DESIGNING OF SOLAR PANEL DEPLOYMENT MECHANISM FOR SMALL SATELLITE

Saqib Kabir

This dissertation is submitted in partial fulfilment of the requirements for

the degree of Master of Philosophy in Space Studies

March 2021

SL20-11M

DECLARATION

I know the meaning of plagiarism and declare that all the work in the document, save for that which is properly acknowledged, is my own. It has not been previously submitted, in part or whole, to any university of institution for any degree, diploma, or other qualification. This thesis/dissertation has been submitted to the Turnitin module (or equivalent similarity and originality checking software) and I confirm that my supervisor has seen my report and any concerns revealed by such have been resolved with my supervisor.

Signed by candidate

Signed________________________ Date : 29^{th} March, 2020

Saqib Kabir, MPhil Space Studies

University of Cape Town

ABSTRACT

In the last few years, there has been considerable progress in development of low-cost space missions. Spacecraft usually include in their systems several appendages that during launch are stowed in order to guarantee their survival from launch loads, and are required to be deployed once the launching phase has ended. These appendages may include solar panels having large dimensions, low mass to size ratio, with large inertia and relatively low structural rigidity, to produce power using photovoltaic principle.

In this dissertation a solar panel deployment mechanism using a spiral flat spring is presented.

ACKNOWLEDGEMENTS

I am thankful to God for my thesis work. This research has been of great benefit to me in learning the methodology used to design deployment mechanisms according to the methodology of Space Mission Analysis and Design (SMAD). The SMAD process has been very beneficial for me and now I have a firmer grasp on a design processes for the space industry.

I am also grateful to my supervisor Prof: Peter Martinez, whose guidance edified my goal and streamlined my research work. I felt great pleasure working under his guidance. His guidance allowed me to materialise this research.

Table of Figures

List of Tables

Abbreviation List

Term	Description
g	Gravitational Constant
2P	Two Point Failure
3P	Three Point Failure
AO	Atomic Oxygen
AOCS	Attitude and Orbital Control System
AIT	Assembly Integration and Testing
CAD	Computer Aided Design
COG	Centre of Gravity
CFRP	Carbon Fibre Reinforcement Polymer
CTE	Coefficient of Thermal Expansion
DOF	Degree of Freedom
EPS	Electrical Power System
FE	Finite Element
FEA	Finite Element Analysis
FEM	Finite Element Modeling
FoS	Factor of Safety
Hex	Hexahedral
MoS	Margin of Safety
MPC	MultiPoint Constraint
OBDH	On Board Data Handling
Quad	Quadrilateral
RMS	Root Mean Square
SMA	Shape Memory Alloys
TBC	To be Confirmed
TRL	Technology Readiness Level
Tet	Tetrahedral
TT&C	Telemetry and Tele Command
Rx	Rotation in X Axis
Ry	Rotation in Y Axis
Rz	Rotation in Z Axis
Tx	Translation in X Axis
Ty	Translation in Y Axis
Tz	Translation in Z Axis

TABLE OF CONTENTS

1 INTRODUCTION

1.1 Power System of a Satellite

A satellite is made up of several subsystems of different functions, which generally consist of two classes: platform system and payload system. The payload system is used to carry out specific missions directly. The platform system is used to maintain the satellite's normal operation, generally including structure system, thermal control system, electrical power system, attitude control system, orbit control system, TT&C system, OBDH system, etc. The platform system can be increased or decreased by the requirements of particular space missions. The electrical power system is necessary for each mission and is a principal part of any satellite. Therefore, every satellite must be equipped with a suitable, reliable electrical power system, to afford safe and reliable electrical power for various equipment of various satellite subsystems. Once the electrical power system stops supplying power the satellite will lose its functions instantly and become useless and will be "space debris".

1.1.1 Electrical Power System Concept and Functions

A satellite electrical power system is a subsystem that produces, stores, converts, regulates and distributes electrical energy, and it is called the EPS for short. Its basic function is to convert light energy, nuclear energy or chemical energy into electrical energy using certain physical change or chemical change, to store, to regulate, and to convert in accordance with request, and then to supply electrical energy to each satellite subsystem. It is just like a ground generating station, transformer substation, power distribution station, and power supply network

The EPS plays a decisive role in promoting satellite performance, prolonging satellite lifetime. So far, space technologies are coming into the new stage of exploiting and making use of low Earth orbit space massively. Various satellites directly serving civil economic and national defense, are developing toward the direction of high performance, many uses, and long operating lifetimes. As payloads are increasingly augmented and satellite lifetime is prolonged continuously, so higher and higher demands are brought forward for the performance parameters of the EPS's power, lifetime, reliability, performance mass ratio, performance-price ratio.

A satellite EPS is made up of electrical power source, power control unit, power convert unit, power distribution unit and harness net, etc, wherein the most important part is electrical power, and the other parts are collocated accordingly in terms of requirements for supply source and load.

Electrical power sources for spacecraft mainly include the following three types:

1.1.1.1 Chemical:

It consists of only a Battery which is useful only for small missions such as Sputnik 2. The battery is made up of Li Cell, Zinc-Silver Cell, and Hydrogen-Oxygen Fuel Cell, etc.

1.1.1.2 Nuclear:

Nuclear electrical power uses Radioisotope Temperature Difference Generator, Reactor Temperature Difference Generator and Heat Ion Reactor (Uranium, Plutonium), etc.

1.1.1.3 Solar:

A solar power system consists of Photovoltaic Solar Cells and the Battery. The Solar Array converts solar energy into electric energy during the spacecraft sunlight period, supplies electric energy to spacecraft and charges the Battery; while Battery supplies electric energy during the eclipse period. The presently adopted Solar Array is made up of a Silicon Solar Cell or Gallium Arsenide, and the Battery is either a Cd-Ni Battery or H2-Ni Battery.

Solar arrays have limited use for interplanetary missions as the intensity of solar radiation reduces at a far distance from the Sun. However, the most common and reliable source of power generation for Low Earth Orbit (LEO) and geostationary (GEO) orbits is Solar Arrays which use photovoltaic solar cells.

1.2 Solar Arrays

There are three main parts of the solar array, Structure, Deployment Mechanism and the Cells [1]. The most widely used Solar Array configuration system is of Photovoltaic cells, usually made up of Silicon, which converts the solar energy into electric energy. Solar Array structures can be designed to be either a body-mounted Solar Array, Flip-out single panel or a deployable multi-panel Solar Array. Over the last four decades, the power requirements for satellites have increased from 10s of watts to 75,000 watts (International Space Station) and so too have the sizes of solar array structures deployed in space [1].

1.2.1 Body Mounted Solar Arrays

The simplest type of Solar Array is a body-mounted band that was used in an earlier type of satellite like the U.S Vanguard I satellite [2], for which the power requirements were low. These types of arrays (solar cells) are normally bonded with the body of the satellite and these arrays are considered as an integral part of the satellite body. Due to the limited outer surface area available, the power generation for this type of configuration is low. To obtain continuous power these panels can be cylindrically mounted on the satellite's outer surface which continuously spins to face the sun and such satellites are termed as spin-stabilized. Figure (1) shows the GOCE satellite with its body-mounted solar arrays.

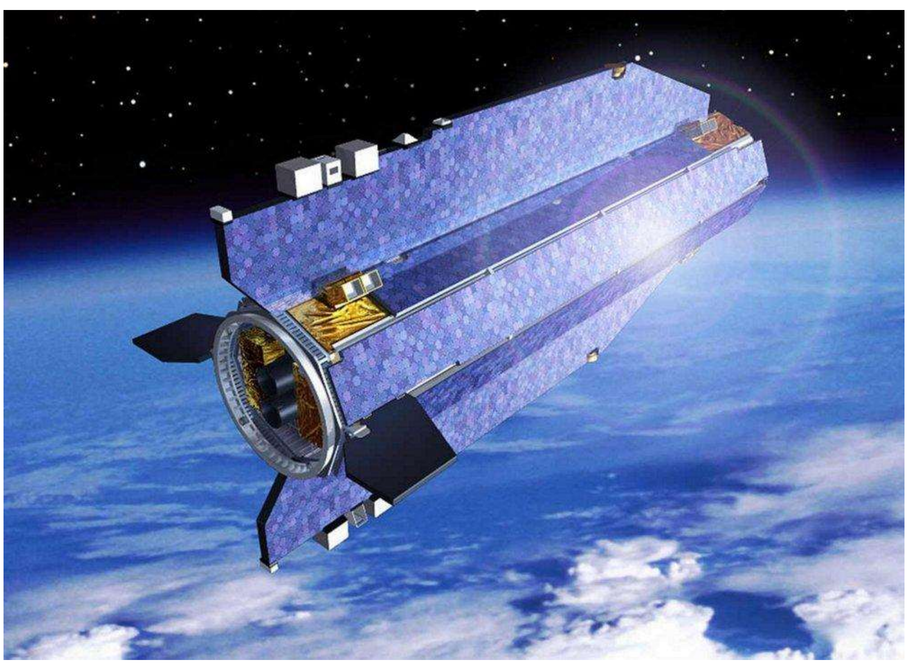
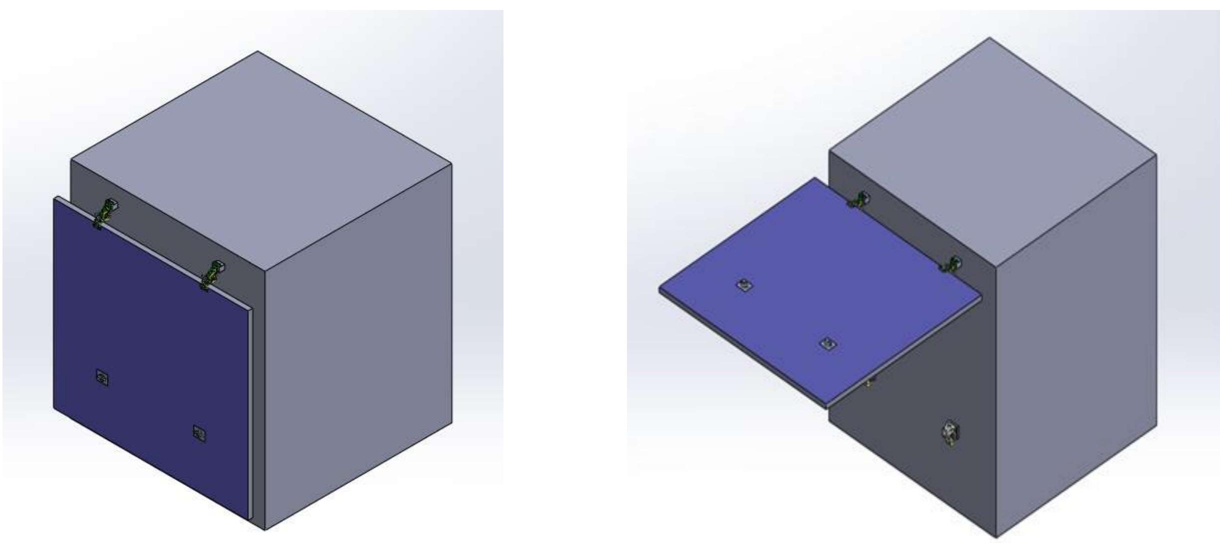


Figure 1: Body mounted Solar Array on the GOCE satellite.

1.2.2 Flip-Out Solar Array

This type of solar array is considered as a deployable solar array. The deployment of the array is performed in two steps. In the first step the panel is released from the satellite body and in the second stage it is deployed using mechanism. This configuration is adopted when the available area is less in comparison to the power requirements of the satellite. Panels after deploying can be sun pointed to obtain the maximum power, where for the body-mounted array the power will be generated from only that side of the satellite which is pointed towards the sun. These types of solar arrays are complex in design in comparison to body-mounted arrays; however they offer more radiative area to radiate heat of the electronic circuit housings. .



(a) (b)

Figure 2 Flip out Configuration (a) Stowed Position (b) Deployed Position.

1.2.3 Multi Panel Solar Array

The power requirements for satellite have significantly increased over time, which has led to the use of multi -panel solar arrays. These solar arrays use stiff Aluminum Honeycomb panels which use Aluminum as a face sheet for panels on which the solar cells are placed. These Solar panels are interconnected with each other using a spring mechanism and the power cable. The arrays are folded in stowed position with the satellite body for launch[3]. The use of a spring mechanism or hinges is the most sufficient and reliable method to deploy the panels from the stowed position. After launching in orbit the panels are deployed along the body of the satellite and continued to spread open until they are straight. These solar arrays can be designed in way to always be oriented towards the sun to generate the maximum solar power. This system is known as a Solar Array Drive Assembly (SADA) and it is usually used for GEO satellites. It has shown in Figure (3)[4].

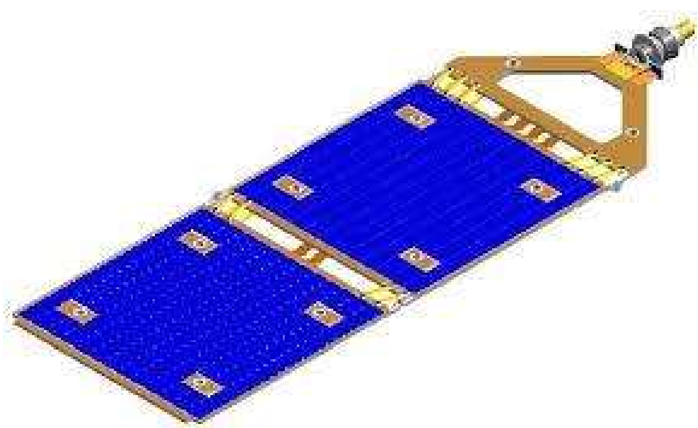


Figure 3 Multi Panel Solar Array.

In order to further reduce the mass of the rigid solar arrays, the Aluminium face sheet can be replaced with the composites. These types of solar arrays are used when the power consumption of satellite is higher in comparison to the available area for solar cells. Moreover, all six sides of the satellite cannot be used to mount solar cells, due to the requirement of some radiative area for the units.

1.3 Deployment Mechanisms

The deployment of solar panels is performed by elastic structures or by some sort of mechanism. In order to deploy any of the Solar Arrays, except body mounted, a deployment system is used [5]. It is consists of Spring, Hinge, Guide, Release mechanism, Latch/Lock mechanism and an Actuator. Failure of the deployment mechanism can lead to the failure of mission so these need to be more reliable in comparison to ground mechanism. Due to the difficulty of mechanical repairing of such mechanisms in space, they must be more reliable than ordinary mechanisms on the Earth. Hence the impact of high vibration load during launch, micro-gravity, high vacuum and power requirements should also need to be considered during the design [6]. There are two types of deployment mechanism; one is active while the other is passive. An active mechanism is normally uses an electrically operated actuator, while in a passive system spring is used to deploy the solar panels. The deployment mechanism needs to work throughout the life of satellite. If the power requirements are more and continuous, then the mechanism should be capable to track the solar arrays in the direction of sun to obtain the continuous power. In the following paragraph each the function of the deployment mechanism part is explained briefly.

1.3.1 Deployment Torque

The function of the spring is to generate the required pushing force or Torque to deploy the panel against the satellite body. There are several types of springs used commercially i.e. Tension, Leaf, Spiral, Compression, etc. However, the choice of spring is made based on the requirements.

As stated earlier, the solar arrays are in a stowed position before launching into a permitted small volume in the launcher. Later it is expanded in orbit to provide a larger area to expose to the sun. The preferred spring for the deployment mechanism is Spiral Torsional and the Compression spring. These springs are capable of reducing the overall volume in the form of stored energy by compressing or by torsional force thus making them most useful for deployment application, these springs are shown in Figure (4)[7].

(a) (b)

Figure 4: Spiral Torsion Spring (a) Compression Spring (b).

The spring-loaded mechanisms are light in weight and can be accommodated in a smaller volume. It does not require any power to activate as the energy can be stored in the form of its strain energy[8]. The movement of the spring is simple in comparison to Gears and Linkages. Generally, the deployment torque is generated using a spiral spring. The spiral spring is integrated with a hinge and it is carefully designed to provide torque which is sufficient to completely deploy the solar panels. The deployment torque of the hinge is proportional to the magnitude of the deployment shock; the higher the deployment torque of the spring, the higher will be the magnitude of the deployment shock.

Therefore in order to avoid a high magnitude deployment shock, the spring is designed to provide the minimum torque which is sufficient to completely deploy the solar arrays.

1.3.2 Hold and Release Mechanism

In order to keep the solar panels in a stowed position, the holding mechanism is provided to keep the panels along the satellite body. There are number of ways to release the solar panels for deployment. These include pyro cutters, latches driven by motor, paraffin (Nylon wire) and the shape memory alloys. The release mechanism should be capable of controlling the speed of deployment to avoid any oscillatory and impulsive forces on the satellite. The effects of these fluctuation forces is reduced by controlling the speed of deployment, either mechanically (most preferable) or electronically (using a stepping motor). In Figure 5, a Pyro Cable cutter [9] is shown.

Figure 5: Pyro Cable Cutter [6].

The Pyrotechnic device uses explosive charges carried by the device, which is initiated by electric current and creates a high speed cutting or shearing force which opens/closes the valve or cuts a wire thus releasing the holding mechanism. The explosive bolt also uses the same working principle. However non-explosive devices such as Shape Memory Alloys (SMA) and Nylon wire cutters can also be used as hold and release mechanisms.

1.3.3 Locking/Latching Mechanism

This is used to assure that the panels achieve the flatness after deployment to obtain the required area exposed to the Sun to generate the power. The latch should be resettable for ground testing [10] and should conform to the load requirements during launch, when the panels are in the stowed position. When the solar panels are completely deployed, it is necessary to keep the solar panels in deployed state. For this purpose a locking mechanism is necessary to keep the solar panels in deployed state. Generally the locking mechanism is integrated with the hinge in which a locking pin rotates and inserted in the slot at the position where locking is required. The locking pin is inserted inside the slot using a latch spring and the latch spring is designed to provide sufficient force to keep the locking pin inside the slot.

1.3.4 Synchronized Deployment

If number of panels to be deployed is more than one, then in order to balance the movement of two solar panels during deployment, it is necessary to deploy solar arrays together and insure that all panels are deployed at the same time. This is achieved by incorporating a Closed Cable Loop (CCL) system in it[3]. In a CCL a pulley is integrated with the hinge and a metallic wire is fixed with the pulley which connects all the hinges of solar array together. The metallic wire of CCL is designed to withstand the tension during deployment of solar arrays.

1.3.5 Deployment Status

The status of solar panel locking after deployment can be found by incorporating a micro switch in the hinge. Once the locking pin is inserted inside the slot, the electrical circuit of a micro switch is closed and telemetry is generated and sent to ground. This telemetry confirms that the deployment of solar arrays is successful.

1.4 Scope and Objective

There has been a considerable increase in use of space systems in the past few decades. Space applications play a key role in our daily lives, from both global point of view to national point of view. These technologies play a notable role to foster economic growth at regional level. These technologies are used broadly in our everyday life, through use of internet, forecasting for weather, crops estimation for agricultural, however unfortunately such a useful application of these technologies is not well known at the citizen level, nor it has been fully utilized in collaboration at the local and regional level[11].

The weight of the first satellite, named Sputnik 1, launched by Russia in 1957, was around 80kg.Since then satellites have been grown heavier, larger in volume, more complicated and many of them weigh as much as 5 tons. But in late 1980s, a new paradigm of small satellites opened a new category for space applications as currently space agencies emphasize more on cost reduction in satellite developments and their in orbital operation. The classification of Satellites as per Mass has been defined differently by different space agencies, however I have tabulated the Satellite classification according to American Institute of Aeronautics and Astronautics, Inc (AIAA) in Table [1][12].

Class	Mass Range
Femto	10-100 gm
Pico	0.1-1 Kg
Nano	1-10 Kg
Micro	10-100 Kg
Small	100-500 Kg

Apart from the classification of satellites by mass, they are also classified by volume. That class of satellites is termed as CubeSats, which is also a well-defined category of Small Satellites and has a standard form factor where 1U = 10 x 10 10 cm. This CubeSat standard was developed by California Polytechnic State University's Multidisciplinary Space Technology Laboratory (MSTL) and Stanford University's Space System Development Laboratory (SSDL)[12]. The use of deployable solar panels is not only used by small satellites but it is also extensively used for CubeSats[13] due to the requirements of unique space applications.

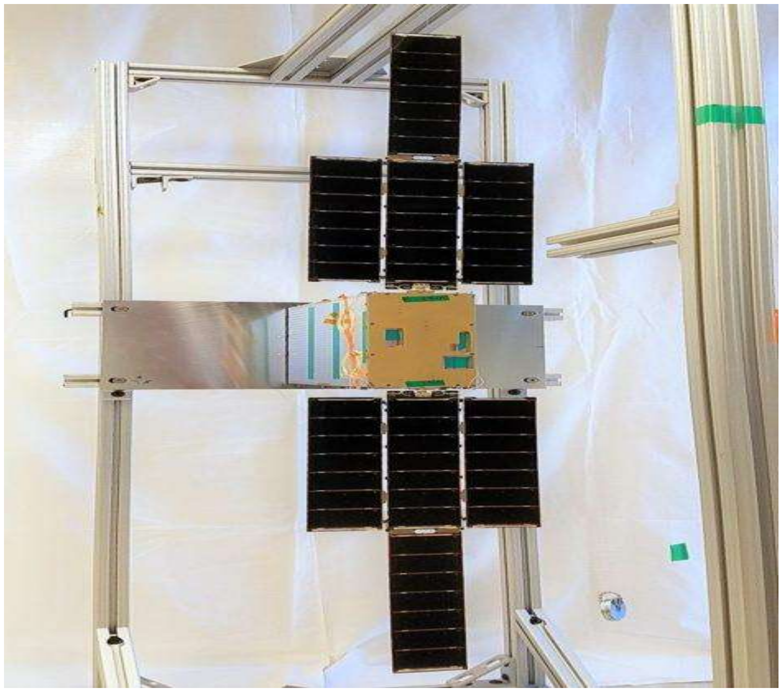


Figure 6: Deployable Solar Array for CubeSats [11].

Most of the satellites launched in recent years are small satellites (also termed as SmallSats) having mass from 25 to 200 kg (or up to 500 kg). Thus the SmallSats application is more beneficial for the industries and is more focused to mission objectives. According to Institute of Defense Technology, USA in upcoming decade the SmallSats imaging market may grow up to $164 million till 2020[14]. The industrial investment for SmallSats has shown in Figure 7[14].

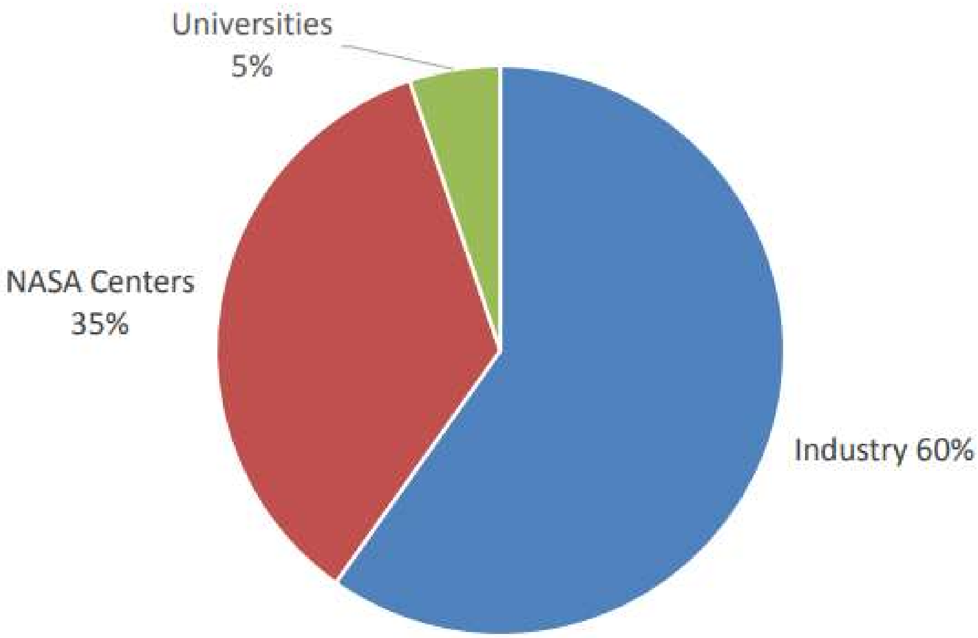


Figure 7: Industrial Investment Trend for Small Satellite [10].

In the next 10 years there are about more than 6000 SmallSats are expected to be launched [15].

There is no authentic source of determining the exact number of satellites launched by each country. Spy satellites are often launched in secret and not included in open catalogues of space objects. Although the United Nations Committee of Peaceful Uses of Outer Space (UNCOPUOS) defines the responsibility of each member state for useful use of outer space, which addresses from launching, registering (UN Registration Convention 1975)[16] till deorbiting of space objects, many states do not adhere strictly to the Outer Space Treaty (OST). Therefore it is almost impossible to exactly determine the number of satellites in space. However some of the companies like Aerospace Corporation [12], Futron Corporation[17], Northern Sky Research (NSR) [18], Spaceworks [19] etc, have published market assessments for the space industry.

It can be easily concluded from Figure 8, that there is a large market trend of small satellite in future for both communication and earth observing satellites. One of the most widely used application of space is the remote sensing by using optical payload. These payloads are mostly launch in Low Earth Orbit (LEO) and are carried by small satellite.

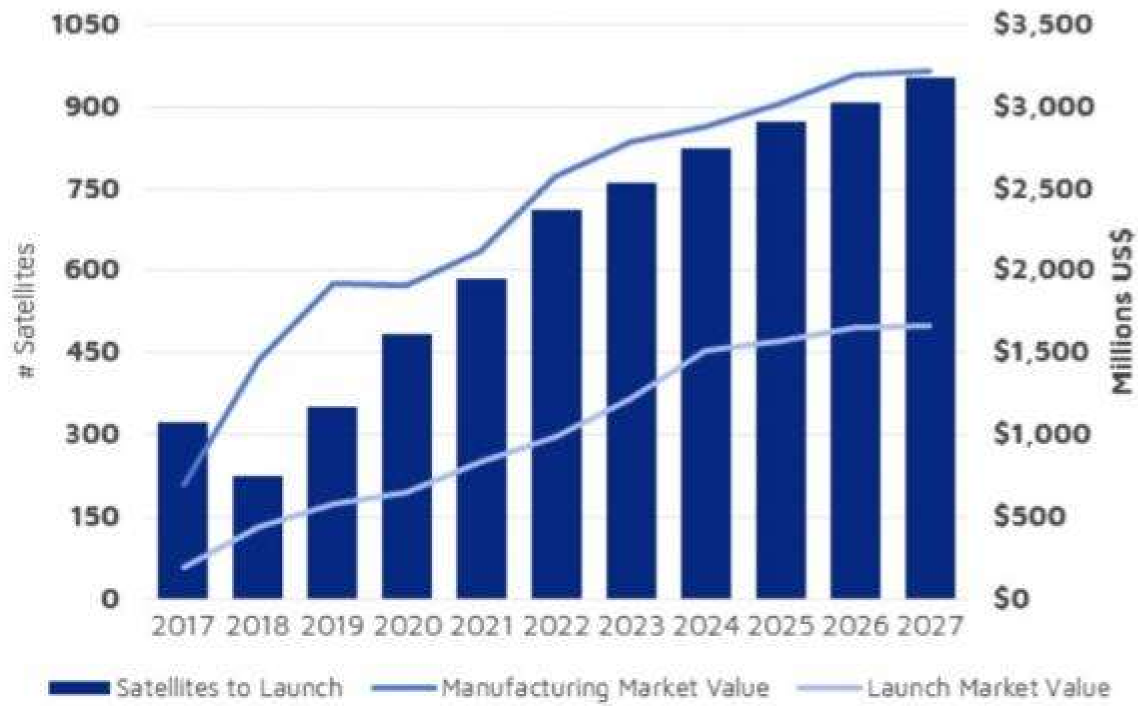


Figure 8: Global Small Satellite Launch and Market Value (Source NSR).

