallel direction. With this style of fabric, there are no gaps between fibers, and those fibers lay flat. There is no cross-section weave that divides the fiber strength in half with another direction. This allows for the concentrated density of fibers that provide maximum longitudinal tensile potential—greater than any other weave of fabric. For comparison, this is 3 times the longitudinal tensile strength of structural steel at one-fifth of the weight density.

### 7.1.5 ACRYLIC PLATES

Acrylic is a transparent plastic material with outstanding strength, stiffness, and optical clarity. Acrylic sheet is easy to fabricate, bonds well with adhesives and solvents, and is easy to thermoform. It has superior weathering properties compared to many other transparent plastics.

Acrylic sheet exhibits glass-like qualities—clarity, brilliance, and transparency—but at half the weight and many times the impact resistance of glass. From durable signs and skylights, to eye-catching retail store fixtures, displays and shelves, acrylic plastics provide outstanding versatility, durability, and aesthetic qualities.

|                                   |                                         |                                             |                                                                            |
|-----------------------------------|-----------------------------------------|---------------------------------------------|----------------------------------------------------------------------------|
| Density                           | 1.18 - 1.19 g/cc                        | 0.0426 -<br>0.0430 lb/in <sup>3</sup>       | Average value:<br>1.19 g/cc Grade<br>Count:29                              |
| Water Absorption                  | 0.130 - 0.800 %                         | 0.130 - 0.800 %                             | Average value:<br>0.257 % Grade<br>Count:25                                |
| Water Absorption<br>at Saturation | 0.650 - 2.60 %                          | 0.650 - 2.60 %                              | Average value:<br>1.40 % Grade<br>Count:16                                 |
| Moisture<br>Expansion             | 0.500 %                                 | 0.500 %                                     | Average value:<br>0.500 % Grade<br>Count:3                                 |
| Moisture Vapor<br>Transmission    | 55.0 cc-mm/m <sup>2</sup> -<br>24hr-atm | 140 cc-mil/100<br>in <sup>2</sup> -24hr-atm | Average value:<br>55.0 cc-mm/m <sup>2</sup> -<br>24hr-atm Grade<br>Count:3 |

**Table 11: Mechanical Properties**

## **CHAPTER 8**

### **FABRICATION PROCESS**

#### **8.1 CNC MACHINING**

AC Manufacturing is a complete 5 axis cnc machine shop capable of machining parts to customer specification (conventional and 5 axis machining). We can machine to print and accept the most common file formats including: .dwg .dxf .iges .step .sat as well as .pdf files. We can accept both small and large run jobs of custom machined parts.

#### **8.2 LASER CUTTING PROCESS**

A laser cutting machine is a device that uses a beam of radiation to cut, fuse, connect, and evaporate materials. It's a technology centered around high speed and precision. A typical laser cutter connects a computer and works in the X and Y axes. (In other words, it's a bidimensional machine.) The computer controls the laser (power, frequency, speed). Other aspects — like table setup, lenses and focus, exhaust, gas, and material preparation — must be done manually. All these variables can influence the quality of the laser cut.

With the acrylic sheet in place, the laser will vaporize the solid material, creating a thin, smooth, and glossy surface with extreme precision.



**Figure 20: Laser Cutting Process**

### **8.3 PRESS BRAKE FORMING**

This is a form of bending used to produce long, thin sheet metal parts. The machine that bends the metal is called a press brake. The lower part of the press contains a V-shaped groove called the die. The upper part of the press contains a punch that presses the sheet metal down into the v-shaped die, causing it to bend.[18] There are several techniques used, but the most common modern method is "air bending". Here, the die has a sharper angle than the required bend (typically  $85^{\circ}$  for a  $90^{\circ}$ ) and the upper tool is precisely controlled in its stroke to push the metal down the required amount to bend it through  $90^{\circ}$ . Typically, a general purpose machine has an available bending force of around 25 tonnes per metre of length. The opening width of the lower die is typically 8 to 10 times the thickness of the metal to be bent (for example, 5 mm material could be bent in a 40 mm die). The inner radius of the bend formed in the metal is determined not by the radius of the upper tool, but by the lower die width. Typically, the inner radius is equal to 1/6 of the V-width used in the forming process.