According to Spaceworks market analysis report, the satellite application trend of small satellite for Earth observation, communication and remote sensing will be as per Figure 9.

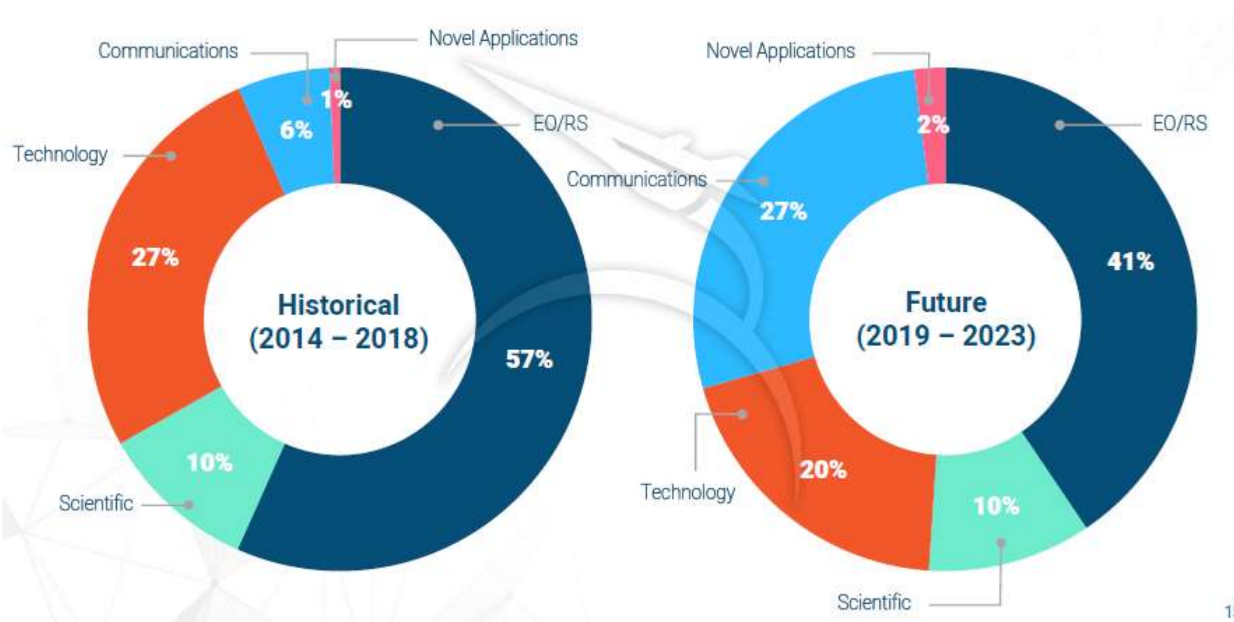


Figure 9: Market Trend for different application small satellite (Source Spaceworks).

In the last few years, there has been a considerable increase of low-cost space missions. Almost every small satellite has appendages like Solar Panels or Antennas. These appendages have large inertial forces with low structural rigidity and mass to size ratio. As stated earlier these appendages are in a stowed position in the launcher to accommodate the whole satellite in the launcher. However, after launching, these appendages are deployed to obtain the required power for the satellite operation. Their power producing capability is directly related to their area of exposure to sun, which is why these panels are available in different sizes and configurations. The requirement for the deployment mechanism for these appendages needs to be cost effective, compact, lighter in weight and reliable. It is entirely dependent on the structural designer to create such a robust mechanism that conforms to the requirements of the mission [20].

It can also be forecast that a big change in load carrying capacity of launcher is not expected in the near future, so the only way to fulfill the power requirements of any of the space mission is to design the mechanism, which can hold large structures in a smaller volume. But these deployable structures should be capable to withstand the launching loads in the stowed configuration. The increase in commercial use of space systems, which focuses on both small and low cost in comparison to Military use of satellite also requires the cheap and reliable system. So keeping in mind that the requirements of a power system for satellite which solely depends on its solar panels, there is need that the mechanism system should be reliable and low cost as a failure in space are very expensive and cannot be corrected.

The primary objective of this research work is to propose a solar array deployment mechanism for space applications, keeping mind the trend of the commercial space industry to develop small satellites for low cost and reliable space missions.

This research work is concerned with the mechanical aspects of the deployment mechanism, like its stiffness, strength, type of material, manufacturing, deployment force, etc. Other related design parameters, like orbit, propulsion system, AOCS system and communication system will not be addressed in this research work.

1.5 Thesis Overview

In Chapter 2, I will provide an overview on satellite failures with some leading cause of the power system failures due to deployment mechanism, like failures due to Pre-Loading, Space Environment, Lubrication, Cold welding, Analysis, Zero ’g’ Testing.

In Chapter 3, I will discuss about the requirements for designing the deployment mechanism. The requirements are normally generated by system engineering but for this project I will assume the power requirements for a small satellite with honeycomb structure solar panels with an exposed area of $1m^2$.

Chapter 4 will provide an overview of the 3D Model for a proposed deployment mechanism from its conceptual design (TRL1) until proof of concept (TRL3).

In Chapter 5, I will discuss the Zero ‘g’ considerations for the deployment mechanism with its Failure Mode Effective Analysis (FMEA).

In Chapter 6, I will conclude this dissertation with conclusions and recommendations for future work.

2 DEPLOYMENT FAILURES

When a satellite is launched in space, the first step is to power up the subsystems as per the required operating mode of the mission. This is to be achieved by solar arrays, which convert solar energy to electrical energy as stated in section 1. Although the battery is part of the power subsystem, it cannot last for a long period and the power requirements of all subsystems cannot be fulfilled only by the battery. So solar panels must be deployed in a required specific period of time and it is considered as prerequisite for the normal operation of the satellite. The unexpected behaviour of a subsystem during the satellite's operation is termed an "anomaly". Sometimes an anomaly in a subsystem indicates an expected failure of the satellite. The failures can be due to Design, Environment or long term operation (wear and tear). Unexpected failures are more likely to occur in moving parts, such as mechanisms and reaction wheels. The reaction wheels are mounted inside satellite and have controlled environment, However the mechanism are mounted on the outer surface body of satellite which is exposed to harsh space environment. Therefore its probability of occurrence is high in continuously moving parts, like trackable solar arrays. Due to the high power requirements for the satellite the mechanism has become the crucial part of the spacecraft and therefore the possible failures during its operational life require more attention. The percentage of satellite failures due the solar arrays is the highest in comparison to other subsystem, as shown in Figure 10.

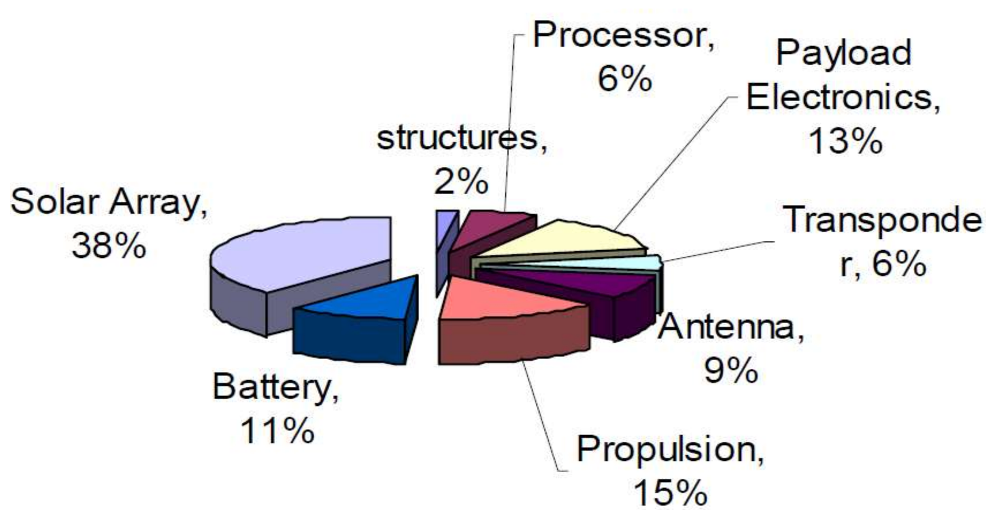


Figure 10: Percentage of Satellite failures [21].

According to the failure statistical analysis report for 1584 in orbit satellite missions between the years of 1990 to 2008, most of the satellite failed due to the failures of solar arrays [22]. Some of the failures were also reported within 30 days after the launch. The study of this important issue of failures is highly necessary due to the unpublished reports of failures [23]. As stated earlier, the function of the deployment mechanism involves the process of its holding, unlocking, deploying and finally keeping the solar panels in the required position.

It has been mentioned in Chapter 1 that the proposed deployment mechanism for this dissertation is a flip out solar panel mechanism with rigid panels. As the solar panels are the only source to fulfil the power requirements of a satellite for a specified mission life and its failure may be considered as the failure of the mission, therefore its study is very important. In subsequent sections we will discuss in detail the impacts of some parameters which result in deployment failures.

2.1 Space Environment

The mechanism is one of the important parts of a satellite which is mostly exposed to the vacuum, cold welding and thermal cycling load due to the harsh space environment,

The space environment is very hazardous due to the strong ultraviolet and nuclear radiation from the Sun. The material used for the mechanism design shall be such that the specified performance requirements (weight, stiffness, thermal and shock loads) will meet when the satellite is operated in the actual on-orbit environments over the Mission Life. Therefore, Aluminium Alloy AL-6061-T6 is selected for the hinge design, and Stainless Steel Alloy 304L is chosen for its locking mechanism due to its high strength. The impact of the solar activity is different for each orbit and therefore the radiation environment for each orbital regime is different. The space environment for each orbit type has been tabulated in Table 1.

Table 1: Environment for different Orbits

Orbit	Environment
LEO (Low Earth Orbit) <1000Km	Vacuum, Solar UV, Earth Albedo, Wide spectrum of energetic Electrons and Protons (Solar flux 10^4 to 10^6 particles/cm^2-sec)
MEO (Medium Earth Orbit) 2000<h<18,000Km	Vacuum, Solar UV Wide Spectrum of energetic electrons and protons (Solar flux 10^{11} particles/cm^2-sec)
GEO (Geosynchronous Earth Orbit) h>35,000Km	Vacuum, Solar UV $1x10^4$ keV protons (2×10^8 protons/cm^2-sec) $1x10^4$ keV electrons (2×10^8 protons/cm^2-sec)

In addition to the transfer of heat in space by means of radiation and conduction only (i.e. no convection), the following features of the space environment at Low Earth Orbit (LEO) have major impact on the satellite:

- Vacuum
- Atomic Oxygen
- Ultraviolet Radiation
- Particulate or Ionizing Radiation
- Plasma
- Temperature Extremes and Thermal Cycling (Coefficients of Thermal Expansion [CTE] Mismatch)
- Micrometeoroid/Orbital Debris Impact

The level of vacuum for satellite is different for each altitude. The higher the altitude of satellite, lower will be the vacuum. The standard density of air at sea level is $\rho = 1.225 \times 10^{-3}$ gm/cm^2 and the standard pressure is 101.325 Pa. At the height of 90km, the pressure is 0.2 Pa. This is further reduced to 1.5×10^{-4}Pa at the height of 200km, 10^{-6}Pa at 500 km and 10^{-8}Pa at 1000 km. The space environment presents challenges in the form of solar ultraviolet

(UV) radiation, charged particle (ionizing) radiation, plasma, surface charging andarcing, temperature extremes, thermal cycling, impacts from micrometeoroids and orbital debris (MMOD), and environment‑induced contamination. Given that the satellite is exposed to the harsh space environment during its entire operational life, the possibility of failure of the deployment mechanism due to degradation of material cannot be ignored. The major cause is the presence of single-oxygen atoms (atomic oxygen [AO]) in Low Earth Orbit (LEO), ranging from 200 to 1000 km. This AO plays a key role in degrading metals and materials containing nitrogen, carbon, sulphur and hydrogen bonds, which means that many non-metallic materials also erode quickly. The speed of a satellite at LEO is ~ 8km/sec, which means that a strike by atomic oxygen generates 4.5eV of energy.[24].

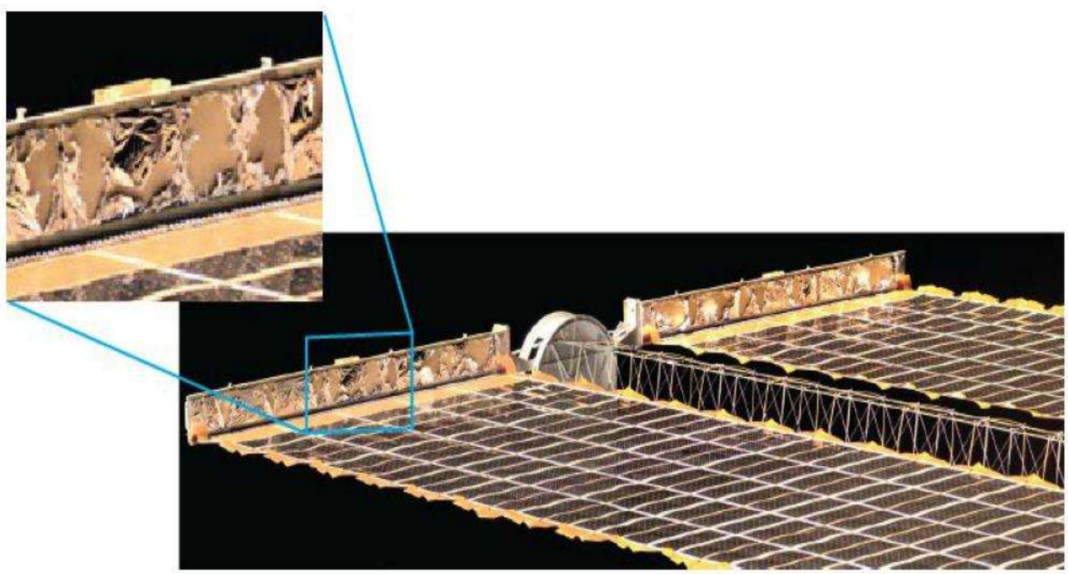


Figure 11: Damage to the blanket of Solar Array wing by Atomic Oxygen [25].

The degradation mechanisms due to space environment are summarized in Figure 12.

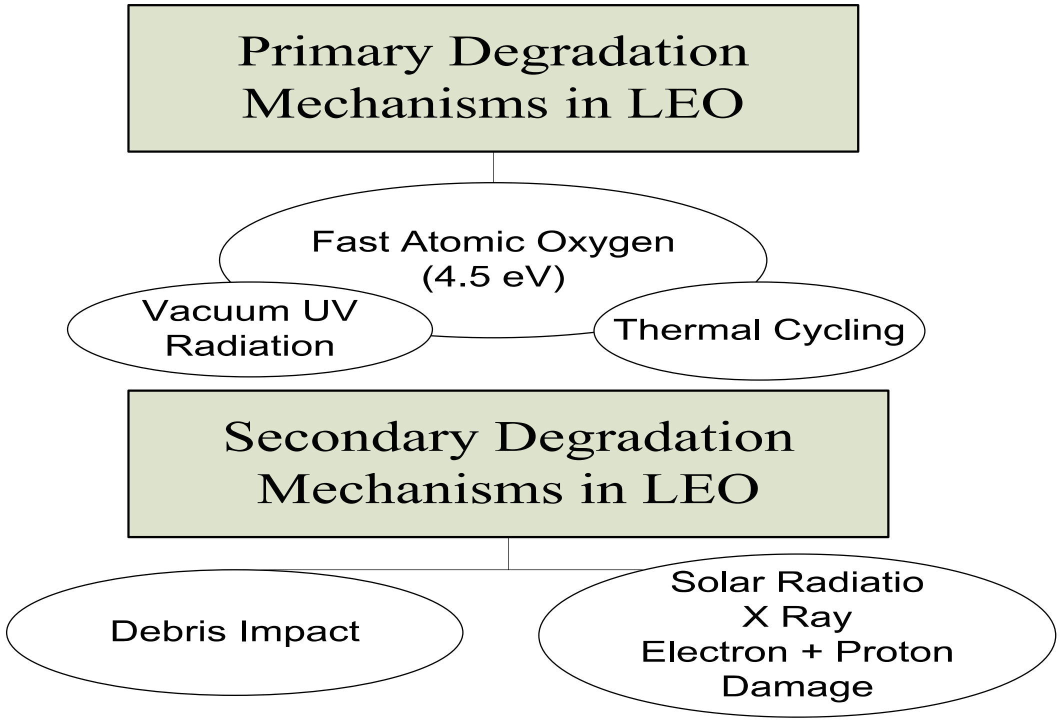


Figure 12: Mechanism degradation due to space environment [25].

The hostile space environment related to “0” G, vacuum and severe thermal cycling (-170ºC to 165ºC) needs to be considered when designing a deployment mechanism.

2.2 Tribology

The term tribology defines the study of friction, lubrication, wear and is the science of the interaction of surfaces in relative motion. The study of lubrication field for the reliable performance of satellite in space environment is termed as space tribology. Tribological systems play a key role for the reliability, performance of space missions. As previously stated, failure of solar arrays are a cause of the highest rate of mission failure. Therefore a proper lubrication study is required for the design/selection of lubricants for deployment mechanisms for solar arrays. In comparison of the whole space mission cost, the tribological components represent a friction of cost. However these components are often considered as single point failure (SPF) and therefore can render the failure of the high cost mission.

An example of the failure of a space mission due to an anomaly of the tribological system is the Galileo mission. This mission was launched in 1989 from Space Shuttle Atlantis. In this mission an umbrella-like antenna was used for ground system communication. The antenna had the capability of deploying itself in space after launching. However it was found in 1991 that the antenna started malfunctioning as four out of its 18 ribs stuck together due to cold welding. This was anticipated while performing some tests on the same antenna on the ground. The possible reason was the drying out of the lubricant used in the mechanism [26]. The failure of space missions due to deployment mechanism failures has been tabulated as Table 2.

Table 2: Satellite failures due to deployment mechanism [27].

S,No	Satellite	Failure Cause
1	Kosmos 844 / Yantar' 22 July 1996	Solar Panel did not Deployed
2	NOAA 13, August 1993	Solar Panel not Deployed
3	STEP M 4 / P95-1 October 1997	Solar Panel not Deployed
4	TERRIERS, May 1999	Solar Panel not Deployed
5	LDREX, December 2000	Antenna not Deployed

In order to perform the required functioning of the mechanism, both solid and liquid lubricants can be used as recommended by the designer. However the following properties need to be considered while selecting the lubricant:

- Vapour pressure should be low
- High temperature range
- Low flow out properties to avoid unintended flow
- Electrically and Thermally conductive
- Shouldn’t be insensitive to vacuum or air
- Wear life should be large enough
- Need to fulfil required frictional characteristics

The recommended solid and liquid lubricants recommended by NASA are Molybdenum disulphide (MoS_2) or Niobium Di-selenide ($NbSe_2$) and Braycote 600 series, respectively [21].

2.3 Mechanical Loading

The main function of the mechanism is to deploy appendages (Solar Arrays, Antenna) and hold it throughout the design life of spacecraft. The present work focuses only on the passive deployment as described in Chapter 1 and the proposed system is composed of flat spiral spring as stated in section 1.3.1.

Figure 13: Flat Spiral Spring.

The spring is considered as an elastic component that has the capability of storing the deformation energy when it is loaded or wound up. This deformation will disappear when the energy is released. Spiral springs have already been used for the deployment of solar panels in many space missions [28] [29].

Figure 14: Deployment Hinge using Spiral Spring [28].

The process of storing the energy in a spring is termed as mechanical loading. This energy is stored in the spring by applying the torque at its free end. When required, this stored elastic energy can be converted to a useful form of energy. In our case this stored elastic potential energy can be used to deploy the solar panels. In order to fulfil the deployment requirements, the mechanism should have the capability to overcome the resistance torque due to friction between two parts and should produce the required acceleration [30]. The spring loading should be carried out in way that there should not be any coil interaction when the spring is wound. If a minimal gap between coils is not maintained then in space there is possibility of cold welding between adjacent coils and the spring will not uncoil to release its stored energy.

It is therefore necessary that the mechanism should have enough required torque with margin to deploy the panels at required position or the mechanism should have enough margin of its static torque.

2.4 Modeling and Analysis

Simulation software provides additional tools for satellite simulation to engineers and various other team support personnel. It will help in power and fuel budgeting, satellite orbit and constellation studies and maneuver planning, as well as to analyze and visualize complex structures with 4D dynamic datasets.

They will take in a multitude of parameters for measuring and running calculations on a satellite's orbit, such as the amount of power produced from solar energy and complex astrodynamics. Modern models can also point out the exact times a satellite can interact with the Earth. All of these data are taken into account when constructing an ideal satellite launch and maintenance mission profile. Effective modelling has given the ability for engineers, scientists, astronauts and other team members to optimize the space systems necessary for ambitious missions and develop the skills needed to operate and sustain them. The chances of safety and success are significantly increased by optimizing familiarity and reducing the risk of unknown factors interfering with a mission. Although simulation and modeling provide the sector with wide-ranging resources to understand and optimize the diverse mission performance systems needed, however the whole mission design cannot be only rely only on simulation. The simulation results need to be validated through testing. Having said that, deployable structures and mechanisms, particularly for large, lightweight structures, are often extremely difficult to model / analyze with precision as a zero ‘g’ environment and high vacuum are hard to simulate for large structures.

2.5 Assembly Integration and Testing

The main purpose of testing is to:

- Provide empirical design data
- Evaluate the functional capabilities of system
- Evaluate design limits
- Assess the capacity to work in the necessary environment
- Determine whether production units are of the same standard as qualification units

Technology Readiness Level (TRL) is a metric used to estimate the maturity of the product/unit during its acquisition phase for space or industrial application and it was originated by NASA, where it was developed as a tool for assessing the maturity of space exploration technology and since then it has been co-opted by other sectors, such as the EU Research Council and the European Space Agency (ESA).The TRLs are based on a scale of 1 to 9, with the most mature technology being 9.

These levels have been defined in Table 3.

Table 3: Technology Readiness Level (TRL).

TRL-1	Evaluation of basic principles
TRL-2	Conceptual Design
TRL-3	Analytical Analysis and Experimental proof of concept.
TRL-4	Lab Validation of Technology
TRL-5	Validation in relevant environment
TRL-6	Demonstration of System/Subsystem in relevant environment (Space or Ground)
TRL-7	System prototype demonstration in space environment
TRL-8	Actual system completed and “Flight Qualified” through test and demonstration
TRL-9	Actual system “flight proven” through successful mission operations

In order to be qualified for the space application, a product has to pass through a series of testing and prototyping. This whole process is termed as System Verification and it is performed in various sequential phases according to a space mission project's life-cycle. The various verification phases pertaining to ECSS-E-10-02A are [31].

- Qualification
- Acceptance
- Pre-Launch verification

- In orbit verification
- Post Landing Verification

The verification is performed by either of the following methods.

- Testing
- Analyzing
- Design reviewing and
- Inspection.

Testing is a method of verification in which technical means, such as the use of special equipment, instrumentation, simulation techniques, and the application of established principles and procedures, are used for the evaluation of components or equipment to determine compliance with requirements. Some test methods include [2]:

- Development tests
- Design verification tests
- Prequalification tests
- Qualification tests
- Preproduction, pilot model, pilot lot tests
- System integration tests
- Production acceptance tests
- Production monitoring tests,
- Quality verification tests
- Reliability tests

The modal philosophy for the qualification of the unit is adopted on the basis of available test facilities, available time frame for testing and, most importantly, the available budget. Normally three models are proposed for qualification of any space unit, i.e. Structure Thermal Model (STM), Engineering Model (EM) and Flight Model (FM). The qualification of a Deployment Mechanism requires Vibration, Shock, and Thermal Vacuum Test Facilities at a bare minimum level and most important is the Zero 'g' facility. The level of testing is generally a balance between the testing needed to ensure reliability, and the time, resources, and facilities available to perform the test. If the proper level of testing is not selected, it may lead to the failure of space mission.

The approach for the verification of the model plays a key role for its qualification. If a proper approach from designing to verification is not adopted to obtain the TRL of the unit,70 to 80 per cent of all assembly and integration issues are triggered by a lack of good systems or technological integration planning [21]. Many of the deployable structure / mechanism issues relate to manufacturing errors; cables / interconnects / solder joints and switches (pyro). The deployable mechanisms should be self-supporting during storage conditions. Verify final assembly operations, in particular on deployable mechanisms that constitute single-point risk of failure. It’s hard to achieve complete fidelity of space environment simulation during ground testing, specifically a zero ’g’ environment with high vacuum condition in the presence of radiation. However tests endeavour to simulate as much of the space environmental conditions as possible during ground testing

2.6 Risk/Mitigation Deployment Mechanism

The deployment mechanism requires having the least possible number of parts without compromising the accuracy requirements of the system.

In the following section, the general Risk and the Mitigation techniques for Deployment Mechanism hereafter called as deployable are discussed in detail.

2.6.1 Stiffness and Strength

The deployment requires some alignment accuracy for sun pointing; therefore there is risk of losing the required accuracy if the mechanism does not fulfil the launching and deployed load requirements. However this can be mitigated to design a system with materials which have high strength and stiffness to survive environmental and space launching loads. The high stiffness and strength will also be helpful during on ground testing of mechanism, otherwise these is a high risk of sag, consequently leading to a loss of pointing accuracy.

2.6.2 Redundancy

The deployable shall utilize redundancy wherever applicable in the design to improve the reliability of deployment. Techniques such as use of redundant spring or lead screw, use of backup deployment device shall be considered to satisfy the requirement.

2.6.3 Deployment Motion

The deployment motion should be restricted by a full range of motion by using a locking pin other suitable means. If this is not restricted then the required pointing accuracy of the panels towards sun may not be achieved, so to mitigate it, the deployable should have a locking mechanism. If a spring force is used for the deployment, the effects of temperature should be considered for better performance of the spring and its stress level should be kept below the material proportional limit.

2.6.4 Centre of Gravity

There is a high risk that the deployable may shift the C.O.G of the whole system after deployment. Therefore it is necessary that its C.O.G should coincide with the axis of rotation or axis of deployment. This is necessary for the ground testing of the system.

2.6.5 Cold welding and Lubrication

There is a high risk of permanent joining of the two adjacent moving parts in contact with each other, which is termed cold welding. This phenomenon occurs where parts are in contact with each other for a long period of time under vacuum conditions. To avoid it, a dry film based lubricant should be used instead of liquid or soft lubricant as there is a high risk of deposition of out greased material from lubricant on the lenses. This can also be mitigated by using a heating source at the mating points. To avoid corrosion, the dry film lubricant that is used shall not contain carbon or powdered metal.

2.6.6 Harnessing

The risk of entanglement of the harnessing in the deployment path should be considered while defining the harnessing path. Therefore there is a need to define the loop sizes where a harness crosses over a rotation interface.

2.6.7 Preload Application

The expected response of Load to Displacement for any of the mechanisms is always nonlinear. The possible reason is that there is always a clearance between the two joining parts of a mechanism. Due to this clearance there is a free play before both parts join together and one part pushes the other, as shown in following Figure 15.

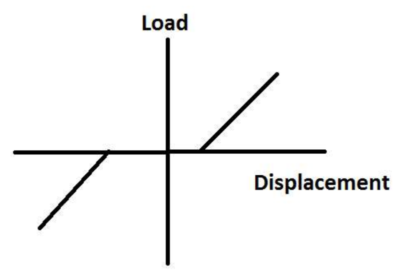


Figure 15: Free play between mating parts.

This free play causes the nonlinear load to displacement curve which results some loss of the required displacement of the deployable. This mismatch of the parts is avoided by incorporating a mechanical preloading device to maintain the contact in load transferring interfaces. This will eliminate any free play and slippage between parts and will maintain a stable contact between parts.

2.6.8 Thermal Instabilities

There is a great impact of mismatch between two parts due to the presence of a thermal gradient. This impact cannot be ignored in high precision deployable. Therefore to avoid this mismatch, the selected materials should have the low coefficient of thermal expansion (CTE), to minimize the differential expansion/contraction between the two parts as much as possible.

2.6.9 Cost

An effort will be made to propose a deployable which is commercially available in the market and fulfils our requirements. The active systems cost more in comparison passive system. So it is recommended to design a system using Spiral, Compressible/Coil Springs, Linkages, and Guiding Rails, which consume less power to release and have lower risk of failure to and consequently minimize the cost.

2.6.10 Time line and Expertise

The risk for the development of a new system from a concept to qualified model is high. However this can be mitigated by proper utilization of sources and by carrying out some activities in parallel, like Structural Analysis, Dynamic analysis and Thermal Analysis. Together with this the development of Zero ‘g’ facility can be started in parallel.

2.6.11 Zero ‘g’ gravity with compensation system testing

Ground testing for any deployable space mechanism is mandatory for a space qualified product. The main constraint for deployment testing on the ground is to create a zero ‘g’ environment under vacuum conditions. The reason for creating a Zero ‘g’ environment is that the spring is designed by considering only the Moment of Inertia for the Solar Panels at 0 g. If it is designed at 1 g, it means after deploying in space (0 ‘g’) it will create a large amount of shock and consequently detumbling the satellite will require a long period of time. Therefore it is necessary to minimize the gravity effect on ground using a compensation system. The following methods can be used to simulate the 0 ‘g’ environment:

- Air Floating System
- Truss Structure Suspension system using steel rods.

3 REQUIREMENTS

It has been mentioned in Chapter 1 that by using a deployment mechanism for flip-out or the multi panel deployment configuration, it is possible to achieve a larger area solar array for power generation. This will also reduce the cost of the solar panel by using the low efficiency solar cells over a larger area in comparison to a body mounted solar cell as the Sun vector is not always pointed normal to the satellite body.

The deployment mechanism is normally considered as the electro-mechanical component of a satellite and its requirements are closely related to the other subsystems of satellite such as the power subsystem and attitude control which keeps the orbital position of satellite. As stated in Chapter 1, our aim is to focus on a free deployment like flip-out mechanism, not on a controlled deployment like use of electrical system to track the Sun. In Figure 16 the available options for solar arrays are shown.

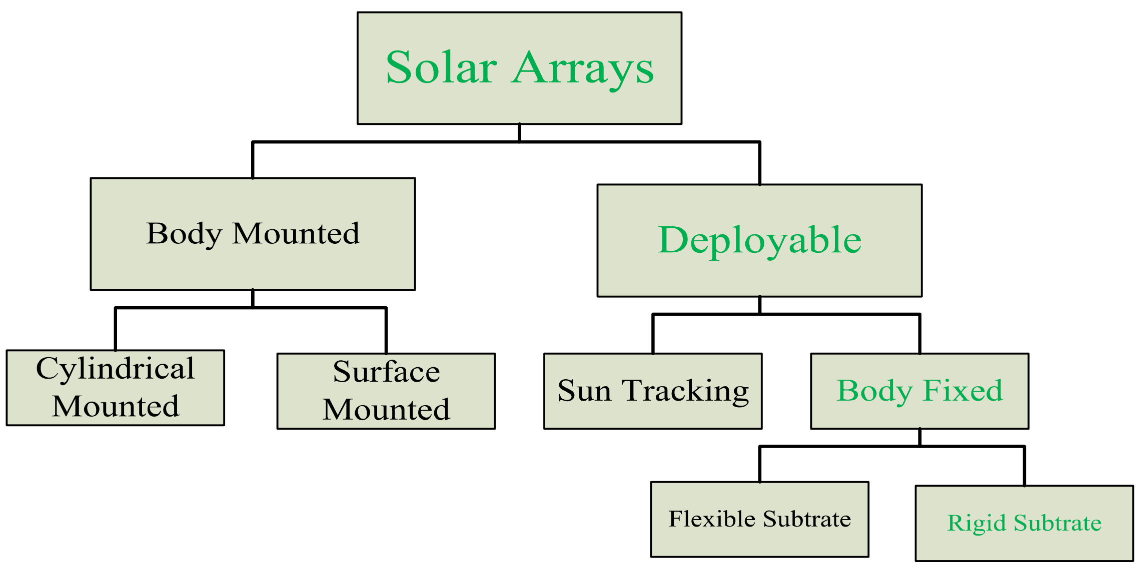


Figure 16: Solar Arrays options for Satellite [2].

The requirements of low cost and weight are considered to be primary requirements and small in volume, reliable in performance, be capable to produce sufficient motive force to deploy both on ground (during testing) and in space are considered as its secondary requirements for the deployment mechanism.

The most widely used space mechanism which has a rigid connection that provides relative rotation of two bodies is the rotating pivot. This mechanism being part of the whole satellite has to fulfill the main requirements such as:

- Weight
- Stiffness
- Clearance
- Envelope
- Interface
- Alignment
- Environment (High and Low temperature, Vibrational loads, Shock loads, Vacuum)

These factors are the most important which influence the design of the deployment mechanism [32].

Current space mechanisms are almost entirely composed of traditional rigid-link assemblies. These mechanisms perform a variety of functions and one of the most demanding functional requirements of such mechanism is generation of a deployment force by using torsional springs. The use of a torsion spring in mechanisms has been widely demonstrated for the proposed mission of Netlander by CNES (French Space Agency) and ESA (Europe Space Agency).

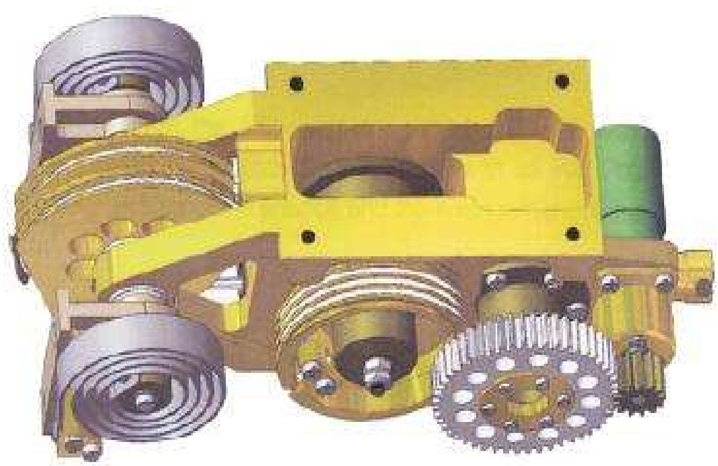


Figure 17: Solar Arrays Deployment Mechanism using Spiral Torsion Spring [31].

The advantages of proposing the Spiral Spring for our system are its capability of providing the torque directly, its linear characteristics and that it decelerates as soon as it reaches the unreleased position, thus providing a big advantage to avoid shock at the end of deployment. [33]. However its main disadvantage of being less stiff to act as a hinge can be overcome by designing a less massive and stiffer housing assembly to hold it.

We have assumed that the deployable panels are required to fit alongside the satellite. These requirements for the mechanism are based on the power subsystem of the satellite, but for our case the assumption made in Table 4 will be used to design the mechanism.

Table 4: Power Subsystem for Small Satellite.

No	Items	Requirements
1	Satellite Volume	1000 x 1000 x 1000 mm
2	Number of Panels	2 panels
3	Dimensions of panels	1060 × 950 mm
4	Deployment Rotation	0^o to 90^o
5	Power at Beginning Of Life (BOL)	320 W
6	Power at End Of Life (EOL)	290 W
7	Mission life	at least 3 years
8	Storage Life	2 years
9	Orbit type	SSO
10	Sun Trackable	No
11	Mass of each panel	4.2 Kg
12	Panel Frequency in Stowed configuration	>200 Hz

The requirements for the stowed frequency can only be fulfilled if the location of the holding points for the panels with the satellite are known. As we are proposing only the deployment mechanism so the holding mechanism of the solar panel with the satellite will not be considered for this dissertation. However the thermal technique guidelines for the designing of the holding mechanism will be followed in this report.

3.1 Functional Requirements

The main function of the mechanism is to restore the solar panel in the stowed position in launcher and provide enough Kick-Off force against the satellite body to deploy the solar panel in space [34]. The other functional requirements are:

- Provide support and connection to moveable components which change their position relative to the satellite body.
- Provide kinematics requirements for each change in position of displacement, velocity, acceleration, time, positioning (orientation) applied force or moment, preload, deployment accuracy and allowable errors.
- Provide required allowable envelop (stowed position in launcher), scope of moving components in various states stowage, deployment, rotation etc. The deployment envelop should not interfere with any other mechanical component of satellite.
- Provide enough strength and stiffness to withstand all predicted loads of handling, testing, launching storage, transportation, orbital life without degradation and within required mass.
- It is necessary to define the coordinate system of mechanism for both deployed and un-deployed position. This is necessary to predict the position of the satellite in orbit with the help of AOCS system. Together with this the degrees of freedom for the mechanism should also be defined including its translational and rotational motion.

3.1.1 Support Structure

The photovoltaic cells are the main components of the solar arrays. To generate power these cells must be fully exposed to the sun vector when the satellite is in orbit. The proposed mechanism requires to deploy the solar panels as a flip-out configuration having solar cells mounted on a rigid surface. This is termed the support structure for the solar cells. It is also termed the mechanical part of the solar panels, while the solar cells are termed as the electrical part of the solar panels. The mechanical part is also termed as the substrate, is a sandwich structure composed of an aluminium honeycomb core with two composite face sheets.

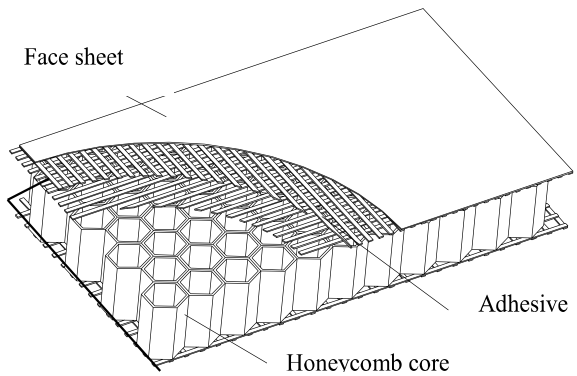


Figure 18: Solar Panel Composition.

Solar cells are bonded on the composite face sheet with adhesive. These types of solar arrays are most widely used due their high stiffness after deployment and therefore termed as Rigid Solar Arrays. These panels are interconnected (with each other in case of more than two) with satellite body by mechanism system and electrical harnessing. These types of solar arrays are designed by considering the following parameters:

- Satellite configuration
- Total area of solar array
- Total mass of solar array electrical part
- Outer panel solar cells orientation at stowage configuration
- Stiffness at stowage configuration
- Stiffness at deployment configuration
- Total mass
- Shear and Bending Strength
- Electrical circuits design

- Deployment torque margin
- Acoustic vibration environment
- Sinusoidal vibration environment
- Static load
- Isolation and grounding requirements

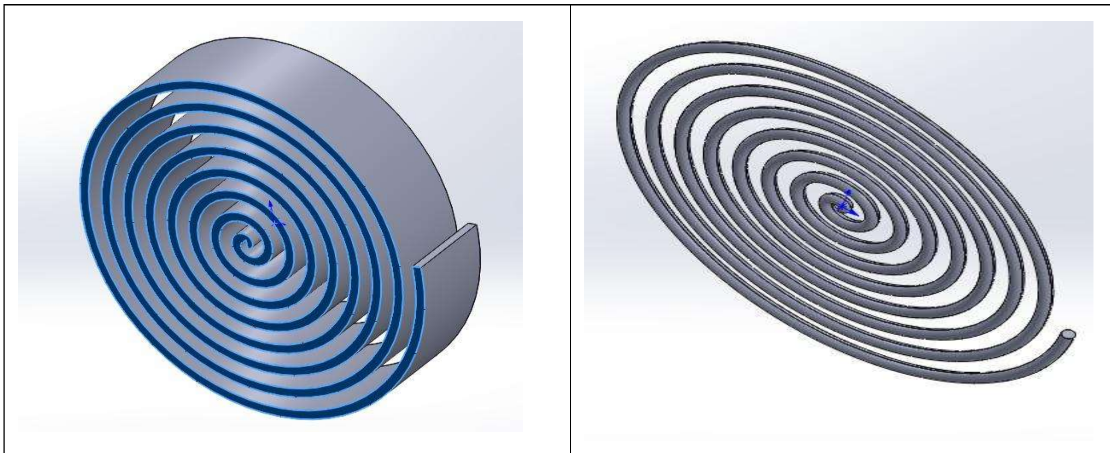
The sizing of the support structure for the solar panels plays a vital role in achieving the mission requirements from its cost, weight, volume in stowed position, to generation of power until the end of life for the mission. Therefore the mechanism system should have all characteristics to withstand against all the requirements of solar panels, from absorbing shock (during ground testing) till its stiffness requirements (both on ground and space).

3.1.2 Deployment Torque

As stated earlier (Chapter 1) the proposed mechanism is based on a Spiral Torsional Spring which has the capability of applying the required torque with storing the rotational energy in elastic form. This stored energy can be released in its required form, like for our case "To deploy the solar panels". The proposed spring has a uniform rectangular cross section having the shape of an Archimedean spiral.

Spiral Torsion Springs are made up of material like high carbon steel or stainless steel (SS304-L). It is a rod (Circular cross section) or bar (Rectangular cross section) curved on a plane with one end fixed while other end produces the rotating torque. The coils are wound in a way that they do not contact with each other during operation resulting in a friction free force when mounted correctly.

I have proposed the flat torsional spring having rectangular cross section with width "b" and thickness "t" as they rotate in exact circular motion in comparison to round shape spring. These type of springs are shown in Figure 19.



(a) Rectangular Crossesction	(b) Circular Crossection

Figure 19: Types of Spiral Torsion Spring

The torque delivered by these types of springs is linear for each revolution for the first 360^o, however as soon as the coil get closed to the fixed point the value of torque increases rapidly and it becomes non-linear. As the proposed mechanism type is used for flip-over deployment of solar panels (Chapter 1),this type of spring will be very useful to fulfil the angular rotation from 0^o to 90^o (<360^o) as per Table 4.

The coils in a helical torsion spring are usually closely wound around each other like an extension spring, but do not have any initial tension as shown in Figure 20. However the coils of the spiral spring are wound with clearance between the coils so that they do not have any contact with each other.

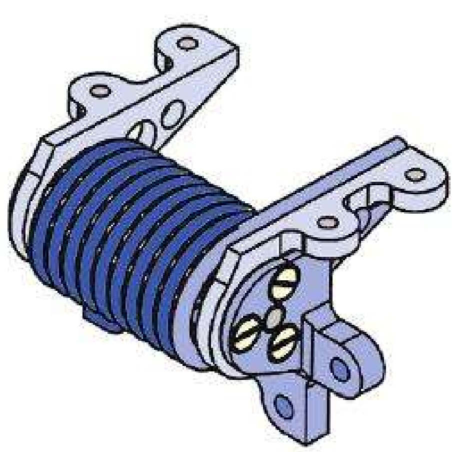


Figure 20: Solar Arrays Deployment Mechanism using Helical Torsion Spring [31].

The stresses produces due to application of force or torque in a spiral spring are bending stresses. These springs are wound either left or right handed in a way that the deflection in the spring due to application of force winds up the spring in the free position. The diameter of such a spring reduces as it winds up and a small clearance is provided between the coils to avoid the friction. [35] As stated earlier a torsion spring exerted the force (torque) in circular arc, while its arm rotates along its central axis.

The applied force P at the free end of the spring will produce a torsional moment PL and the spring wire will subjected to bending stress over its active length.

Due to application of the force at the free end of the torsion spring, bending is produced in spring and due to the curvature effect of the spring this stress will be maximum at the inner coil radius. The spring will behave as a simply supported beam with a point load at end free end, as shown in Figure 21.

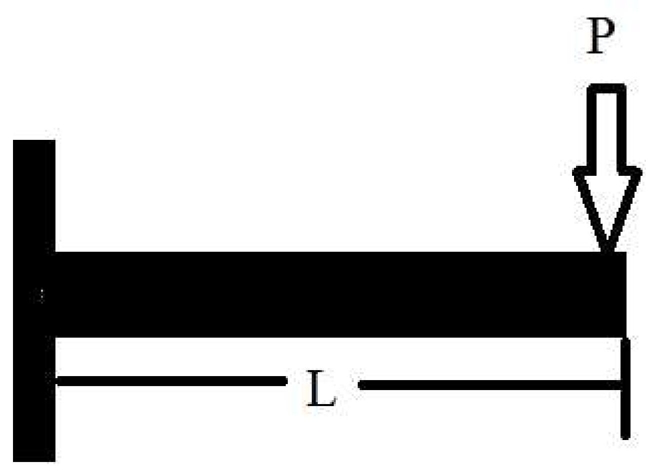


Figure 21: Simply supported beam with point load.

This deformation produces tensile stress in one section of the cross section and compressive stress in the other section. Between these sections there exists a portion where no stresses are developed, which is known as the neutral axis [36].

We consider that the outer/free end of the spring connected to the Solar Panel and its inner end is connected to the satellite body and we induce energy in the spring by rotating its outer end. Due to rotation of the coils by application of force, the torque (M) in the spring will be function of its rotation (ϴ).

$M = M(\theta)$

As we assume the deflection of the spring is linear, so according to Hooke's Law,

$M = K(\theta)$

Where, K is the spring rate.

If we consider that the bending in the spring is linear then according to flexure formula [36],

$$\frac{\sigma_b}{y} = \frac{E}{R} = \frac{M}{I} \text{-----------} (1)$$

Where,

σ_b: is maximum bending stress and its value will be maximum at the farthest away point from the neutral axis of the member (N/mm^2).

M: is the bending moment/Torque produce at the neutral axis of the member (N.mm).

I: is the moment of Inertia for beam cross section (mm^4)

y: is the distance of layer subjected to bending from the neutral axis (mm).

R: is the radius of curvature of the bend beam (mm)

E: is Young's Modulus (N/mm^2)

According to equation (1) the maximum bending stress will be at layer farthest away from the neutral axis. As the proposed cross section of the spring is rectangular with width as "b" and thickness "t" as shown in Figure 22.

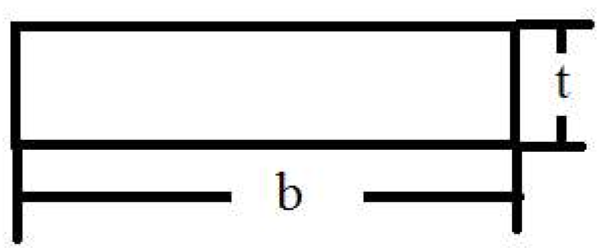


Figure 22: Cross section of Spiral Spring.

From Equation (1)

$$\frac{\sigma_b}{y} = \frac{M}{I}$$

The stress can be written as

$$\sigma_b = \frac{M}{\frac{I}{y}} \text{-------} (2)$$

For a rectangular cross section

$$I = \frac{bt^3}{12}$$

The section Modulus can be written as

$$Z = \frac{I}{y}$$

Equation (2) can be rewritten as

$$\sigma_b = \frac{M}{Z}$$

If we assume:

Space between Coils a: 1 mm

Width of spring b: 10 mm

Thickness of spring t: 1mm

Outer Radius R_e: 20 mm

Inner Radius R_i: 4mm

Angular deflection Θ: 90 Degree

Required Torque M: ?

Spring Rate K: ?

Number of Coils n : ?

From geometry of the spring as shown in Figure 2(a).

L: Length of the spring

E: Modulus of Elasticity of spring material: 193 GPa

σ_y : Yield Stress : 1700 MPa

ρ : Density : 7920 Kg/m^3

Page 20. 42 Table 3-1, 3-13 AssocSpringBarnes

$$L = \frac{\pi \cdot \left(Re^2 - Ri^2\right)}{a + t}$$

$$L = 603.186 \ mm$$

Also

$$L = \pi \cdot n \cdot (Re + Ri)$$

From the above relation the number of coils n=8

The angular deflection θ can be written as [36]

$$\theta = \frac{ML}{EI} ------(3)$$

Where

$$I := \frac{b \cdot t^3}{12} = 0.833 \ mm^4$$

From equation (3) the required torque M can be calculated as

$$M := \frac{\theta \cdot \pi \cdot E \cdot b \cdot t^3}{12 \cdot 180 \cdot L} = 434.028 \ N \cdot mm$$

The spring rate can be calculated as

$$M = K(\theta)$$

$$K := \frac{\pi \cdot E \cdot b \cdot t^3}{12 \cdot 180 \cdot L} = 4.82253 \ N \cdot mm$$

This value is used as an input for Analysis.

The distance from neutral axis to the outer most part of the spring can be written as

$$y_m = \frac{t}{2}$$

$$y := \frac{t}{2} = 0.5 \ mm$$

Therefore the allowable bending stress in the spring according to equation (1) will be

$$\sigma_b := \frac{M}{\frac{I}{y}} = 260.417 \ MPa$$

As we have assumed that the deformation of the spring is within its elastic limit, the applied stress σ_b should be less than with σ_y (Yield Stress) to keep the spring in its elastic limit.

According to ECSS-E-30A, the Factor of Safety (F.O.S) for the designing of yield stress should be 1.25. Also the Margin of Safety (M.O.S) should be greater than 1 as per following relation.

$$M.O.S. = \frac{\sigma_y}{\sigma_b \cdot F.O.S} - 1 > 1$$

For this case

If

$$F.O.S := 1.25$$

$$M.O.S. := \frac{\sigma_y}{\sigma_b \cdot F.O.S} - 1 = 4.412$$

Which complies with the design requirements as per ECSS-E-30 Part 2A standard.

Using above calculations, the design specifications for the spring can be tabulated in Table 5.

Table 5: Design Specification of the spring.

Parameters	Values	Units
Length	603	mm
Width	10	mm
Thickness	1	mm
Loading of Spring (Bending Moment)	435	N-mm
Number of coils	8	-
Outer radius	20	mm
Inner radius	4	mm
Angular deflection	90	deg
Torque spring rate	4.653	N-mm/deg
Cross section of Spring	10 x 1	mm
Elastic Modulus (E)	193	MPa
Yield Stress (σ_y)	1700	MPa
Bending Stress (σ_b)	260	MPa
Density	7920	Kg/m^3
Space between coils	1	mm

As per Table 5, the loading torque (M) is the torque required to be induced in the spring or to be loaded in spring when the solar panels are in stowed configuration. The calculated value is *M= 435 N.mm*. This torque (M) is termed as preloading torque. Let's denote it as M_{PL}

M_{PL}=435 N.mm.

Now we will calculate the torque (M_{DT}) required to deploy the Solar Panel in space. This torque is calculated on the basis of power requirements for the satellite as per Table 4.

Mass of Solar panels m= 4.kg

Length of Solar panel L=1000 mm

Width of Solar panel W=960

Deployment Angle ϴ= 90°

Time to deploy the Solar panel= 3sec

According to the Kinematics of angular motion,

$$\theta = \omega_i t + \frac{1}{2}\, a_\omega t^2$$

Where ϴ is the total angular deflection (90°) and ω_i is the initial velocity (0) for our case.

If we consider the deployment time t as 3 sec, then

$$a_\omega := 2\,\frac{\theta}{t^2} = 0.349\,\frac{rad}{s^2}$$

Also, Torque = Mass Moment of Inertia x Angular acceleration

$$M = I.a_\omega$$

$$I := \frac{1}{12}\cdot m \cdot \left(L^2 + W^2\right) = 0.675\ kg\cdot m^2$$

$$M := I \cdot a_\omega = 235.747\ N\cdot mm$$

M_{DT}= 236 N.mm

The total torque M_T required to deploy the Solar Panels in space will be

$M_T = M_{DT} + M_{PL}$

$M_T = 236 + 435 = 671$ N.mm

3.1.3 Synchronization

After being jettisoned from the Launcher, the satellite remains in a tumbling state for some time. The stabilization of the satellite is achieved by using the available attitude control system (Reaction wheel, Gyros, etc) in the satellite. When this stage has achieved then the solar panels are released using a pyrotechnically driven cutter or thermally controlled released mechanism, which cuts the wire holding the panel in a stowed position with satellite body. The panels need to be deployed smoothly to avoid a mechanical shock at the locking point, i.e 90^o. The angular velocity during deployment of the panel needs to be in a way that the satellite should have as small a shock as possible at the end of the deployment. The deployment speed of the panels for each side should be similar to each other to avoid tumbling of the satellite,. In order to balance the moment of two solar panels during deployment, it is necessary to deploy solar arrays together and insure that both panels deployed at the same time.