The press usually has some sort of back gauge to position depth of the bend along the workpiece. The backgauge can be computer controlled to allow the operator to make a series of bends in a component to a high degree of accuracy. Simple machines control only the backstop, more advanced machines control the position and angle of the stop, its height and the position of the two reference pegs used to locate the material. The machine can also record the exact position and pressure required for each bending operation to

allow the operator to achieve a perfect 90 degree bend across a variety of operations on the part.



**Figure 21: Press Brake Forming**

## **CHAPTER 9**

### **PAYLOAD**

#### **9.1 PAYLOAD**

Payload is the carrying capacity of an aircraft or launch vehicle, usually measured in terms of weight. Depending on the nature of the flight or mission, the payload of a vehicle may include cargo, passengers, flight crew, munitions, scientific instruments or experiments, or other equipment. Extra fuel, when optionally carried, is also considered part of the payload. In a commercial context (i.e., an airline or air freight carrier), payload may refer only to revenue-generating cargo or paying passengers.

For a rocket, the payload can be a satellite, space probe, or spacecraft carrying humans, animals, or cargo. For a ballistic missile, the payload is one or more warheads and related systems; their total weight is referred to as the throw-weight.

The fraction of payload to the total liftoff weight of the air or spacecraft is known as the "payload fraction". When the weight of the payload and fuel are considered together, it is known as the "useful load fraction". In spacecraft,

"mass fraction" is normally used, which is the ratio of payload to everything else, including the rocket structure.

## **9.2 ARDUINO UNO**

Arduino is an open-source electronics platform based on easy-to-use hardware and software. Arduino boards are able to read inputs - light on a sensor, a finger on a button, or a Twitter message - and turn it into an output - activating a motor, turning on an LED, publishing something online. You can tell your board what to do by sending a set of instructions to the microcontroller on the board. To do so you use the Arduino programming language (based on Wiring), and the Arduino Software (IDE), based on Processing.

### **9.2.1 SENSORS**

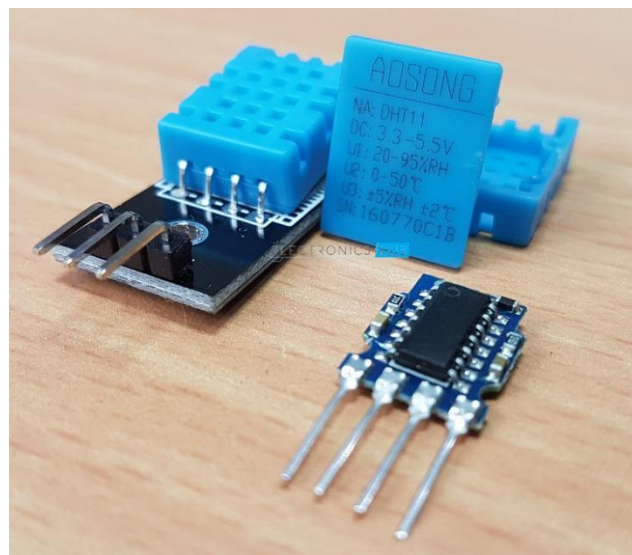
A sensor is a device that measures physical input from its environment and converts it into data that can be interpreted by either a human or a machine. Most sensors are electronic (the data is converted into electronic data), but some are more simple, such as a glass thermometer, which presents visual data.

Arduino based sensors are considered as payload for this Nano Satellite Module. This module is specifically designed for atmospheric parameters. The sensor modules which are used in this project are Temperature and Humidity Sensor, GPS and Bluetooth module.