3.1.4 Locking

When the solar panels are completely deployed, it is necessary to keep them in the deployed state during operating conditions and the photosensitive area should be exposed to the sun as much as possible to convert the maximum solar energy into electrical power. For this purpose the mechanism should have an irreversible capability to lock the solar panels in the required position Generally the locking mechanism is integrated within the hinge in which a locking pin rotates and is inserted in the slot at the position where locking is required. This pin is designed to provide sufficient force to keep the panel at its required position. The locking pin will slide around a circular part of the hinge during deployment. The circular surface is lubricated with space qualified hard lubricant [34] on which this pin slides. The locking pin should have enough strength to withstand against the shock and vibration loads and be readily unlocked for testing purposes.

3.1.5 Deployment Status

In order to ensure that the deployment process has been completed on the ground or in space it is necessary to place some positioning transducers, such as microswitches, which can provide the status of the deployment mechanism to the ground through telemetry. A micro-switch is attached at the slot of the deployment pin. When the pin enters inside the slot after completion of deployment the switch circuit will close and telemetry is sent confirming that deployment is complete. The depth the pin is inserted inside the slot is very important; if it is too large the unlocking of pin in the ground during stowing of panels after deployment tests will be very difficult, and if it is too small then the hinge will not be properly locked.

3.2 Design Requirements

The design of the hinge mechanism does not only include the material selection but also includes analysis and verification as a part of process development. The first step of the design process is to generate the requirements. The design requirements are derived from the Functional Requirements as defined in Section 2.1, which describe the variety of technical characteristics of the system. This may include the performance and operational requirements.

The main performance requirements for the mechanism [32] are:

- To perform deployment
- Physical support to panels
- Deployment status
- Operating Speed
- Operating Life
- Pointing Accuracy
- Holding Duration

The mechanism is required to withstand for the following environments from its assembly until end-of-life operation in space:

a) Operation on Ground: Assembly and Integration, Transportation of Satellite;

b) Launch and Separation: Pressure drop in Space, Static, Acoustic, Vibration load from Launcher, Temperature gradient in space, outgassing due to high vacuum in space; and

c) Operation in Orbit: Thermal Loads due to Temperature, Stringent Space Environment of radiation and Atomic Oxygen. Micro Gravity, Disturbance due to space debris.

The above-mentioned environments cause different types of loads on the satellite structure. These loads are generated from the Launcher and pass through the satellite structure and consequently to the solar arrays through the deployment mechanism. In order to fulfil its functional requirements, the mechanism has the mechanical interface with the structure as shown in Figure 23. The design of each subsystem has an impact on the other subsystems, however there is a major impact of both structure and mechanisms on each other as both have the some common design requirements like minimum weight, vibration interaction and material selection.

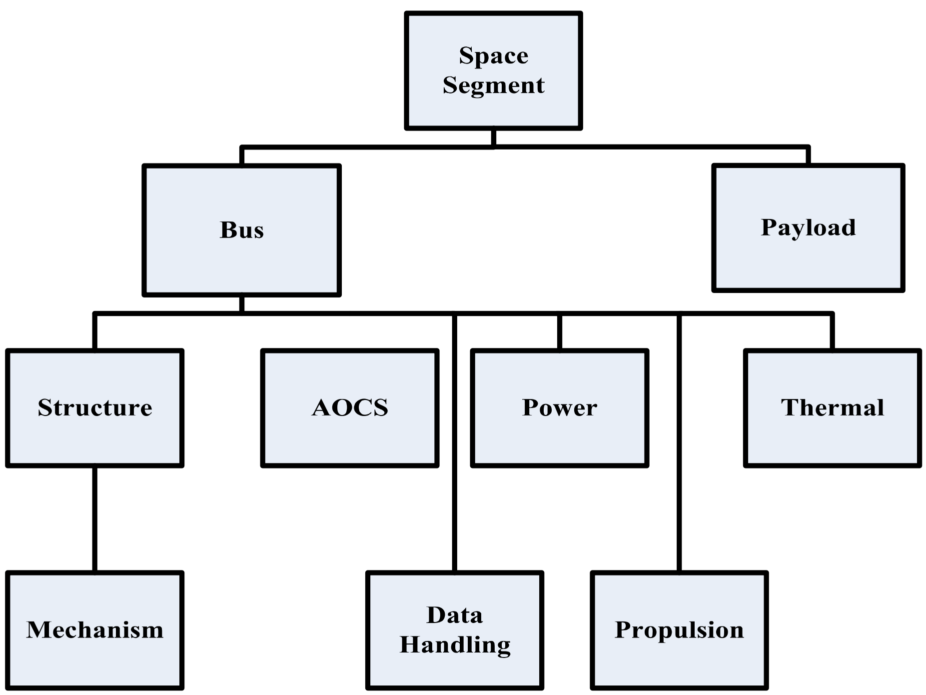


Figure 23: Space Segment interface with Satellite Subsystem.

Due to these loads, significant mechanical stresses are developed on each subsystem of the satellite depending on the interface of each subsystem with the rest of the satellite.

The mechanism requires to comply with the specific Loads, Temperature, Stiffness and the Mass, which are generally derived from the launcher and the satellite structure. Out of these requirements the requirement of the load termed as the Quasi Static Load (Loads that occur at the same level throughout the whole satellite) is the most important as the other parameters like temperature, stiffness and the mass depend on the mechanical properties of the material proposed for the deployment mechanism. The most severe loads experienced by the mechanism are during launch, stage separation and payload separation. The design loads are defined by the system, which considered the different factor of safety as per standard ECSS-E-30 Part 2A.

The factor of safety is defined as per project design approach like Engineering, Qualification, Flight, and Proto Flight Model. The approach to define the load requirements is shown in Figure 24.

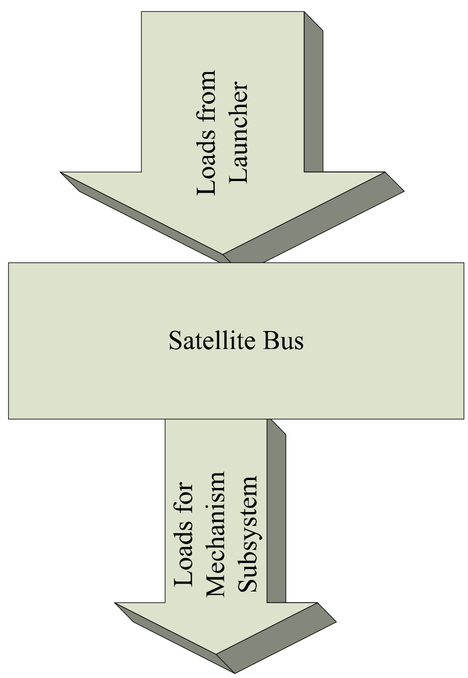


Figure 24: Design Limit Loads for Mechanism.

The determination of loads is a continuous iterative process and it matures as the different milestones of the project like PDR, CDR etc. are achieved. Estimation of loads for the payload is an iterative process. At the start of the design process preliminary design loads are either estimated or defined on the basis of previous programs considering different limiting factors. This is very helpful in initial sizing of the structure elements and performing the initial load analysis. The preliminary resulting loads are analysed and based on these loads the design may require to be adjusted to fulfil the other requirements of stiffness, mass and temperature of the mechanism. In Table6, we have defined the preliminary design requirements for the mechanism.

Table 6: Design Requirements.

Quasi-static Load Cases Requirements			
Direction	X-direction [G]	Y-direction [G]	Z-direction [G]
Displacement	Fixed in all DOF at mounting point	Fixed in all DOF at mounting point	Fixed in all DOF at mounting point
Quasi-Static Load	28	28	68

Table 7: Frequency Requirements.

Natural Frequency Requirements			
Component	X-direction [Hz]	Y-direction [Hz]	Z-direction [Hz]
Requirement	>120	>120	>120

4 STRUCTURAL ANALYSIS OF THE HINGE DEPLOYMENT MECHANISM

This chapter presents the Structure design and Finite Element Analyses (FEA) of the Solar Panel Hinge Deployment mechanism. The FEA includes Natural Frequency, Quasi-static, Dynamic Sinusoidal vibration and Random vibration analyses. The acceleration response calculated using random analysis are further used to perform Random-Static analysis, in which accelerations are used as inertial loads to calculate displacements, stresses and constraint forces.

Key parameters taken for building the structure model and calculating the results of the analysis are described in this chapter. The structure was designed on PTC Creo Parametric software and structural analysis was performed using MSC Patran and Nastran 2012.

4.1 Structural Requirements

Four Deployment Hinges will be mounted on the outer panels of low Earth orbiting satellites. First the fundamental frequency of Hinge should be greater than 120Hz, illustrated in 9.

Table 8: Natural Frequency Requirements.

Natural Frequency Requirements			
Component	X-direction [Hz]	Y-direction [Hz]	Z-direction [Hz]
Requirement	>120	>120	>120

- Three Quasi-static load cases (Launch Environment 68g longitudinal and 28g lateral)

Table 9: Quasi-Static Loads.

Quasi-static Load Cases Requirements			
Direction	X-direction [G]	Y-direction [G]	Z-direction [G]
Displacement	Fixed in all DOF at mounting point	Fixed in all DOF at mounting point	Fixed in all DOF at mounting point
Quasi-Static Load	28	28	68

- Three Sinusoidal load cases (Loads are derived from Long March 2D Satellite Launch Vehicle).

Table 10: Sinusoidal Loads.

G's Level for Panel (In-Plan)		G's Level for Panel (Out-Plan)	
Frequency (Hz)	G levels	Frequency (Hz)	G levels
5	1.525	5	2.225
20	1.525	30	2.225
35	7.775	70	6.625
45	7.775	80	19.625
60	1.525	90	19.625
100	1.525	100	3.5625

- Three Random load cases (Loads are derived from Long March 2D Satellite Launch Vehicle).

Table 11: Random Loads.

Frequency Range	Acceleration
20~80	3 dB/oct
80~700	0.15 g^2/hz
700~2000	6 dB/oct

4.2 Mass

The cumulative mass of the hinge assembly considered for the structural analysis is as under:

S. No.	Component	Mass
1	Deployment Hinge	110 g
2	Solar Panel	4.5 kg
	Total	4.61 kg

4.3 Finite Element Model

The finite element model (FEM) is shown in Figure 25. It has been developed using the software MSC. Patran. The Model consists of 6695 grid points (nodes) which are meshed together with 24504 TET4 elements.

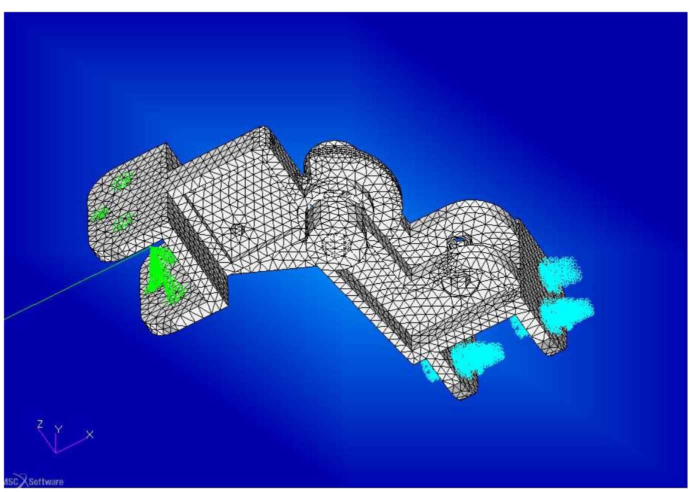


Figure 25: FEM Model of Solar Panel Deployment Hinge.

Figure 25 shows the mounting points of Deployment Hinge. The blue coloration shows the fixed constraints.

In Figure 26, the weight of a typical space-grade $1m^2$ solar panel is considered in the form of point mass at the estimated COG of the panel to keep the FEM model simple enough for calculations and simulations. The point mass is attached to the Deployment Hinge by a Rigid Body Element (RBE3).

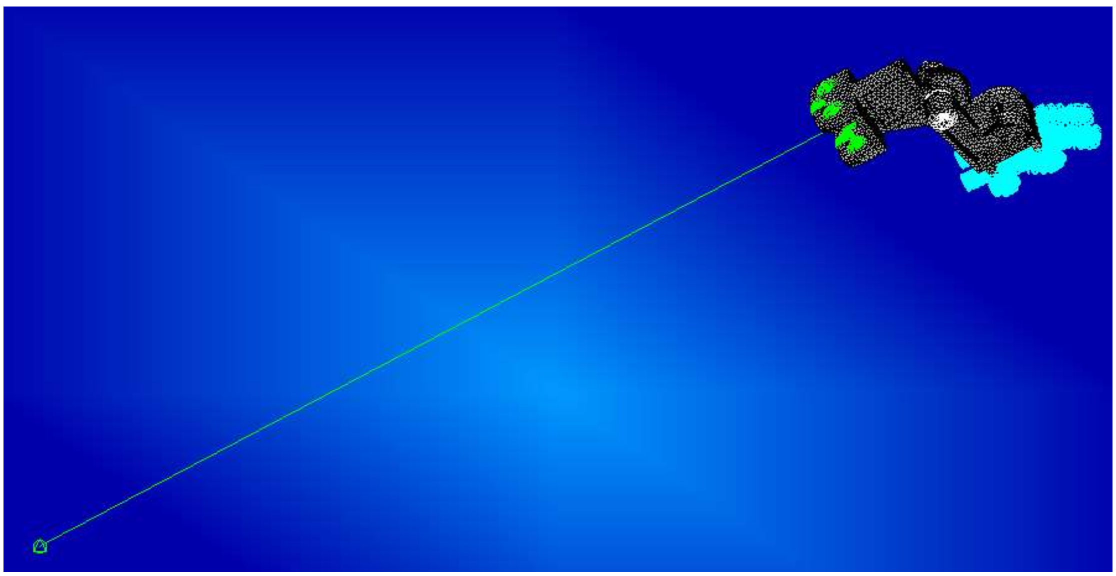


Figure 26: Solar Panel's Point Mass (RBE3 Constraint).

4.4 Modal Analysis

The deployment hinges will be mounted on the MX (Minus X) and PX (Positive X) panel of the satellite. The purpose of modal analysis is to compute the Eigen values and associated Eigen vectors of the structure.

Results of the first ten modes are given below in Table 12. The modal effective mass fractions are given in Appendix A.

Table 12: Normal Mode Analysis results.

Mode	Frequency (Hz)	Mode Shape	Mode	Frequency (Hz)	Mode Shape
1st	247.41	On Rotating part	6th	5803.5	On Rotating part
2nd	673.53	On Rotating part	7th	7272.5	On Rotating part
3rd	2123.5	On Rotating part	8th	7911.7	On Rotating part
4th	2248.2	On Rotating part	9th	8525.0	On Rotating part
5th	2711.8	On Rotating part	10th	10487.0	On Rotating part

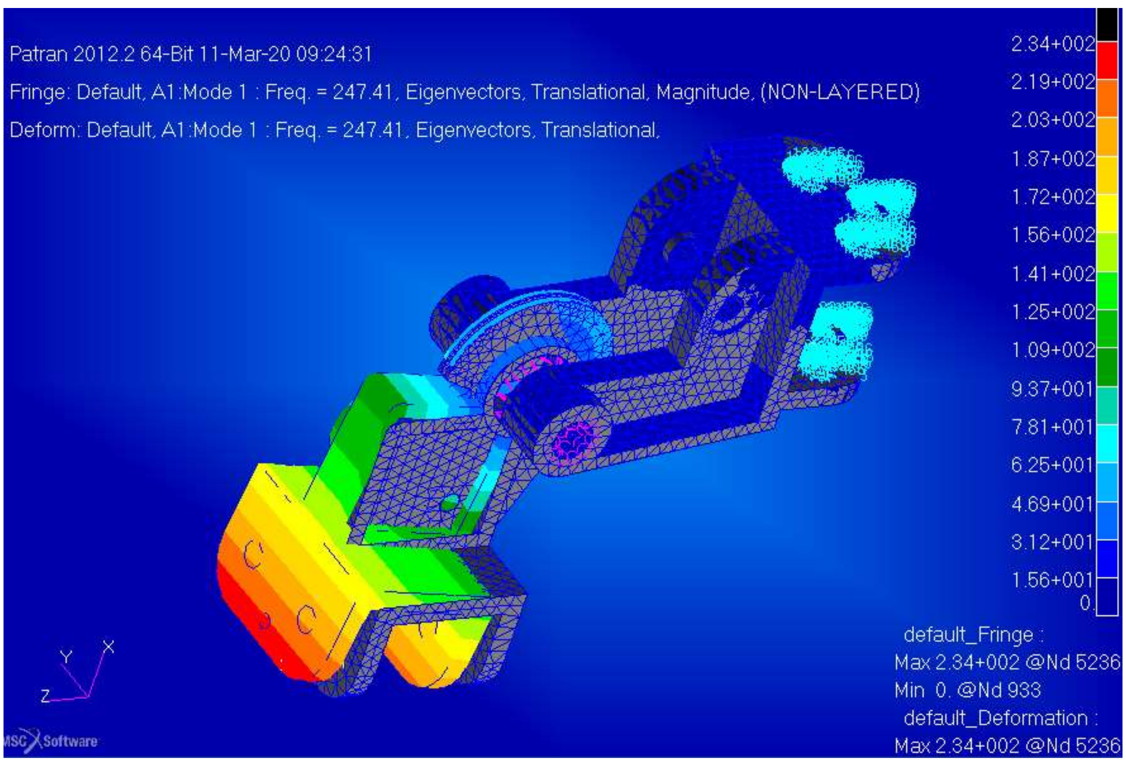


Figure 27: 1st Modal Frequency 247.1 Hz (Natural Frequency of Hinge).

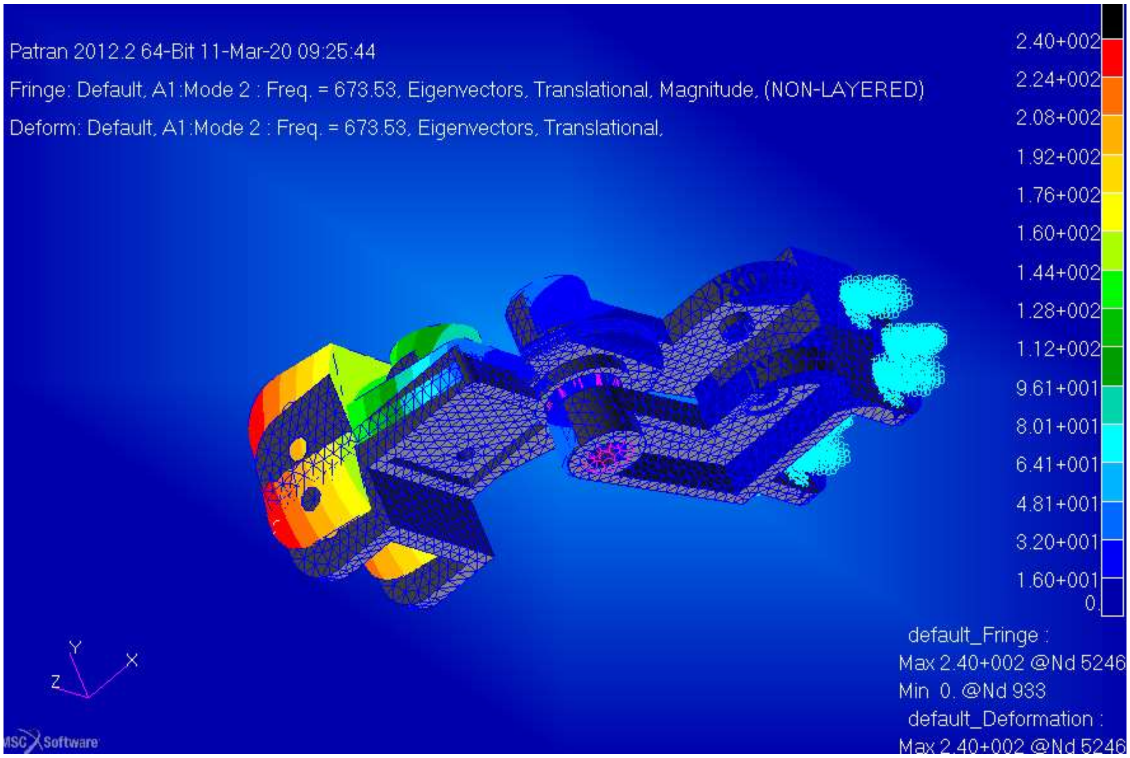


Figure 28: 2nd Modal Frequency 673.53 Hz.

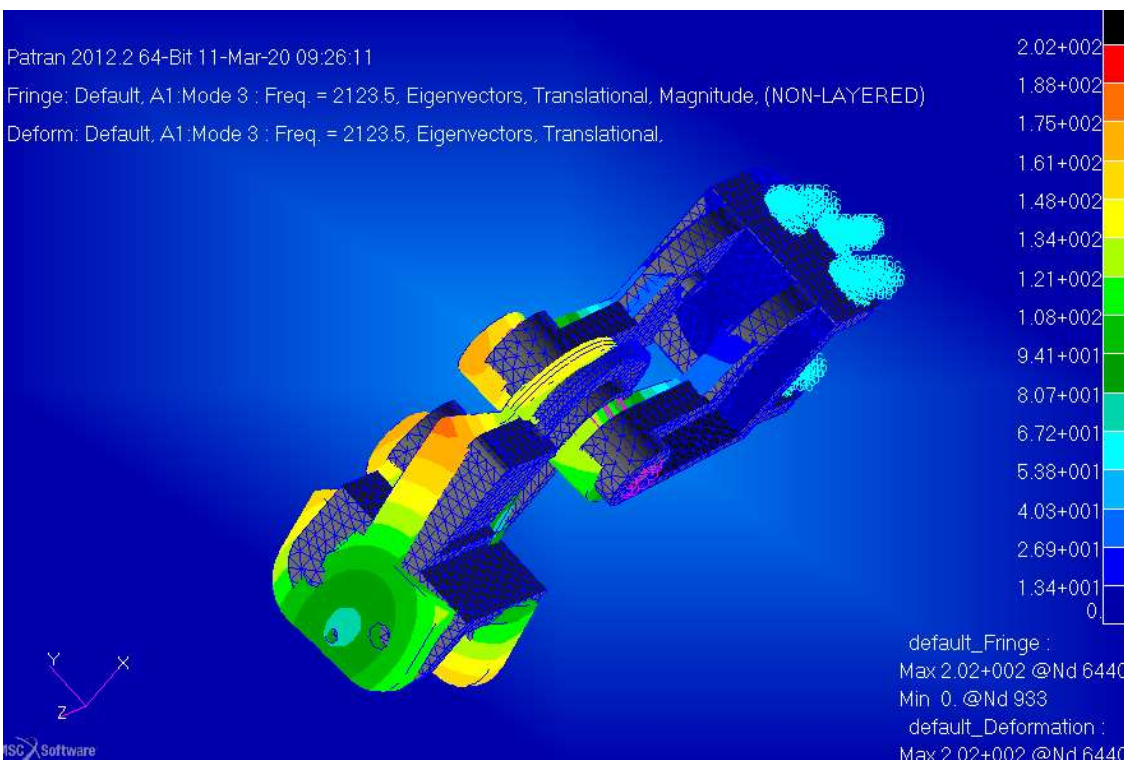


Figure 29: 3rd Modal Frequency 2123.5 Hz.

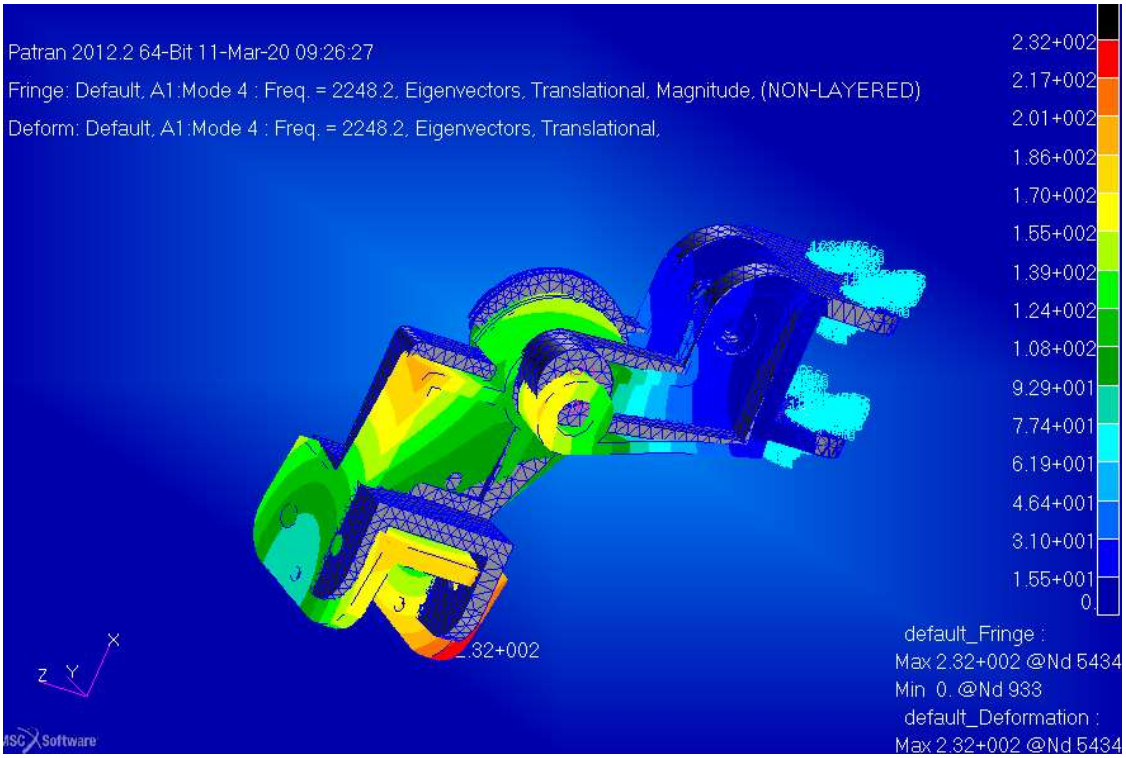


Figure 30: 4th Modal Frequency 2248.2 Hz.