### **9.2.2 TEMPERATURE & HUMIDITY SENSOR**

DHT11 is a Humidity and Temperature Sensor, which generates calibrated digital output. DHT11 can be interface with any microcontroller like Arduino, Raspberry Pi, etc. and get instantaneous results. DHT11 is a low cost humidity and temperature sensor which provides high reliability and long term stability.

The DHT11 Humidity and Temperature Sensor consists of 3 main components. A resistive type humidity sensor, an NTC (negative temperature coefficient) thermistor (to measure the temperature) and an 8-bit microcontroller, which converts the analog signals from both the sensors and sends out single digital signal.



**Figure 22: DHT11 Sensor**

### **9.2.3 GPS MODULE**

GPS sensors are receivers with antennas that use a satellite-based navigation system with a network of 24 satellites in orbit around the earth to provide position, velocity, and timing information.



### **Figure 23: GPS Module**

To determine distance, both the satellite and GPS receiver generate the same pseudocode signal at the same time. The satellite transmits the pseudocode; which is received by the GPS receiver. These two signals are compared and the difference between the signals is the travel time. Now, if the receiver knows the distance from 3 or more satellites and their location (which is sent by the satellites), then it can calculate its location by using Trilateration method.

#### **9.2.4 BLUETOOTH MODULE**

The HC-05 has two operating modes, one is the Data mode in which it can send and receive data from other Bluetooth devices and the other is the AT Command mode where the default device settings can be changed. We can operate the device in either of these two modes by using the key pin as explained in the pin description.

It is very easy to pair the HC-05 module with microcontrollers because it operates using the Serial Port Protocol (SPP). Simply power the module with +5V and connect the Rx pin of the module to the Tx of MCU and Tx pin of module to Rx of MCU as shown in the figure below



**Figure 24: Bluetooth Module**

### **9.2.5 SOLAR PANELS**

The term solar panel is used colloquially for a photo-voltaic (PV) module. A PV module is an assembly of photo-voltaic cells mounted in a framework for installation. Photo-voltaic cells use sunlight as a source of energy and generate direct current electricity. A collection of PV modules is called a PV Panel, and a system of Panels is an Array. Arrays of a photovoltaic system supply solar electricity to electrical equipment. The most common application of solar energy collection outside agriculture is solar water heating systems.



**Figure 25: Solar Panel**

Each module is rated by its DC output power under standard test conditions (STC). Power typically ranges from 100 to 365 Watts (W). The efficiency of a module determines the area of a module given the same rated output – an 8% efficient 230 W module will have twice the area of a 16% efficient 230 W module. Some commercially available solar modules exceed 24% efficiency.

### 9.2.6 ASSEMBLY

The complete satellite parts mentioned in the above chapters have been fabricated and purchased. Then the acrylic plates have been assembled in the form of a 1U Nanosat. The sensors have been placed inside this 1U module and the connections have been made with Arduino Development Board. The entire module will look in the below format.