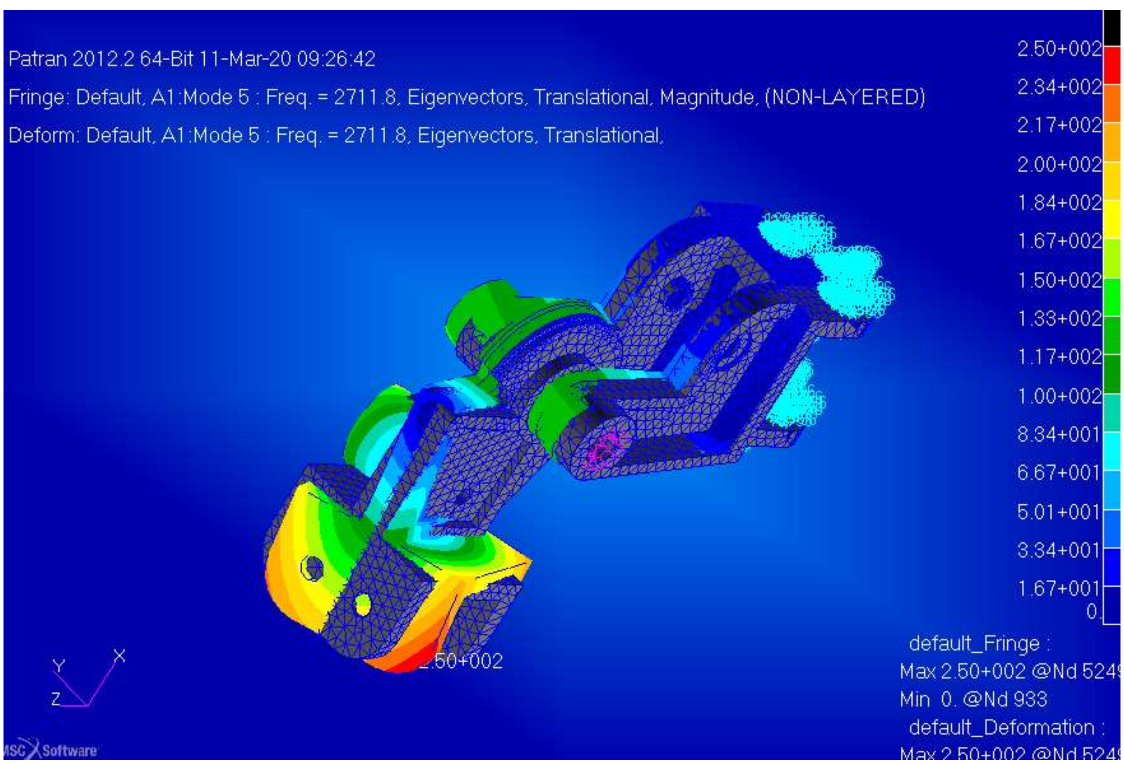


Figure 31: 5th Modal Frequency 2711.8 Hz.

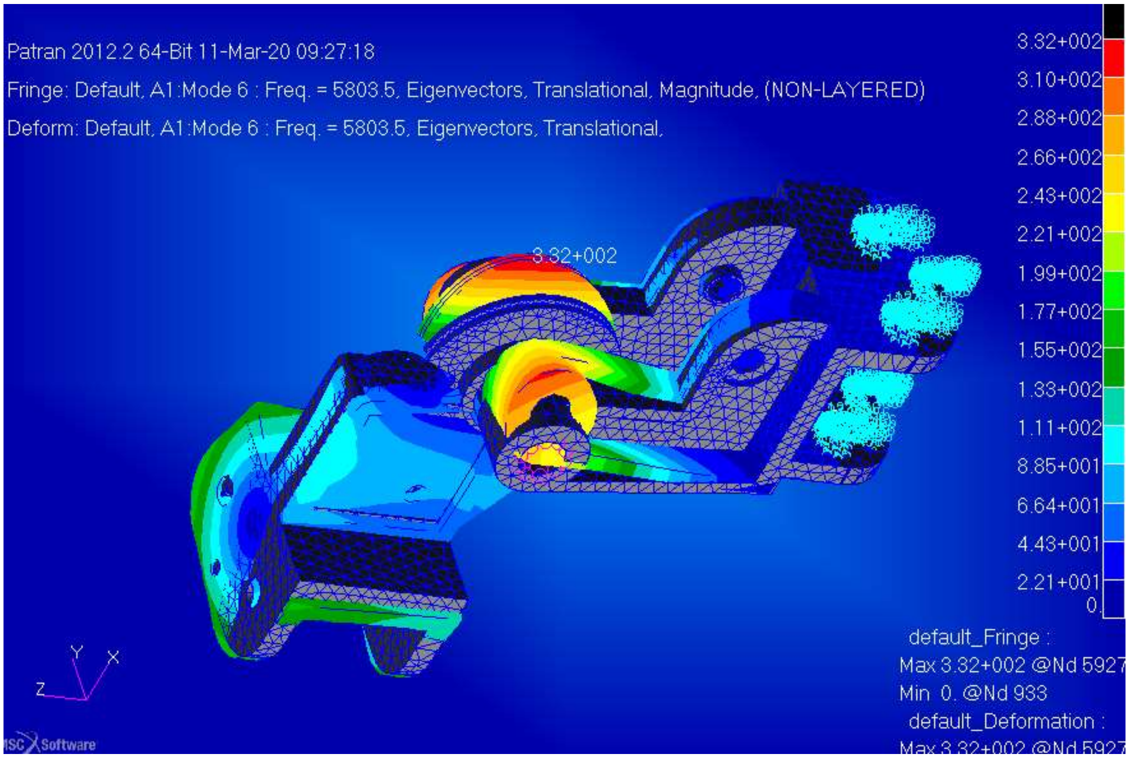


Figure 32: 6th Modal Frequency 5803.5 Hz.

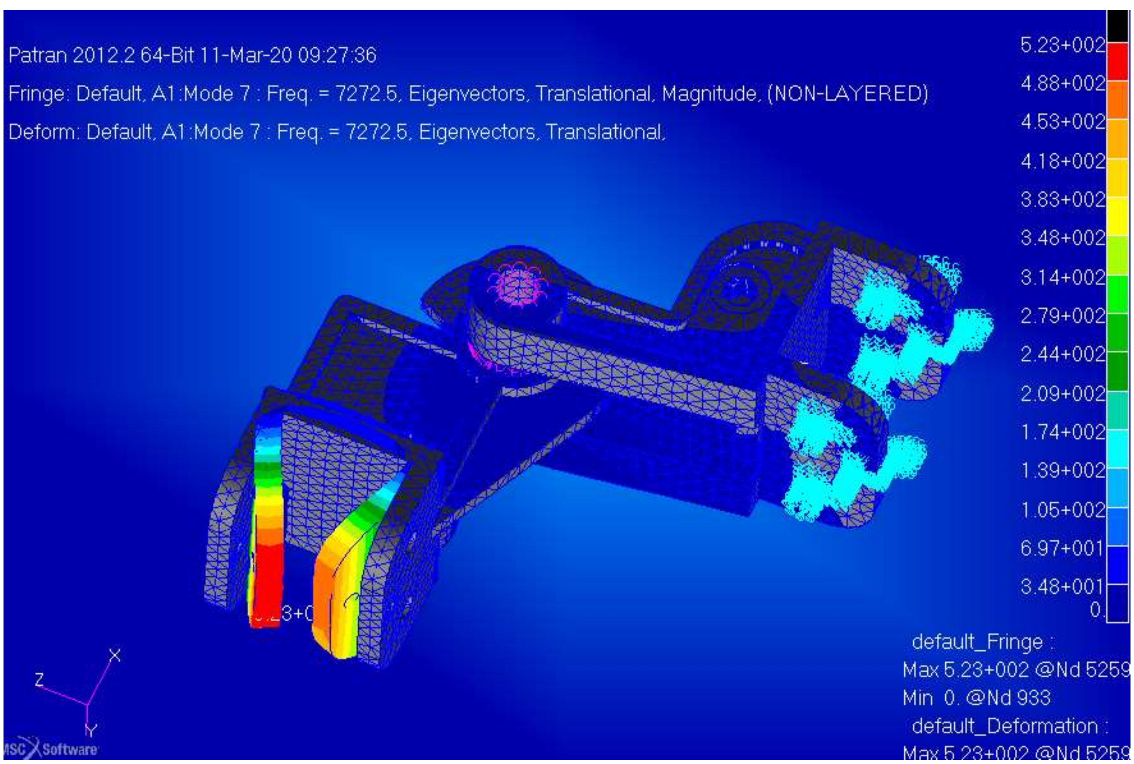


Figure 33: 7th Modal Frequency 7272.5 Hz.

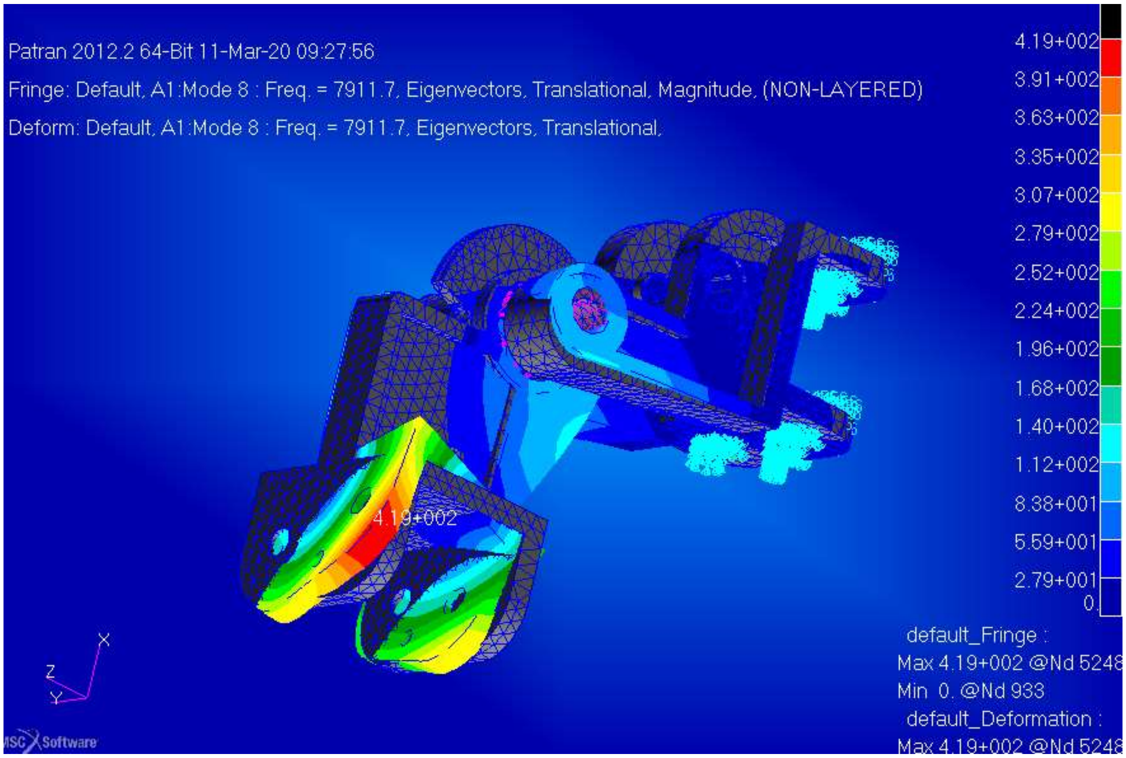


Figure 34: 8th Modal Frequency 7911.7 Hz.

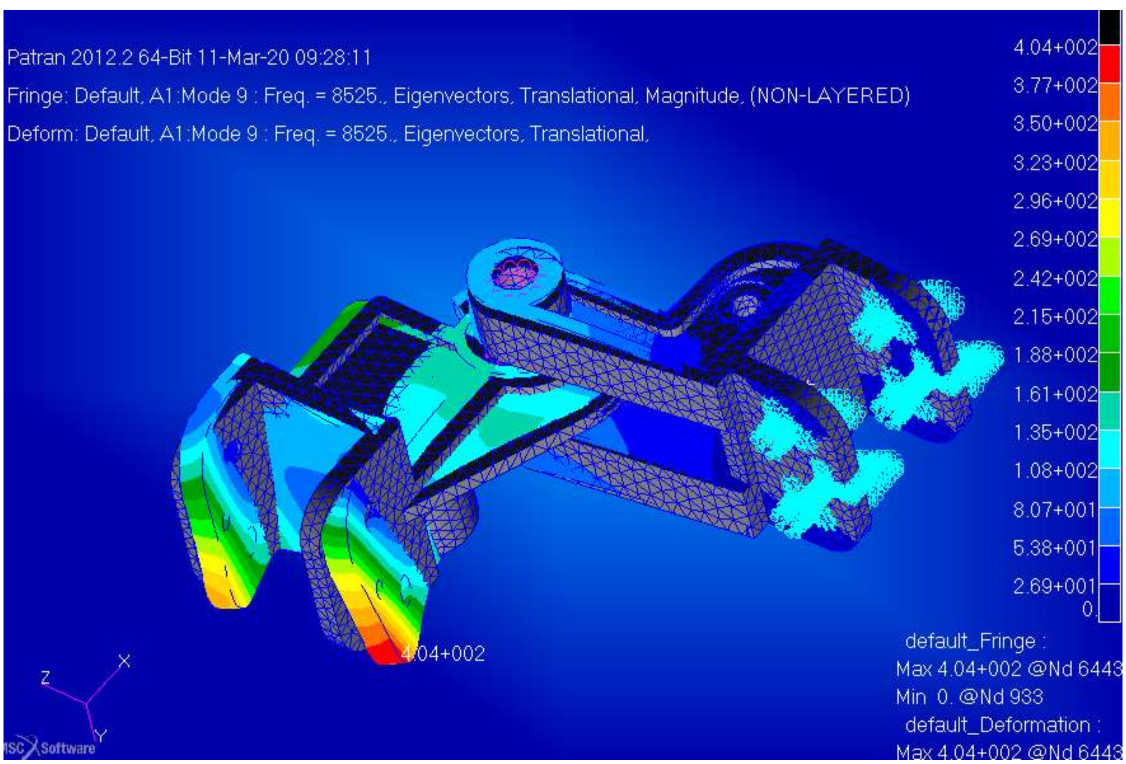


Figure 35: 9th Modal Frequency 8525.0 Hz.

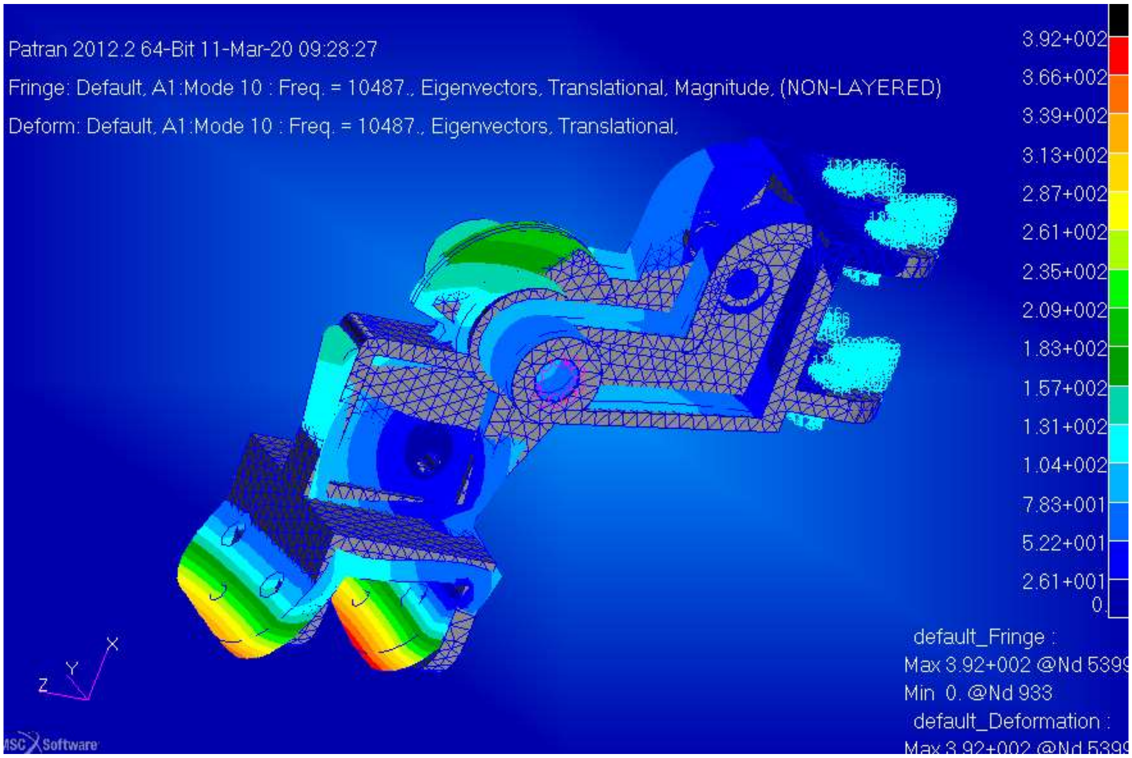


Figure 36: 10th Modal Frequency 10487.0 Hz.

4.5 Dynamic Sinusoidal Analysis

Dynamic analysis has been performed in all three directions (X, Y and Z). The results of acceleration response, deformations and MPC constraint forces are calculated from the FEM Model.

4.5.1 Sinusoidal Analysis in the X-Direction

Figure 377 and

Figure 38 show the input sinusoidal load and output response graphs of the Sinusoidal Analysis in the X-Direction. Maximum acceleration and MPC constraint force response of Deployment Hinge is found at 100Hz which are shown in Table 13.

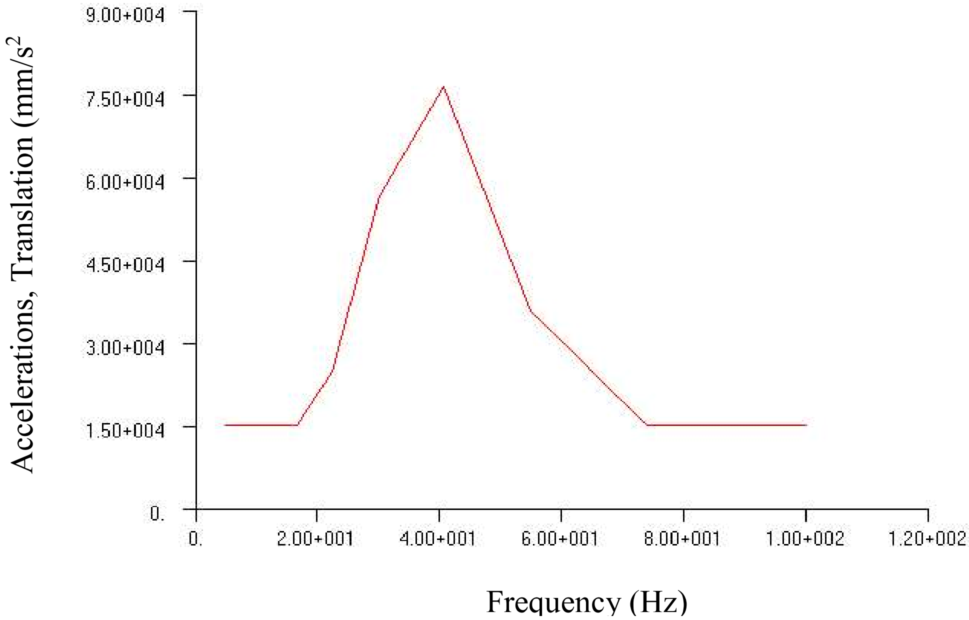


Figure 37: Input Sinusoidal Analysis in X-Direction.

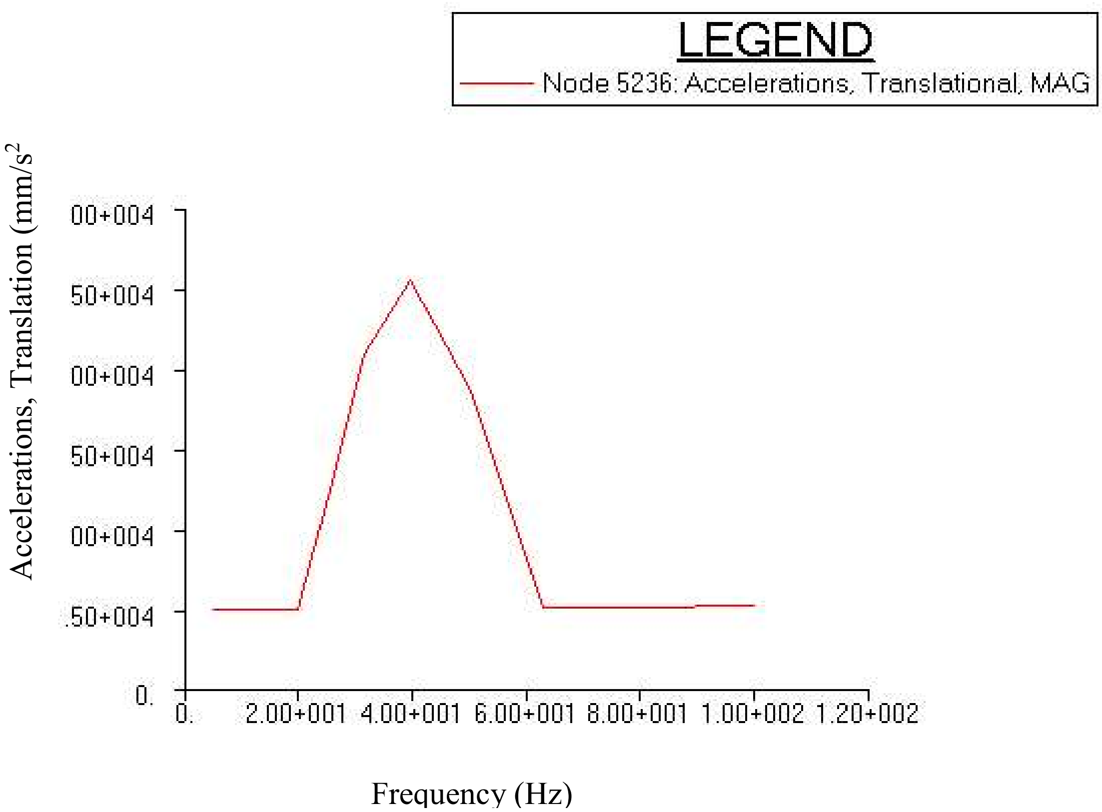


Figure 38: Output Response in X-Direction.

Table 13: Deployment Hinge Dynamic Analysis X-Direction.

Results for Dynamic Analysis – X direction	
Response Parameter	100 Hz
Acceleration (g)	1.61
Deformation (mm)	0.0399
Max. Constraint Force (N)	2.51

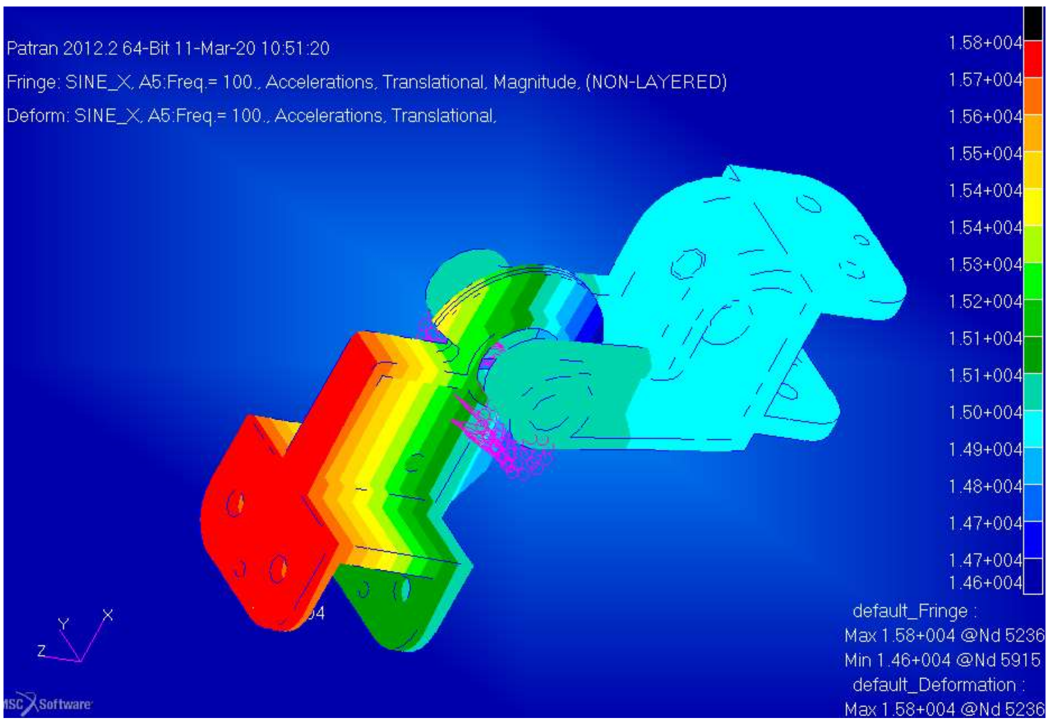


Figure 39: Acceleration in Deployment Hinge.

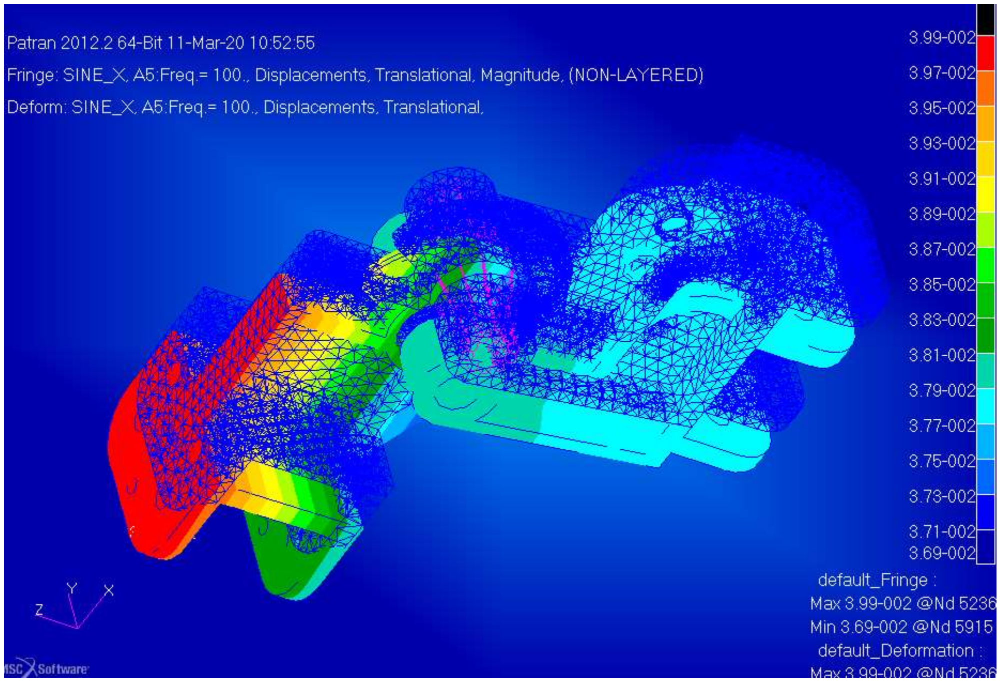


Figure 40: Deformation in Deployment Hinge.