**Figure 26: Assembled Nano Satellite View 1**

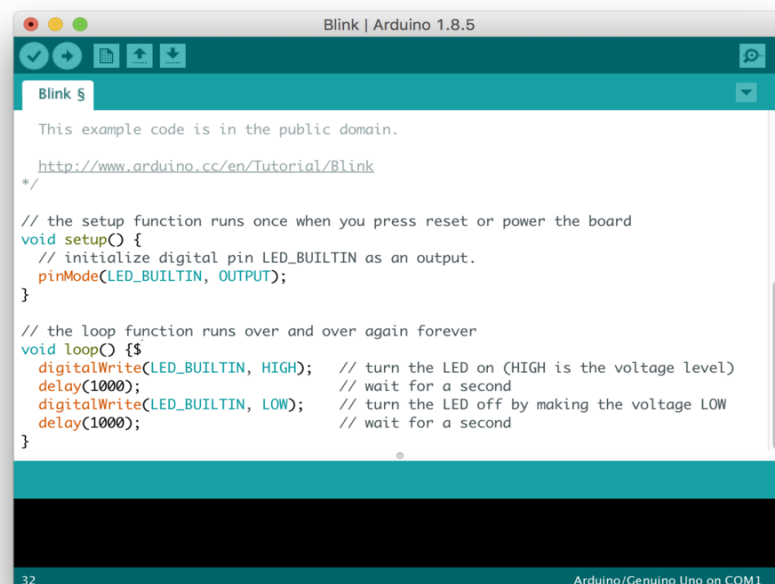


**Figure 27: Assembled Nano SatelliteView 2**

### 9.2.7 INTEGRATED CODING

The Arduino Integrated Development Environment (IDE) is a cross-platform application (for Windows, macOS, Linux) that is written in functions from C and C++.[2] It is used to write and upload programs to Arduino compatible boards.

The source code for the IDE is released under the GNU General Public License, version 2.[4] The Arduino IDE supports the languages C and C++ using special rules of code structuring.[5] The Arduino IDE supplies a software library from the Wiring project, which provides many common input and output procedures. User-written code only requires two basic functions, for starting the sketch and the main program loop, that are compiled and linked with a program stub `main()` into an executable cyclic executive program with the GNU toolchain, also included with the IDE distribution.



**Figure 28: Arduino IDE Screen**

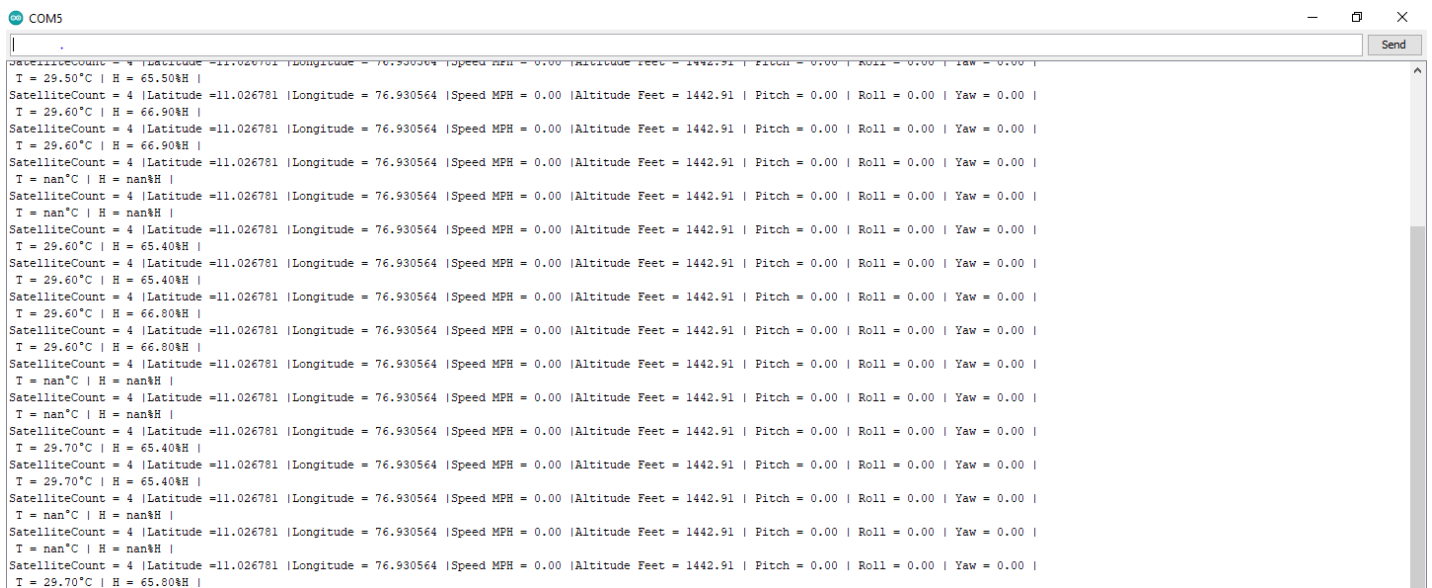
## CHAPTER 10

## RESULTS

### 10.1 SERIAL MONITOR

The Arduino IDE has a feature that can be a great help in debugging sketches or controlling Arduino from your computer's keyboard. The Serial Monitor is a separate pop-up window that acts as a separate terminal that communicates by receiving and sending Serial Data.