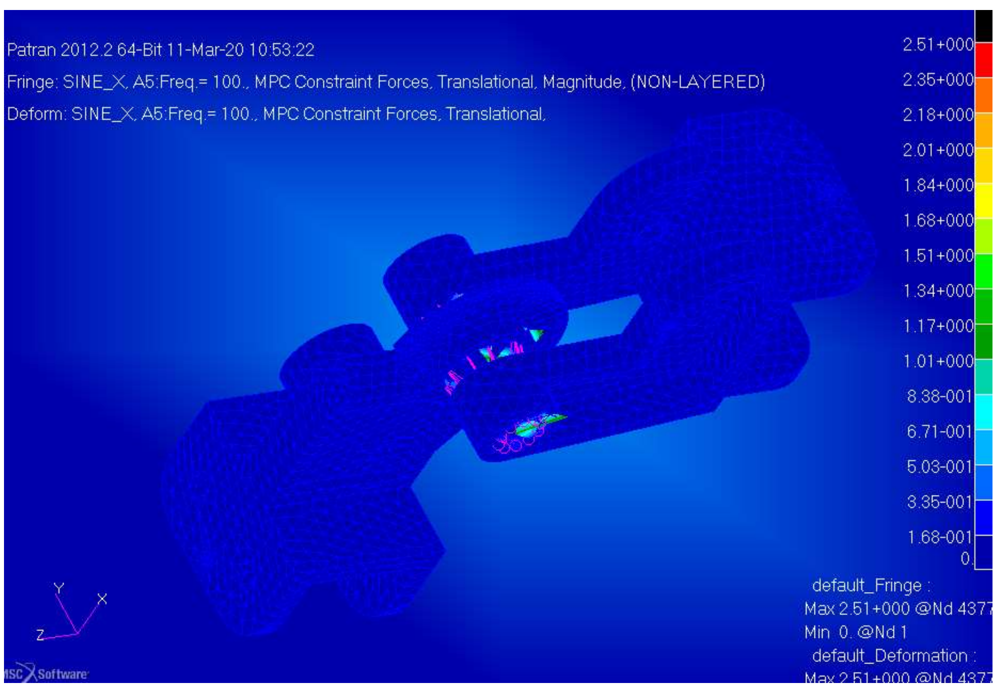


Figure 41: MPC Constraint Forces in Deployment Hinge.

4.5.2 Sinusoidal Analysis in the Y-Direction

Figure 42 and Figure 43 show the input sinusoidal load and output response graphs of Sinusoidal Analysis in the Y-Direction. Maximum acceleration, deformation and MPC constraint force response of deployment hinge is found at 100Hz which are shown in Table 14.

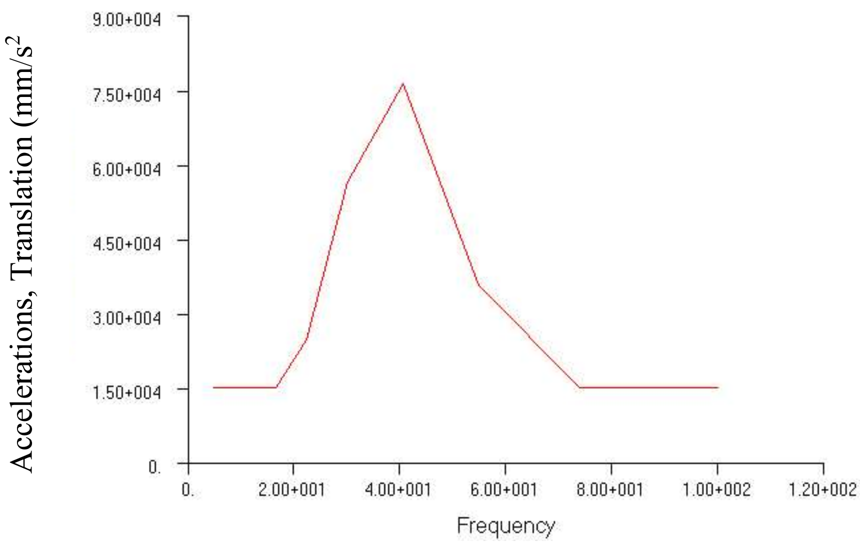


Figure 42: Input Sinusoidal Analysis in Y-Direction.

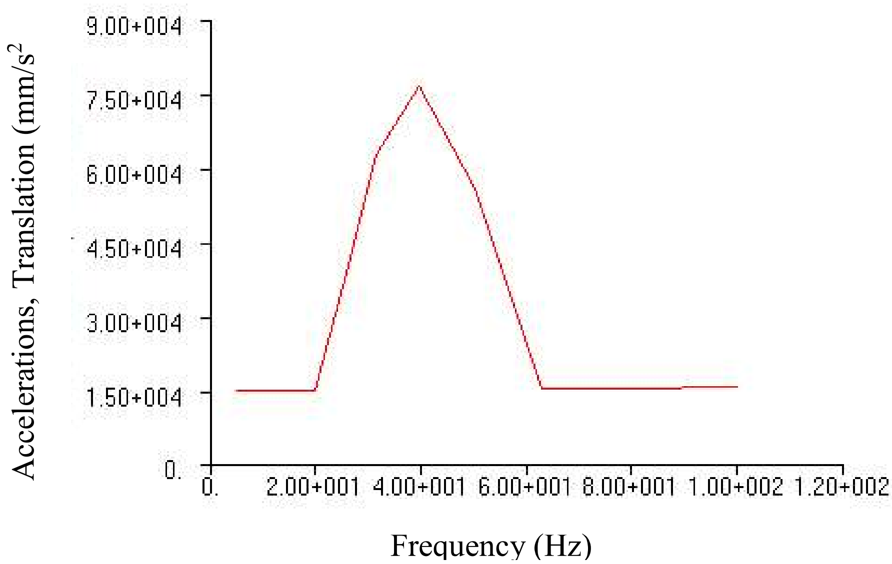


Figure 43: Output Response in Y-Direction.

Table 14: Deployment Hinge Dynamic Analysis Y-Direction.

Results for Dynamic Analysis – Y direction	
Response Parameter	Y Direction at 100 Hz
Acceleration (g)	1.58
Deformation (mm)	0.0392
Max. Constraint Force (N)	0.0881

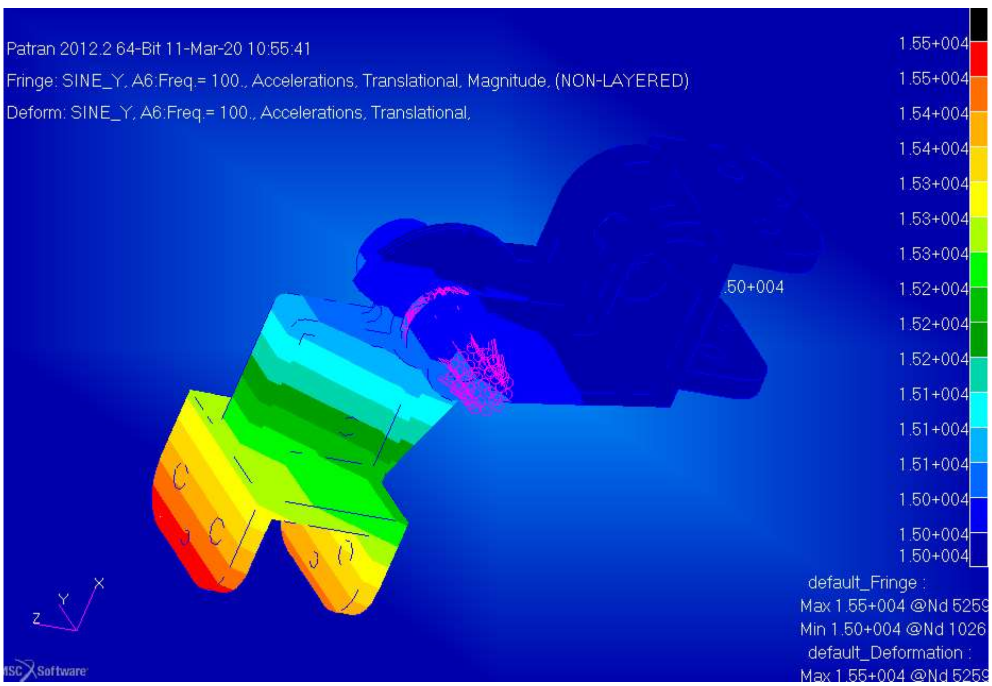


Figure 44: Acceleration in Deployment Hinge.

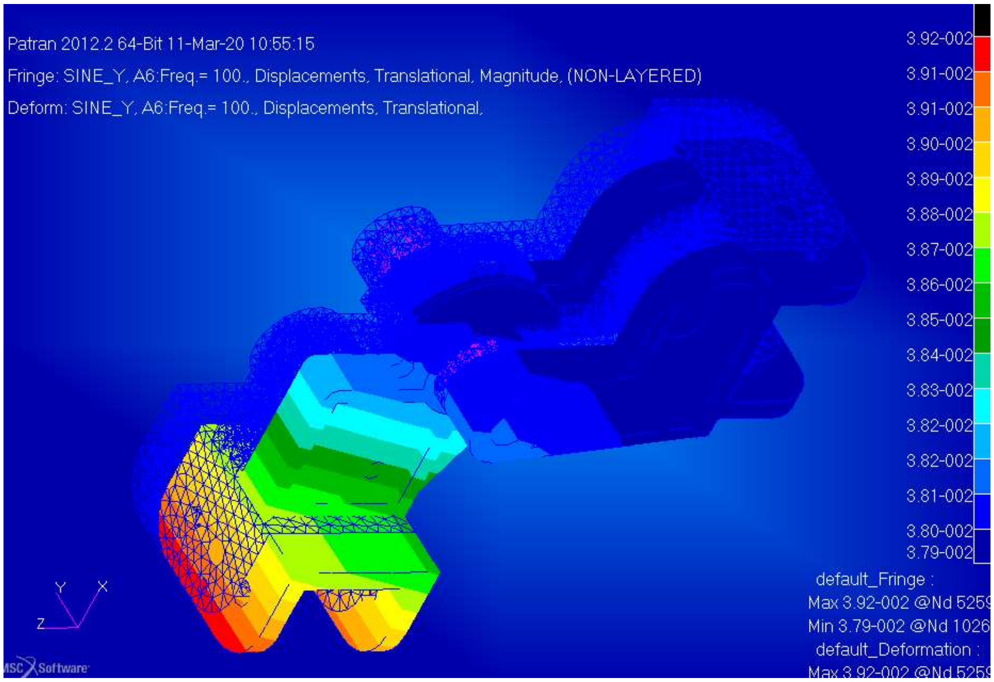


Figure 45: Deformation in Deployment Hinge.

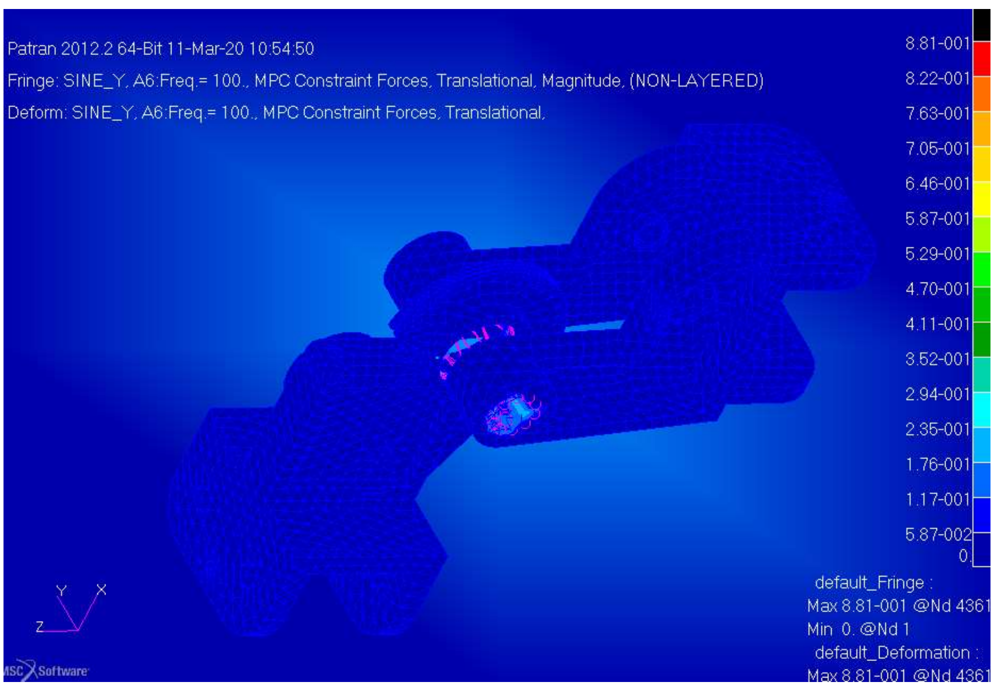


Figure 46: MPC Constraint Forces in Deployment Hinge.

4.5.3 Sinusoidal Analysis in the Z-Direction

Figure 47 and Figure 48 show the input sinusoidal load and output response graphs of Sinusoidal Analysis in the Z-Direction. Maximum acceleration, deformation and MPC constraint force response of deployment hinge is found at 100Hz which are shown in Table 15.

.

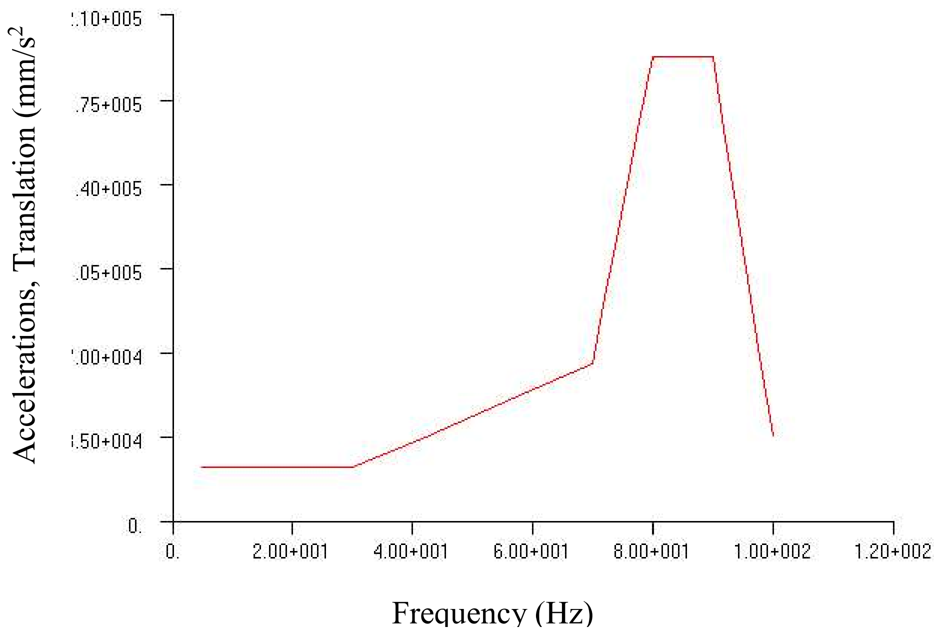


Figure 47: Input Sinusoidal Load in Z-Direction.

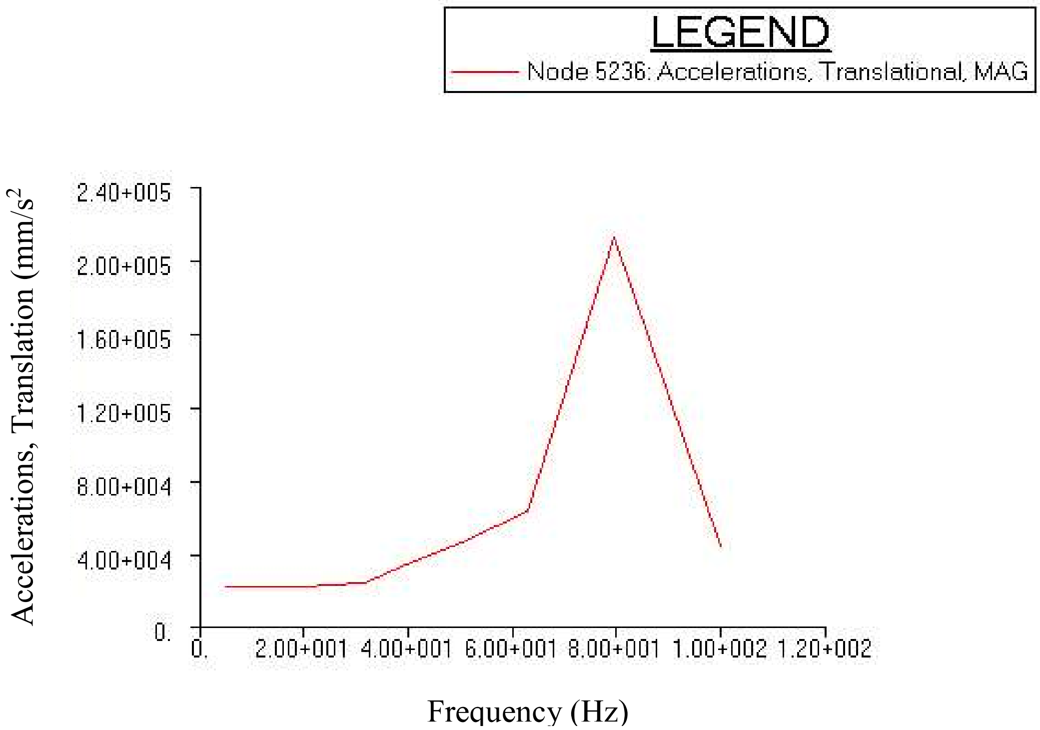


Figure 48: Output Response in Z-Direction.

Table 15: Deployment Hinge Dynamic Analysis Z-Direction.

Results for Dynamic Analysis – Z direction	
Response Parameter	at 100 Hz
Acceleration (g)	4.454
Deformation (mm)	0.111
Max. Constraint Force (N)	16.3

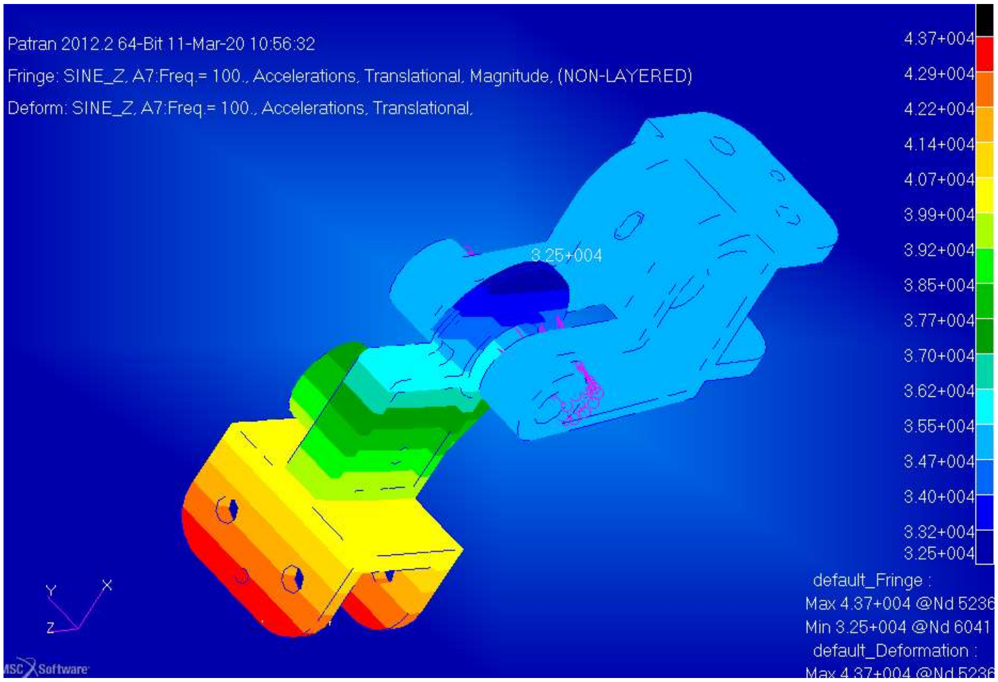


Figure 49: Acceleration in Deployment Hinge.

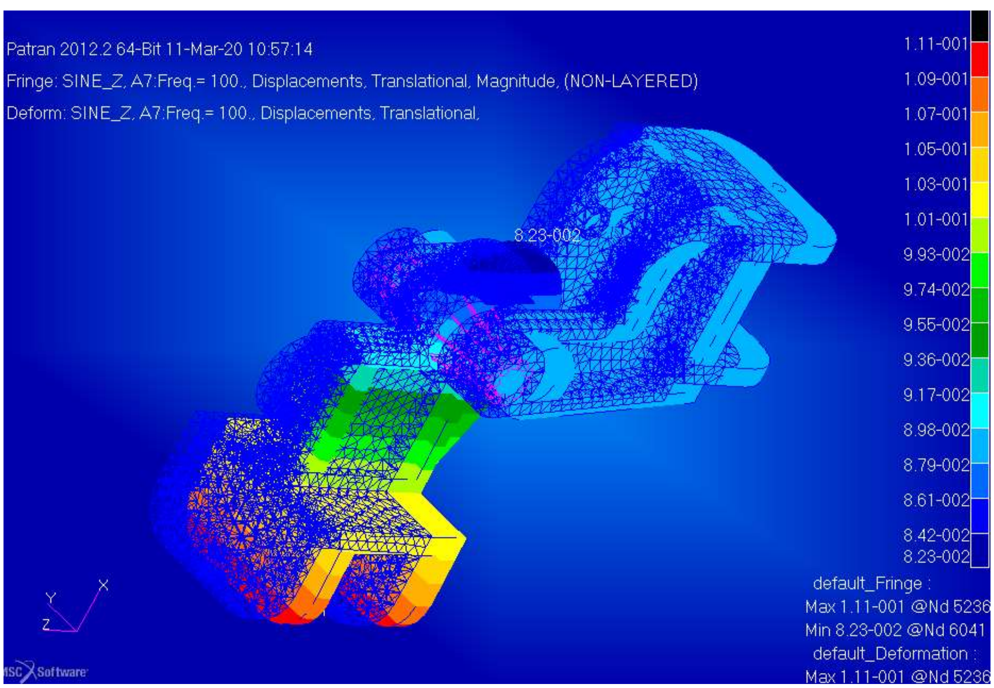


Figure 50: Deformation in Deployment Hinge.

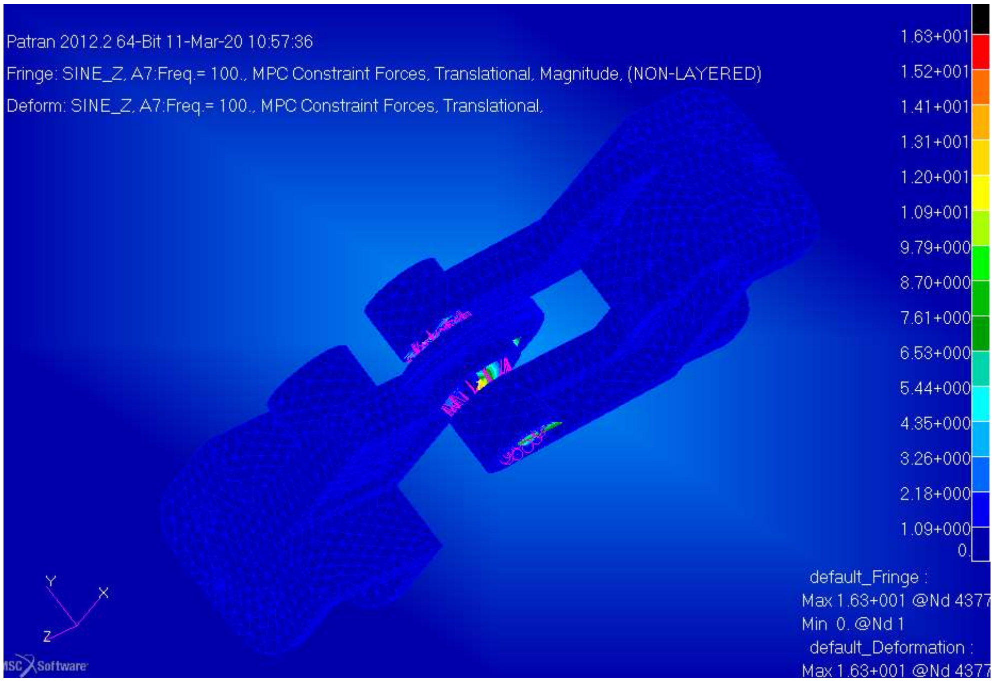


Figure 51: MPC Constraint Forces in Deployment Hinge.

4.6 Quasi-Static Analysis

Results for maximum stress and deformations are listed below in against static loads

Table 16against static loads

Table 16: Results of the Static Analysis.

Load Level (g)	The max. stress and deformation of the Deployment Hinge		Max. Constraint force (N)
	The max. stress (MPa)	The max. deformation (mm)	
a_x=28g	5.61	0.0611	38.1
a_y=28g	3.53	0.0244	15.7
a_z=68g	38.3	0.386	261

4.6.1 Quasi-Static Analysis in the X-Direction

Figure 52 to Figure 54 show the Von-Mises stresses, displacement and MPC constraint forces in the deployment hinge in the X-Direction respectively.

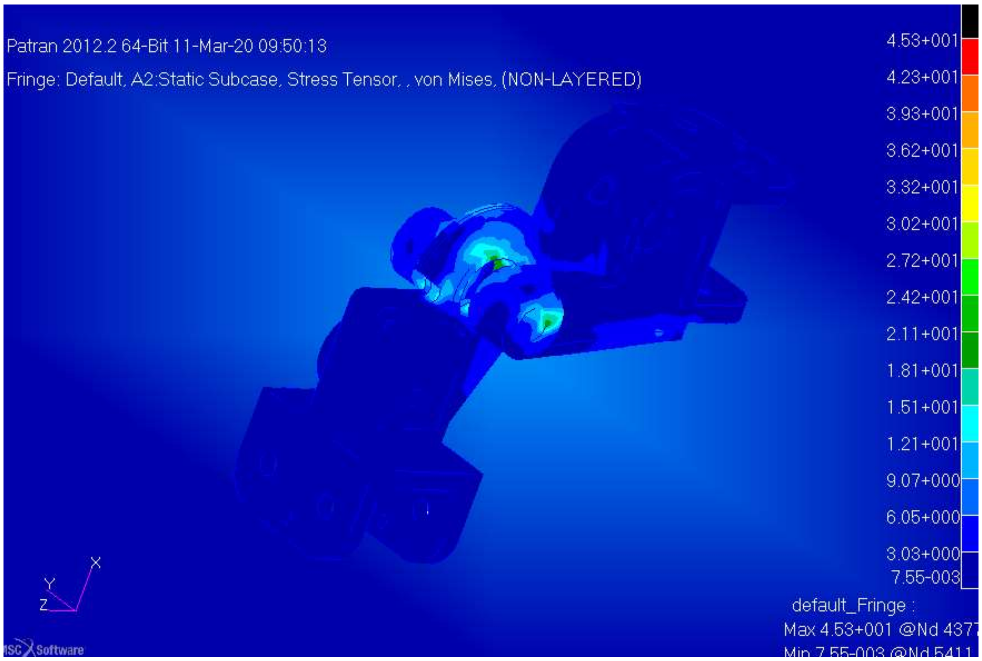


Figure 52: Stress in Deployment Hinge due to Static Load along X-Direction.