The output screen for the satellite module is shown below



```
COM5
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.50°C | H = 65.50%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.60°C | H = 66.90%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.60°C | H = 66.90%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = nan°C | H = nan%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = nan°C | H = nan%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.60°C | H = 65.40%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.60°C | H = 65.40%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.60°C | H = 66.80%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.60°C | H = 66.80%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = nan°C | H = nan%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = nan°C | H = nan%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.70°C | H = 65.40%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.70°C | H = 65.40%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = nan°C | H = nan%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = nan°C | H = nan%RH |
SatelliteCount = 4 | Latitude = 11.026781 | Longitude = 76.930564 | Speed MPH = 0.00 | Altitude Feet = 1442.91 | Pitch = 0.00 | Roll = 0.00 | Yaw = 0.00 |
T = 29.70°C | H = 65.80%RH |
```

Figure 29: Arduino IDE Output Screen

## 10.2 SERIAL BLUETOOTH TERMINAL

Serial Bluetooth Terminal' is a line-oriented terminal / console app for microcontrollers, arduinos and other devices with a serial / UART interface connected with a bluetooth to serial converter to your android device.

The output screen for the satellite module is shown below

```
Yaw = 0.00 |  
19:45:48.422 T = 27.80°C | H = 69.70%H |  
19:45:49.601 SatelliteCount = 7 |Latitude =11.026851 |  
Longitude = 76.930526 |Speed MPH = 0.00 |  
Altitude Feet = 1475.07 | Pitch = 0.00 | Roll = 0.00 |  
Yaw = 0.00 |  
19:45:49.716 T = 27.80°C | H = 69.70%H |  
19:45:50.533 SatelliteCount = 7 |Latitude =11.026851 |  
Longitude = 76.930526 |Speed MPH = 0.00 |  
Altitude Feet = 1475.07 | Pitch = 0.00 | Roll = 0.00 |  
Yaw = 0.00 |  
19:45:50.695 T = 27.90°C | H = 70.80%H |  
19:45:51.669 SatelliteCount = 7 |Latitude =11.026851 |  
Longitude = 76.930526 |Speed MPH = 0.00 |  
Altitude Feet = 1475.07 | Pitch = 0.00 | Roll = 0.00 |  
Yaw = 0.00 |  
19:45:51.833 T = 27.90°C | H = 70.80%H |  
19:45:52.869 SatelliteCount = 7 |Latitude =11.026851 |  
Longitude = 76.930526 |Speed MPH = 0.00 |  
Altitude Feet = 1475.07 | Pitch = 0.00 | Roll = 0.00 |  
Yaw = 0.00 |  
19:45:53.032 T = 27.90°C | H = 70.80%H |  
19:45:54.279 SatelliteCount = 7 |Latitude =11.026851 |  
Longitude = 76.930526 |Speed MPH = 0.00 |  
Altitude Feet = 1475.07 | Pitch = 0.00 | Roll = 0.00 |  
Yaw = 0.00 |  
19:45:54.438 T = 27.90°C | H = 70.80%H |  
19:45:55.260 SatelliteCount = 7 |Latitude =11.026851 |  
Longitude = 76.930526 |Speed MPH = 0.00 |  
Altitude Feet = 1475.07 | Pitch = 0.00 | Roll = 0.00 |  
Yaw = 0.00 |  
19:45:55.425
```

M1

M2

M3

M4

M5

M6

M7



**Figure 30: Serial Bluetooth Terminal Screen**

## CHAPTER 11

### RESULT CONCLUSIONS & SCOPE FOR FUTURE WORK

| S.No | Temperature | Distance | Humidity | SAT<br>COUNT | Altitude | Rodt.         | yaw        | Pitch          |
|------|-------------|----------|----------|--------------|----------|---------------|------------|----------------|
| 1    | 31:00°C 87  | 357      | 58.70    | 8            |          | value<br>357. | value 2.91 | value<br>318.3 |
| 2    | 31.00°C 87  | 355      | 58.70    | 8            |          | 357-37        | 2.94       | 321.39         |
| 3    | 0.000 32.0  | 357      | 0.00     | 8            |          | 357.58        | 3.03       | 318.83         |
| 4    | 31.00°C 87  | 352      | 58.70    | 8            |          | 357.35        | 3.03       | 318.83         |
| 5    | 31.00° 87.  | 354      | 58.70    | 8            |          | 357.31        | 2.89       | 317-01         |

The aim is to design and implement a NanoSat based weather monitoring system. The system must be simple to construct, portable, cost efficient, less power consuming and reliable. The hardware design and the data acquisition system has been demonstrated successfully.

And the fabrication of Nano Satellite with proper material Identification is done. Also Humidity temperature against variation of altitude in a region is measured in given interval of time with the help of satellite. The comparison of

pressure, humidity and temperature with respect to the variation of altitude is tabulated. The future scope of the project is to develop propulsion unit for this specific nano satellite module which will utilize lipo battery power subsystem for the same.

## **ANNEXURE I**

The coding for the integrated payload circuit is as follows

```
#include <SimpleTimer.h>

#include<Wire.h>

#include <SPI.h>

#include<Ultrasonic.h>

#define BLYNK_PRINT Serial

#include <Blynk.h>

#include <BlynkSimpleEsp32.h>

#include "DHT.h"

#define DHTTYPE 22

#define DHTPIN 4

SimpleTimer timer;
```

```

#include <TinyGPS++.h>

#include <HardwareSerial.h>

#include <WiFi.h>

#include<SH1106.h>

char auth[] = "g87DvpTsHlvPwgj3pgYIEv0Xkkv2Xu_N";

char ssid[] = "Karthik";

char pass[] = "12345678";

const int MPU_addr=0x68;

int16_t AcX,AcY,AcZ,Tmp,GyX,GyY,GyZ;

float latitude , longitude;

String lat_str , lng_str;

int minVal=265;

int maxVal=402;

double x;

double y;

double z;

DHT dht(DHTPIN, DHTTYPE);

SH1106 display(0x3c, 21, 22);

WiFiClient client;

TinyGPSPlus gps;

```

```

HardwareSerial SerialGPS(1);

Ultrasonic ultrasonic(18,19);

int d;

void setup(){

Wire.begin();

Wire.beginTransmission(MPU_addr);

Wire.write(0x6B);

Wire.write(0);

Wire.endTransmission(true);

Serial.begin(115200);

Serial.println("Connecting to ");

Serial.println(ssid);

WiFi.begin(ssid, pass);

while (WiFi.status() != WL_CONNECTED)

{

delay(500);

Serial.print(".");          // print ... till not connected

}

Serial.println("");

Serial.println("WiFi connected");

```

```

display.init();

display.flipScreenVertically();

display.setFont(ArialMT_Plain_10);

timer.setInterval(1000L, sendSensor);

Blynk.begin(auth, ssid, pass);

Blynk.virtualWrite(V0, "clr");

}

void sendSensor(){

d = ultrasonic.read();

Serial.println(d);

dht.begin();

Wire.beginTransmission(MPU_addr);

Wire.write(0x3B);

Wire.endTransmission(false);

Wire.requestFrom(MPU_addr,14,true);

AcX=Wire.read()<<8|Wire.read();

AcY=Wire.read()<<8|Wire.read();

AcZ=Wire.read()<<8|Wire.read();

int xAng = map(AcX,minVal,maxVal,-90,90);

int yAng = map(AcY,minVal,maxVal,-90,90);