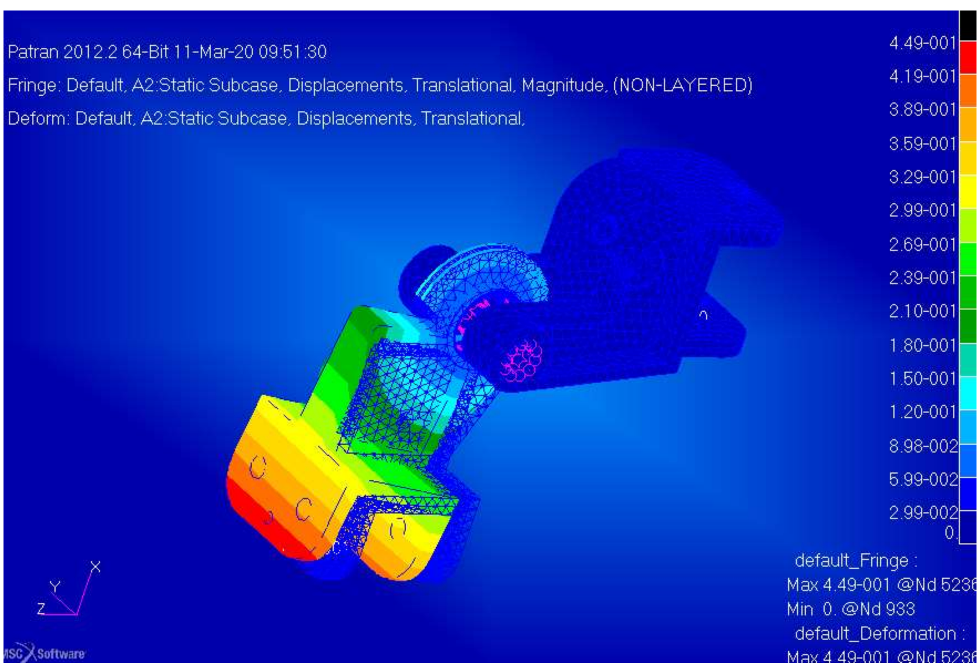


Figure 53: Deformation in Deployment Hinge due to Static Load along X-Direction.

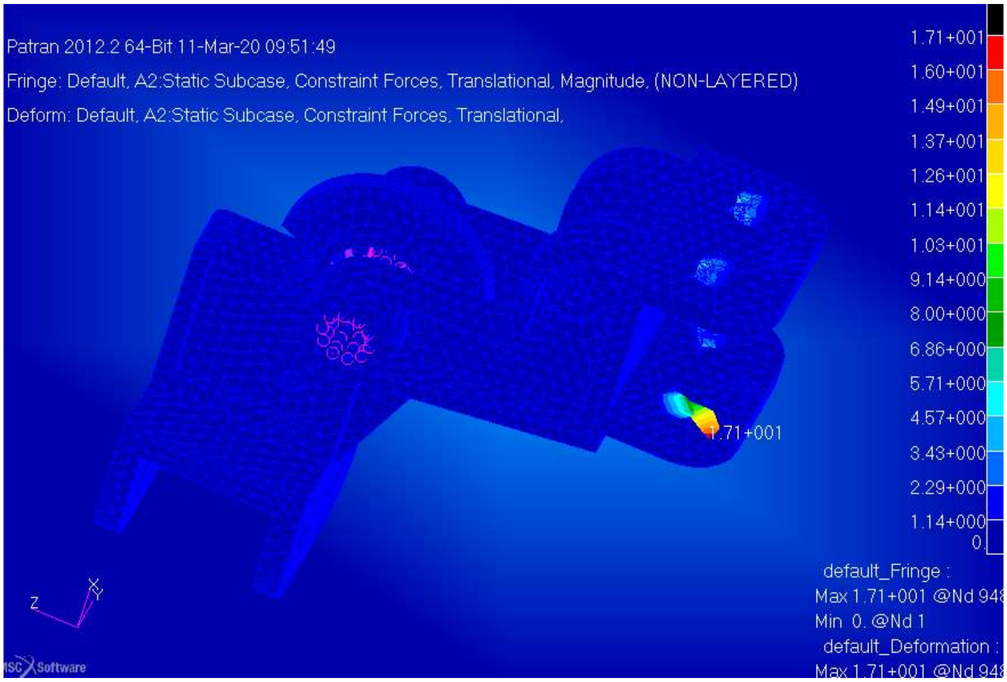


Figure 54: MPC Constraint Forces along X-Direction.

4.6.2 Quasi-Static Analysis in Y-Direction

Figure 55toFigure 57 show the Von-Mises stresses, displacement and MPC constraint forces in the deployment hinge in the Y-Direction respectively.

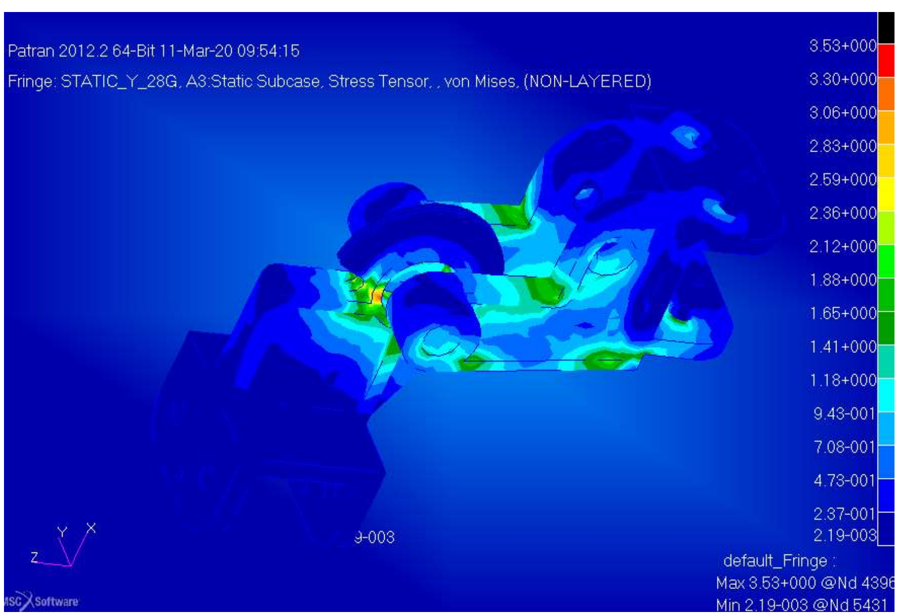


Figure 55: Stress in Deployment Hinge due to Static Load along Y-Direction.

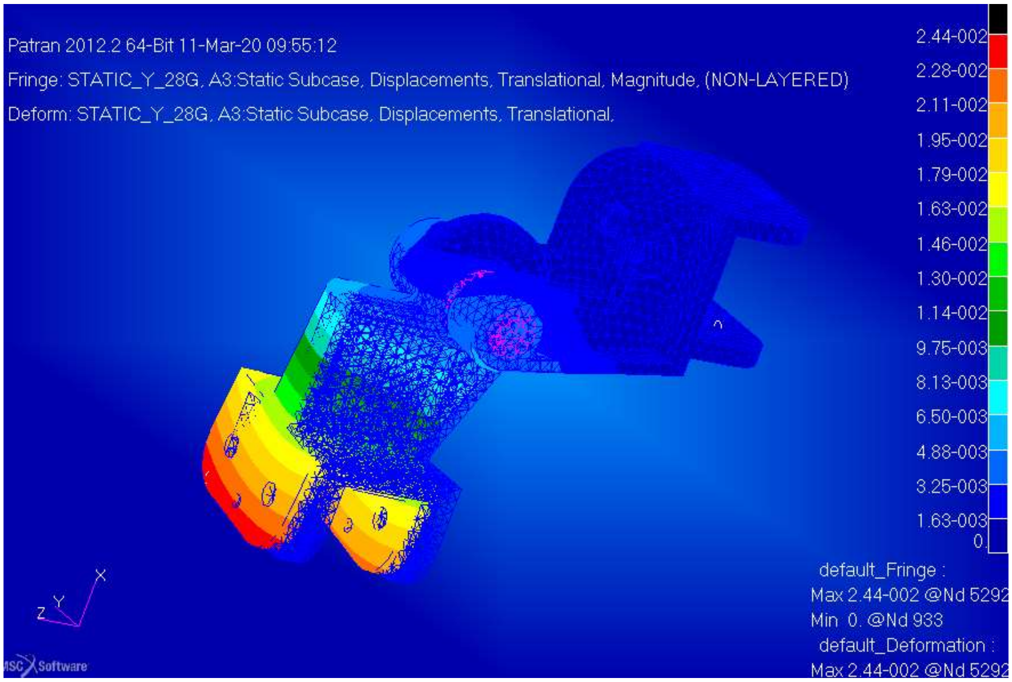
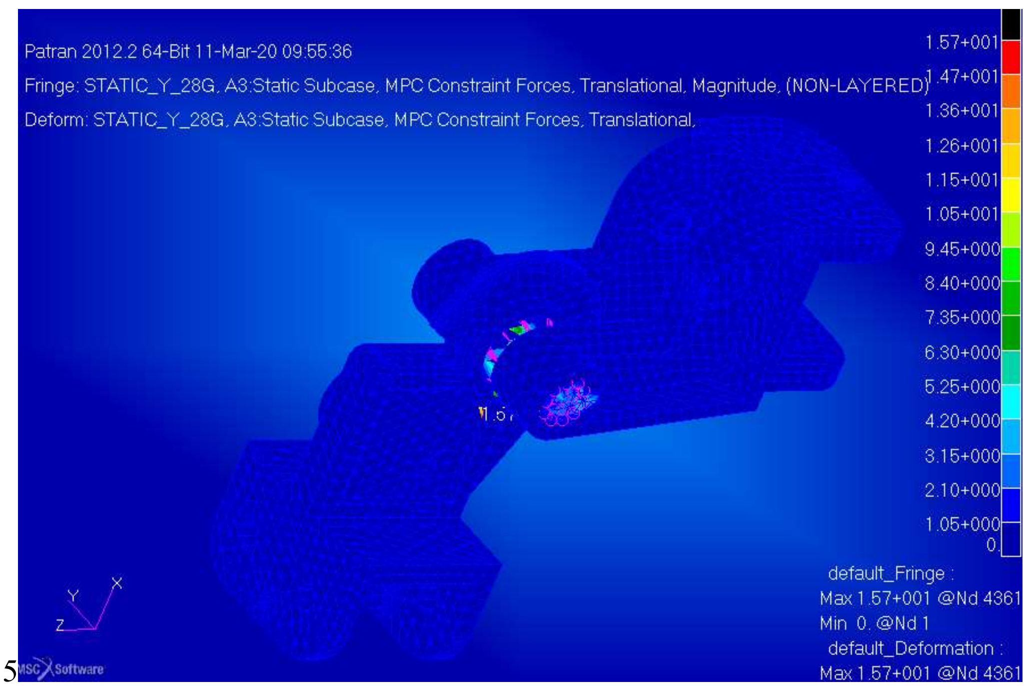


Figure 56: Deformation in Deployment Hinge due to Static Load along Y-Direction.



5

Figure 57: MPC Constraint Forces along Y-Direction.

4.6.3 Quasi-Static Analysis in the Z-Direction

Figure 58toFigure 60 show the Von-Mises stresses, displacement and MPC constraint forces in the deployment hinge in the Z-Direction respectively.

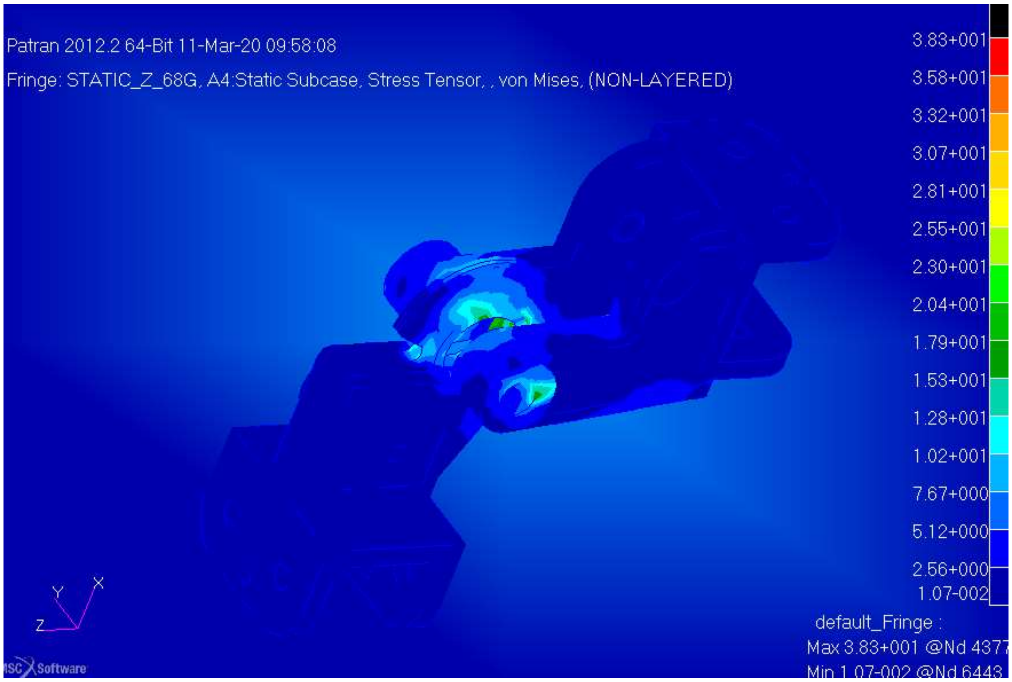


Figure 58: Stress in Deployment Hinge due to Static Load along Z-Direction.

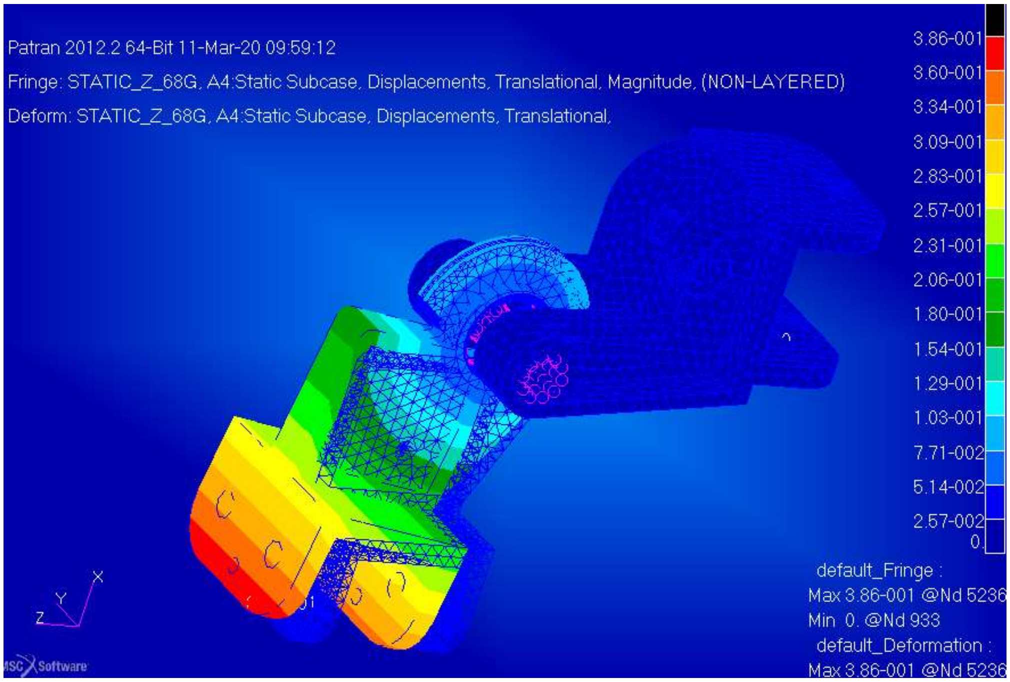


Figure 59: Deformation in Deployment Hinge due to Static Load along Z-Direction.

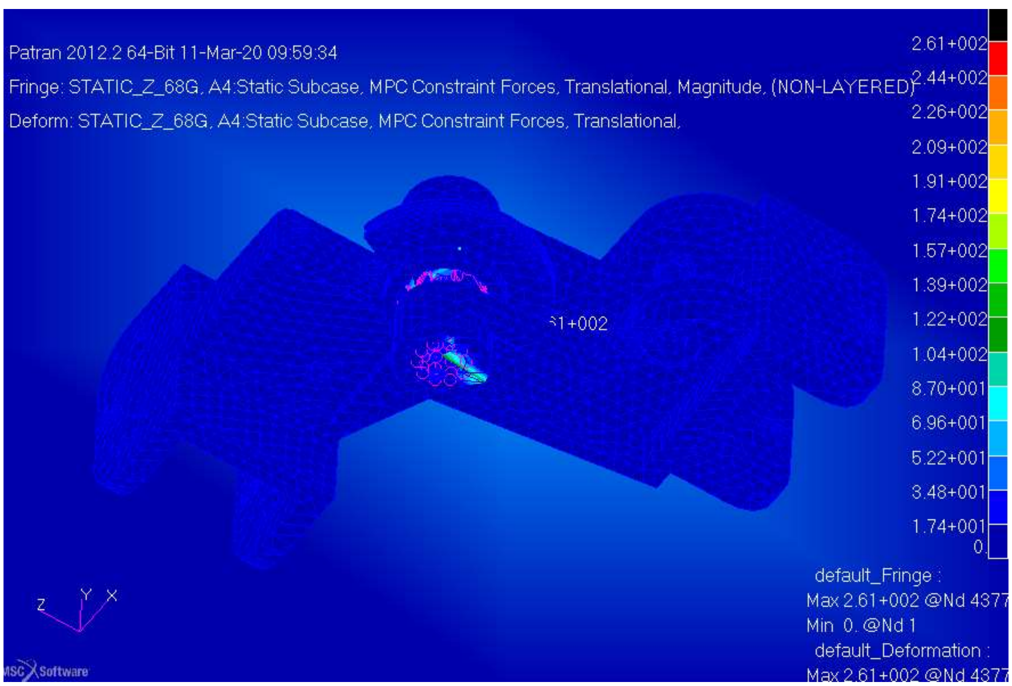


Figure 60: MPC Constraint Forces along Z-Direction.

4.7 Random Analysis

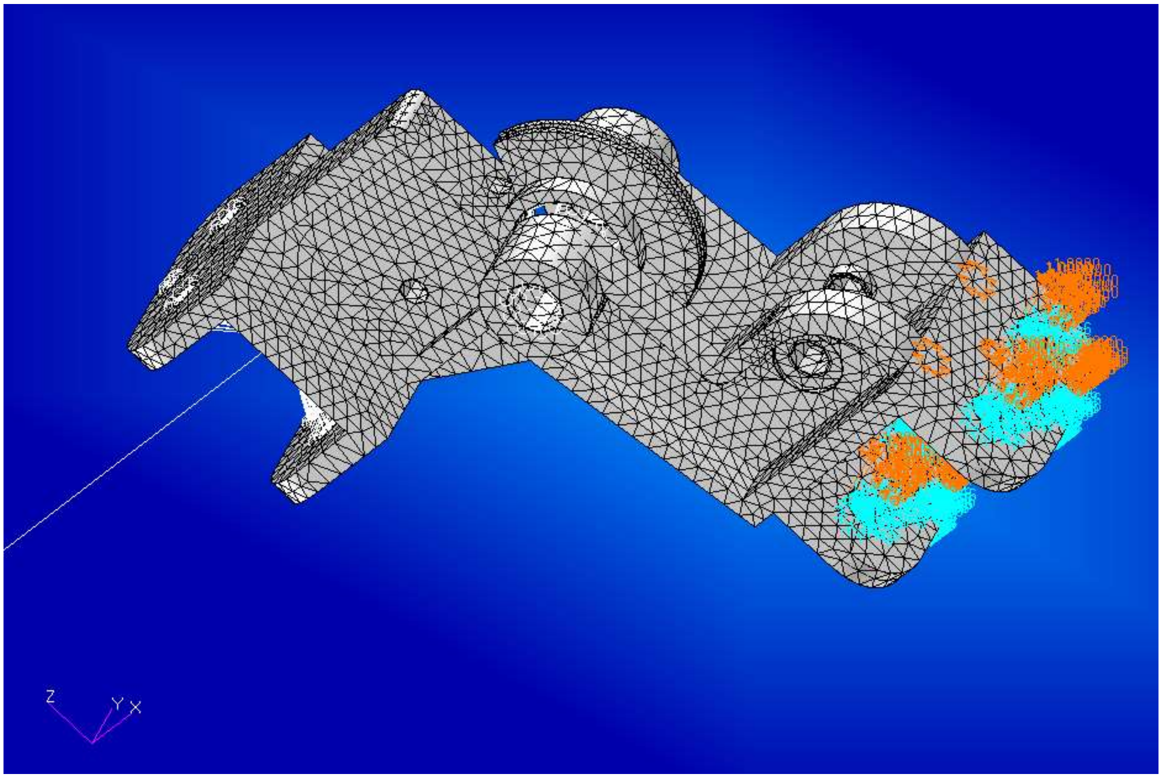


Figure 61: Hinge FEM Model for Random Analysis.

4.7.1 Random Analysis X-Direction Results

Figure 62 to Figure 64 show the random input load, output response and input/output comparison.

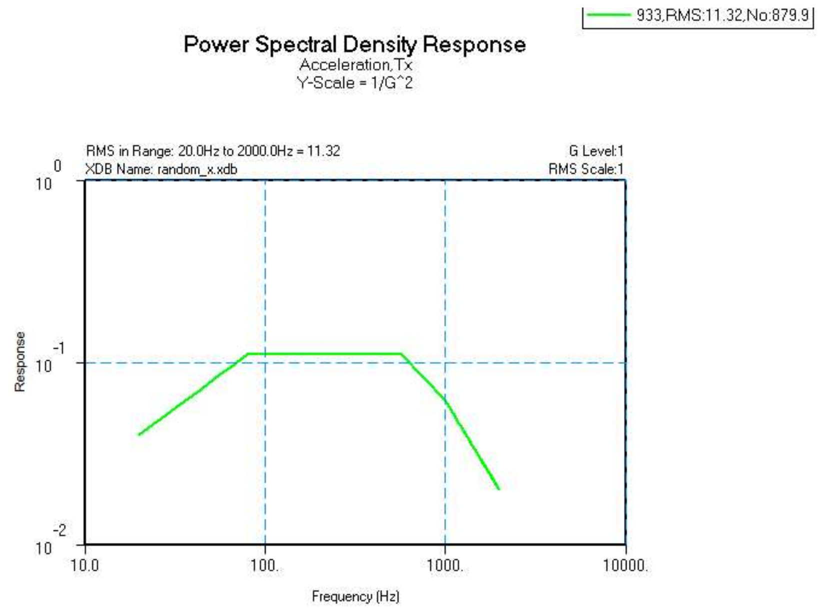


Figure 62: Input Random Load X-Direction (Lateral Direction).

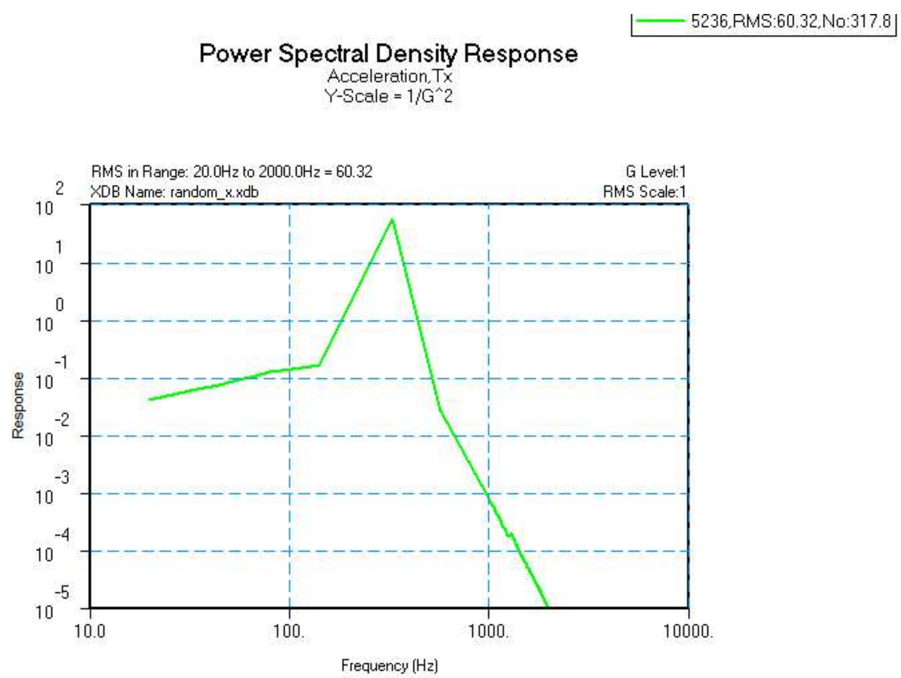


Figure 63: Output Random Response GRMS X-Direction (60.32 G).

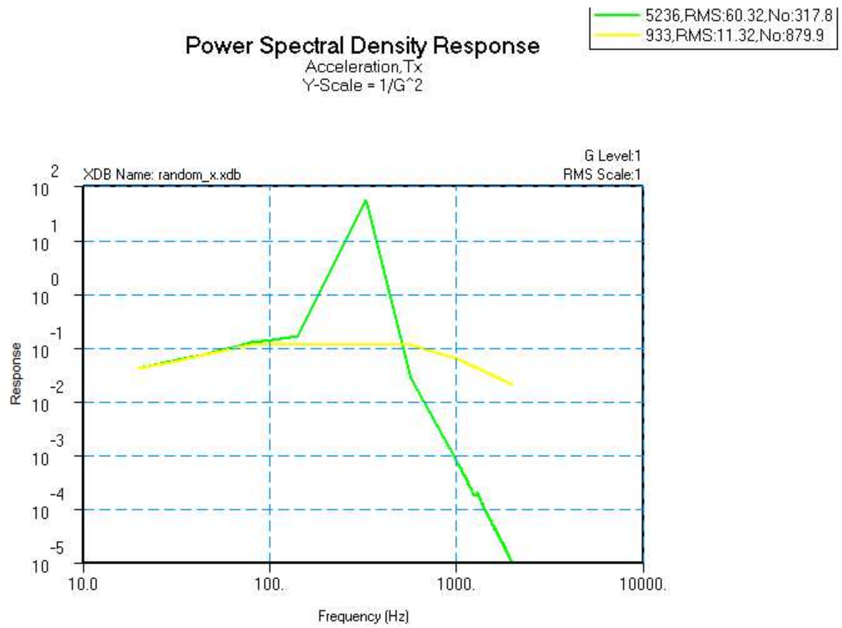


Figure 64: Input and Output Comparison X-Direction.