```

```
int zAng = map(AcZ,minVal,maxVal,-90,90);
```

```
x= RAD_TO_DEG * (atan2(-yAng, -zAng)+PI);
```

```
y= RAD_TO_DEG * (atan2(-xAng, -zAng)+PI);
```

```
z= RAD_TO_DEG * (atan2(-yAng, -xAng)+PI);
```

```
Serial.print("AngleX= ");
```

```
Serial.println(x);
```

```
Serial.print("AngleY= ");
```

```
Serial.println(y);
```

```
Serial.print("AngleZ= ");
```

```
Serial.println(z);
```

```
float h = dht.readHumidity();
```

```
// Read temperature as Celsius (the default)
```

```
float t = dht.readTemperature();
```

```
// Read temperature as Fahrenheit (isFahrenheit = true)
```

```
float f = dht.readTemperature(true);
```

```
// Check if any reads failed and exit early (to try again).
```

```
if (isnan(h) || isnan(t) || isnan(f)) {
```

```
    Serial.println(F("Failed to read from sensor!"));
```

```
    return;
```

```
}
```

```
// Compute heat index in Fahrenheit (the default)
```

```
float hif = dht.computeHeatIndex(f, h);
```

```
// Compute heat index in Celsius (isFahreheit = false)
```

```
float hic = dht.computeHeatIndex(t, h, false);
```

```
Serial.print(F("Humidity: "));
```

```
Serial.print(h);
```

```
Serial.print(F("%    Temperature: "));
```

```
Serial.print(t);
```

```
Serial.print(F("°C / "));
```

```
Serial.print(f);
```

```
Serial.print(F("°F    Heat index: "));
```

```
Serial.print(hic);
```

```
Serial.print(F("°C / "));
```

```

Serial.print(hif);

Serial.println(F("°F"));

Serial.println("-----");

Blynk.virtualWrite(V2, x);

Blynk.virtualWrite(V3, y);

Blynk.virtualWrite(V4, z);

Blynk.virtualWrite(V5, h);

Blynk.virtualWrite(V6, t);

Blynk.virtualWrite(V1, d);

SerialGPS.begin(9600, SERIAL_8N1, 16, 17);

while (SerialGPS.available() > 0)
{
  gps.encode(SerialGPS.read());
}

if (gps.location.isValid())
{
  latitude = gps.location.lat();

  lat_str = String(latitude , 6);

  longitude = gps.location.lng();

```

```
Lng_str = String(longitude , 6);

Serial.print("Latitude = ");

Serial.println(lat_str);

Serial.print("Longitude = ");

Serial.println(Lng_str);

display.clear();

display.setTextAlignment(TEXT_ALIGN_LEFT);

display.setFont(ArialMT_Plain_16);

display.drawString(0, 23, "Lat:");

display.drawString(45, 23, lat_str);

display.drawString(0, 38, "Lng:");

display.drawString(45, 38, Lng_str);

int SatCount= gps.satellites.value();

int Speed = gps.speed.mph();

float Alt = gps.altitude.feet();

Serial.println("Satellite Count:");

Serial.print(SatCount);

Serial.print("    Orbital Speed:");

Serial.print(Speed);

Serial.print(" MPH");
```

```
    Serial.print("    Satellite Altitude:");

    Serial.print(Alt);

    Serial.print(" ft");

    Blynk.virtualWrite(V0, 1, latitude, longitude, "2U NANOSAT");

    Blynk.virtualWrite(V7, SatCount);

    Blynk.virtualWrite(V8, Speed);

    Blynk.virtualWrite(V9, Alt);

    display.display();

}

delay(1000);

Serial.println();

}

void loop()

{

    Blynk.run();

    timer.run();

}
```

## **REFERENCES**

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