4.7.2 Random Analysis Y-Direction Results

Figure 65 to Figure 67 show the random input load, output response and input/output comparison.

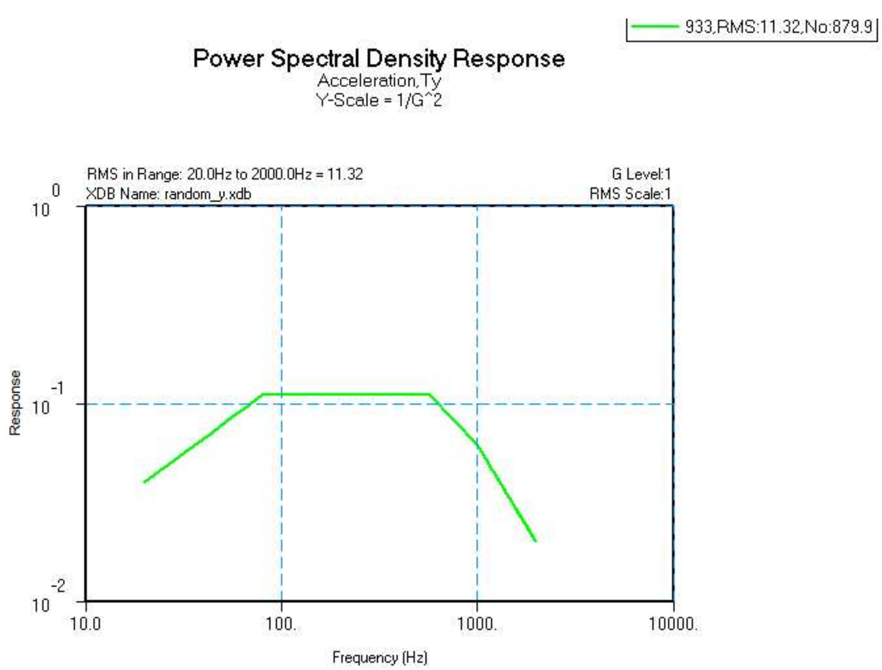


Figure 65: Input Random Load Y-Direction (Lateral Direction).

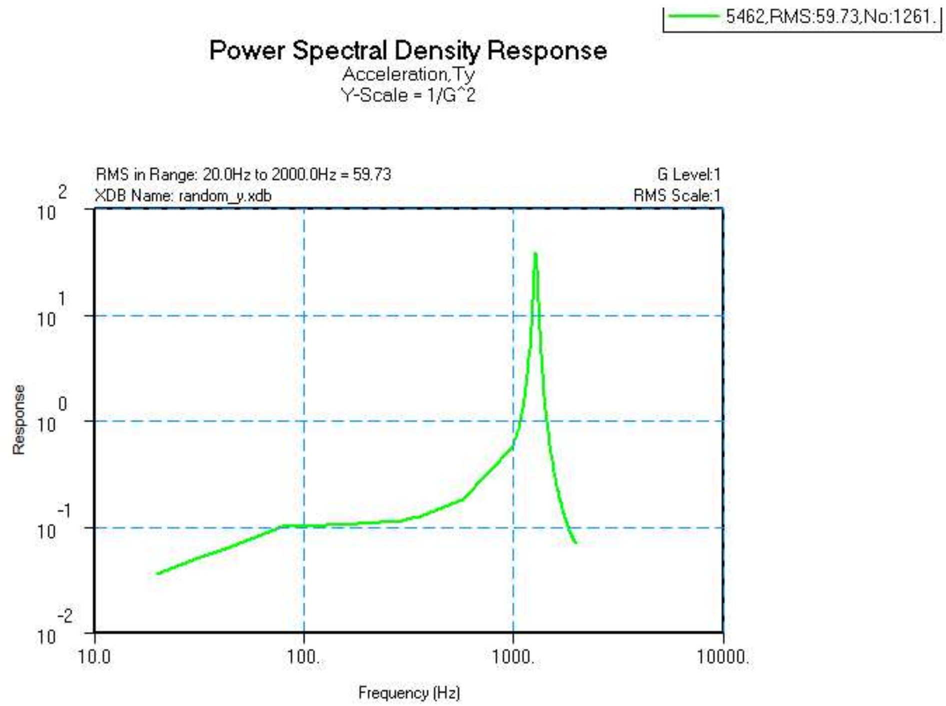


Figure 66: Output Random Response GRMS Y-Direction (59.73 G).

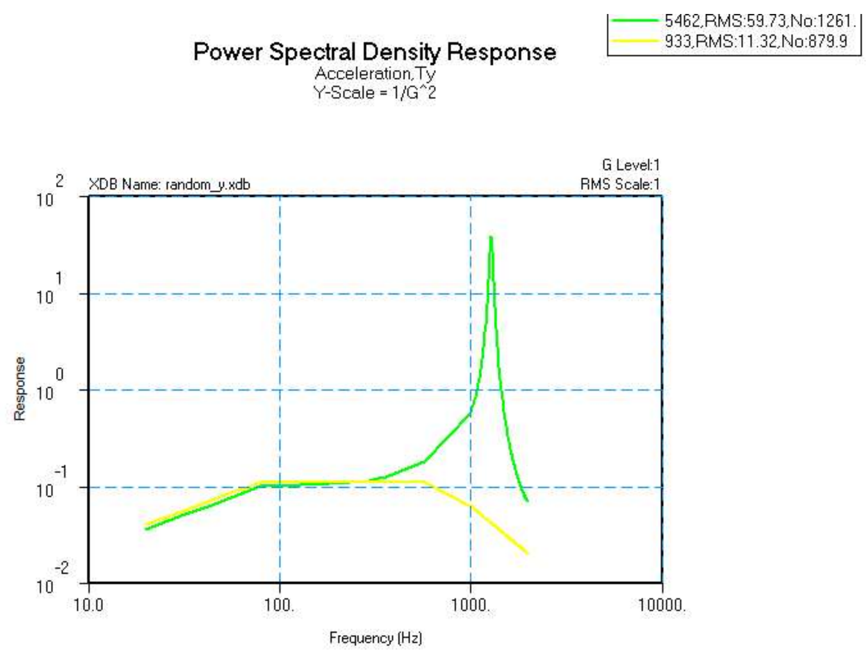


Figure 67: Input and Output Comparison Y-Direction.

4.7.3 Random Analysis Z-Direction Results

Figure 68 to Figure 70 show the random input load, output response and input/output comparison.

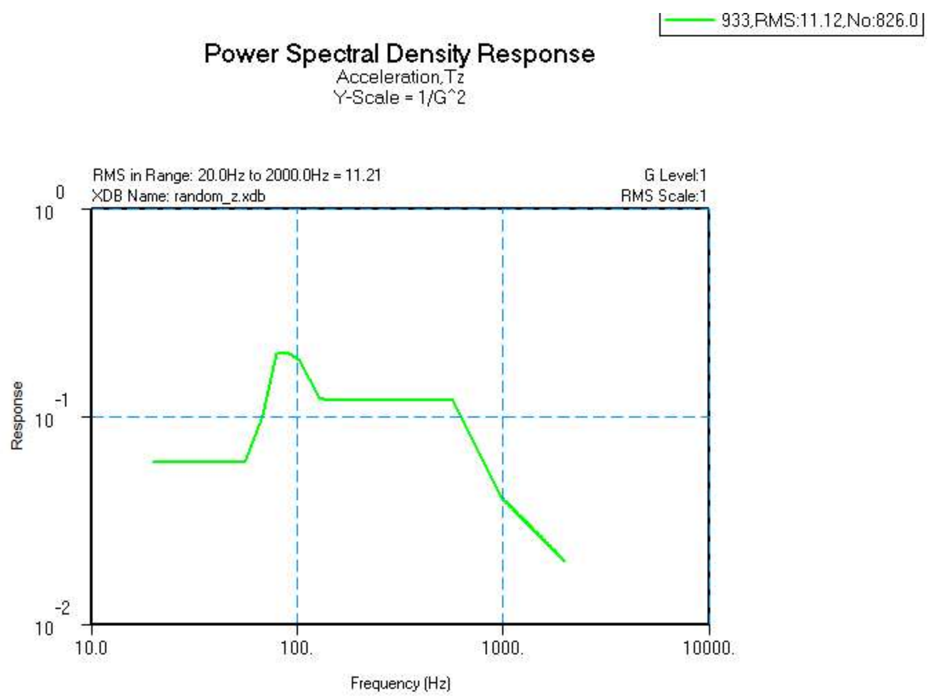


Figure 68: Input Random Load Z-Direction (Longitudinal Direction).

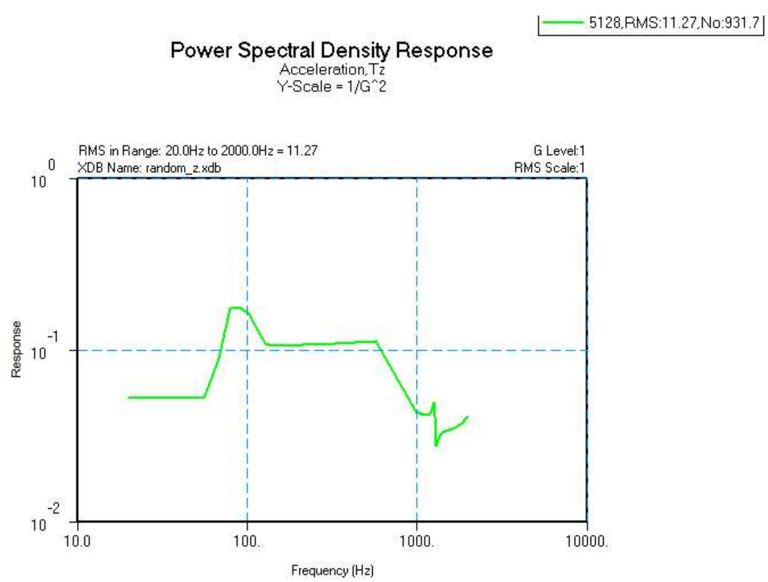


Figure 69: Output Random Response GRMS Z-Direction (11.27 G).

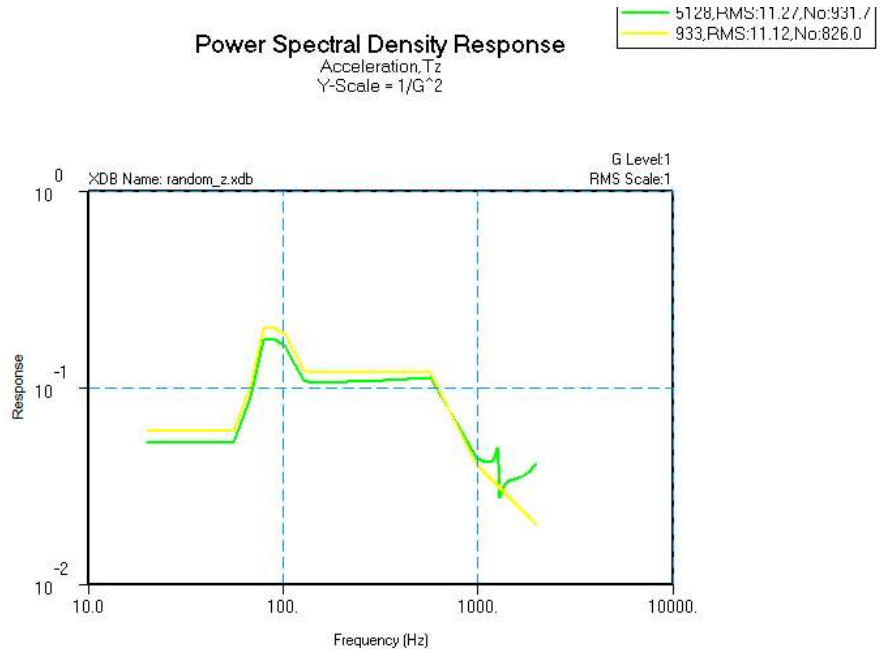


Figure 70: Input and Output Comparison Z-Direction.

Table 17: Random Results in X, Y, Z directions.

X Direction-GRMs (G)	Y Direction-GRMs (G)	Z Direction-GRMs (G)
60.32	59.73	11.27

4.8 Random Static Analysis Loads

GRMs values calculated from the input load field and output from FEM obtained from random analysis are shown in Table 18.

Table 18: Random 3δ GRMS Static Loads.

X direction		Y Direction		Z Direction	
Input 3GRMs (g)	FEM GRMs (g)	Input 3GRMs (g)	FEM GRMs (g)	Input 3GRMs (g)	FEM GRMs (g)
34.29	60.32	34.29	59.73	33.84	11.27

The larger of the values of static loads in all three directions will be used for Random Static Analyses and subsequent stress calculation.

Stresses obtained from Random-Static analyses are tabulated in Table 19.

Table 19: Results of the Static Analysis

Load Level (g)	The max. stress and deformation of the Deployment Hinge		Max. Constraint force (N)
	The max. stress (MPa)	The max. deformation (mm)	
$a_x = 60.32g$	12.1	0.132	82.1
$a_y = 59.73g$	7.53	0.052	33.6
$a_z = 33.84g$	19.1	0.192	130

4.8.1 Random Static Analysis X-Direction Results

Figure 71to Figure 73 show the Von-Mises stresses, displacement and MPC constraint forces in the deployment hinge in the X-Direction respectively.

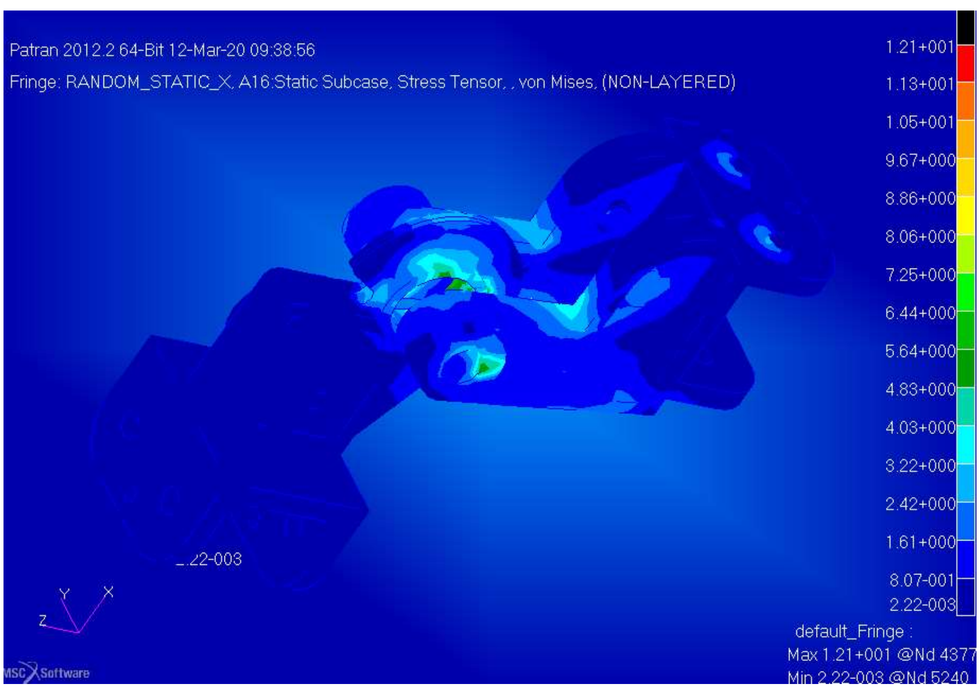


Figure 71: Stress in Deployment Hinge due to Static Load along X-Direction.

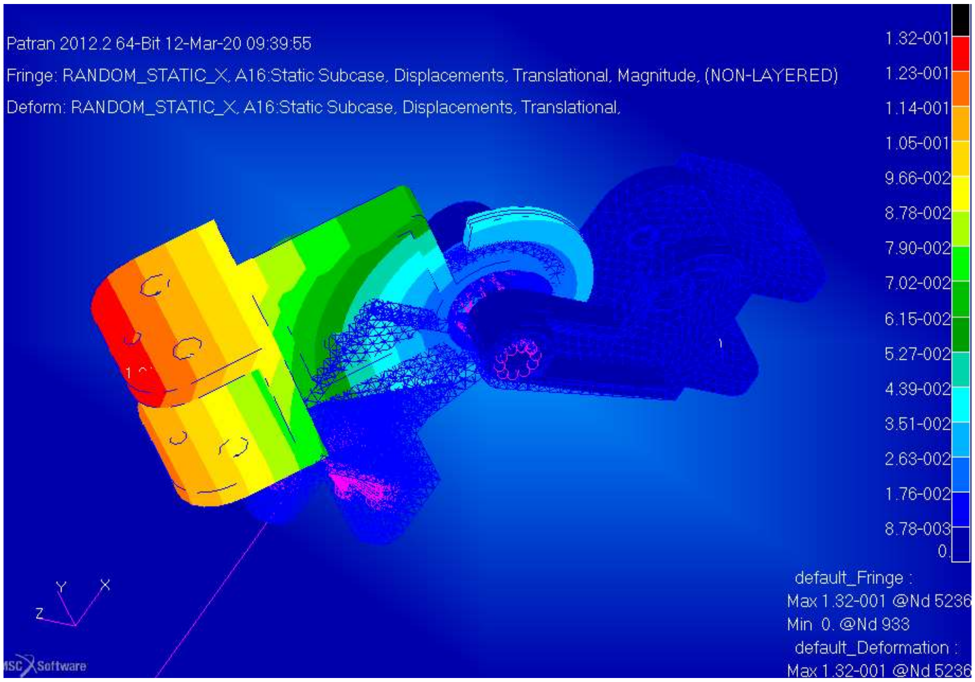


Figure 72: Deformation in Deployment Hinge due to Static Load along X-Direction.

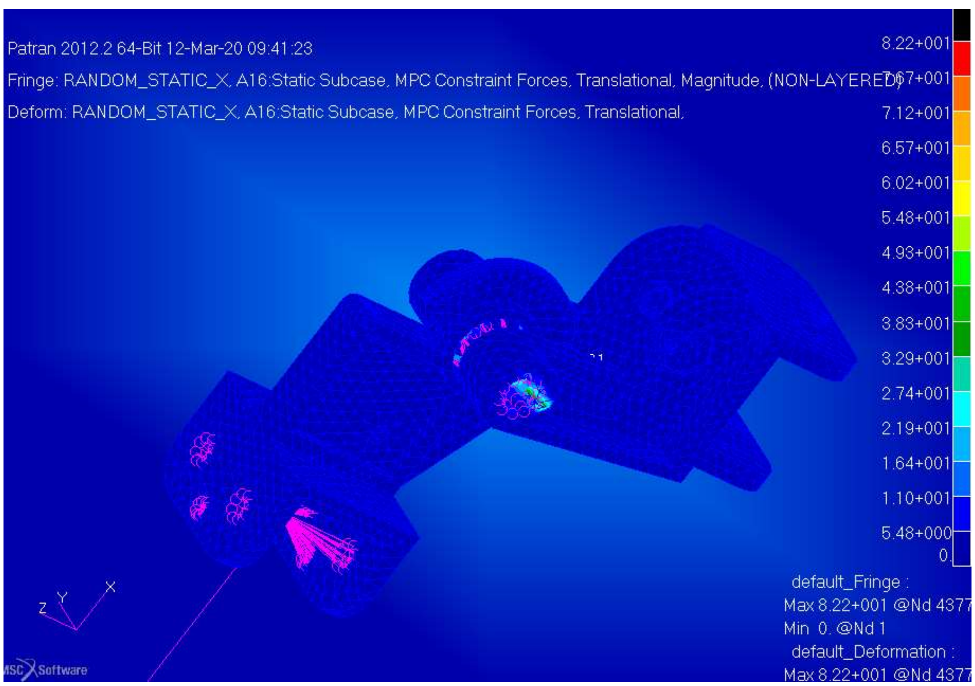


Figure 73: MPC Constraint Forces along X-Direction.

4.8.2 Random Static Analysis Y-Direction Results

Figure 74 to Figure 76 show the Von-Mises stresses, displacement and MPC constraint forces in the deployment hinge in Y-Direction respectively.

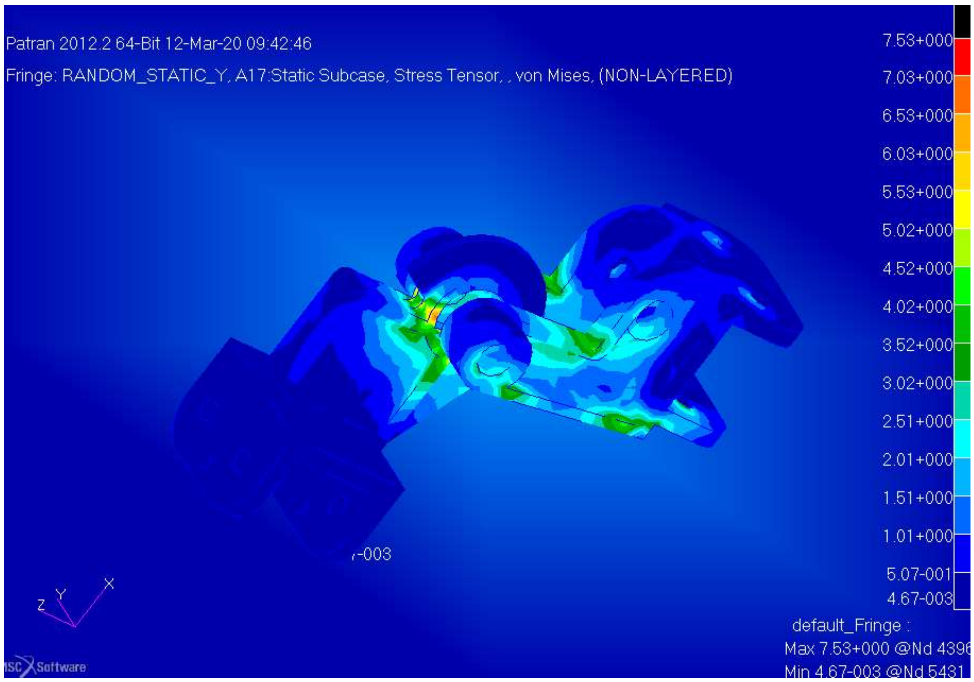


Figure 74: Stress in Deployment Hinge due to Static Load along Y-Direction.

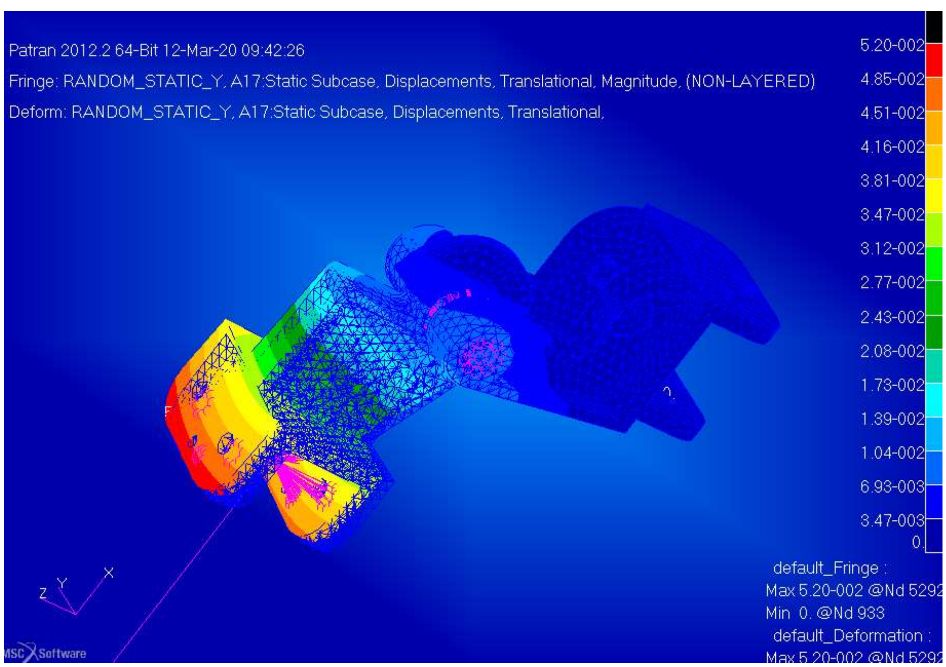


Figure 75: Deformation in Deployment Hinge due to Static Load along Y-Direction.

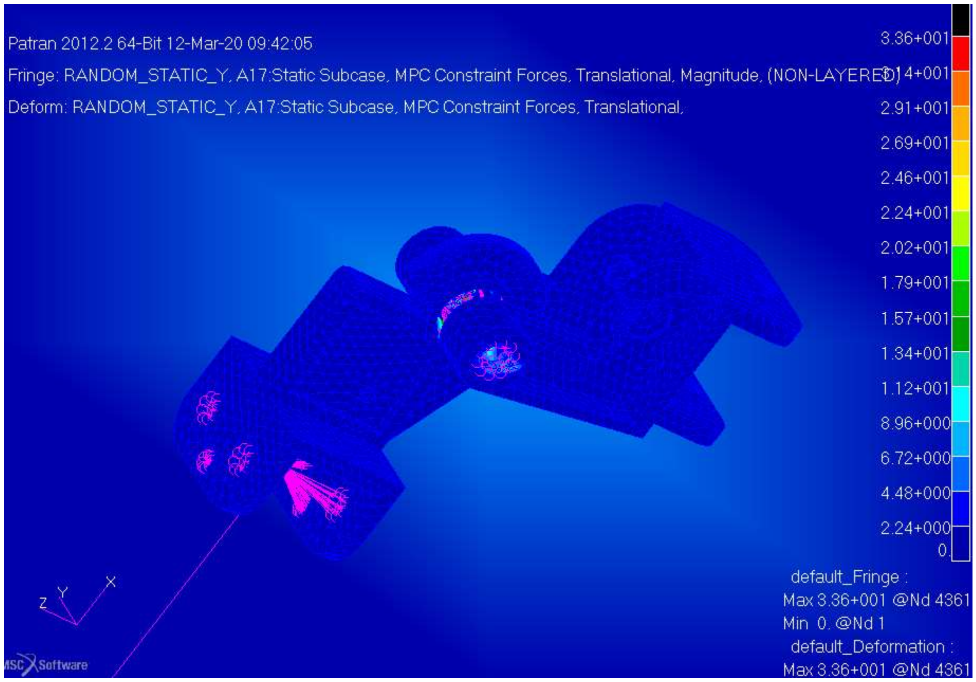


Figure 76: MPC Constraint Forces along Y-Direction.

4.8.3 Random Static Analysis Z-Direction Results

Figure 77 to Figure 79 show the Von-Mises stresses, displacement and MPC constraint forces in the deployment hinge in Z-Direction respectively.

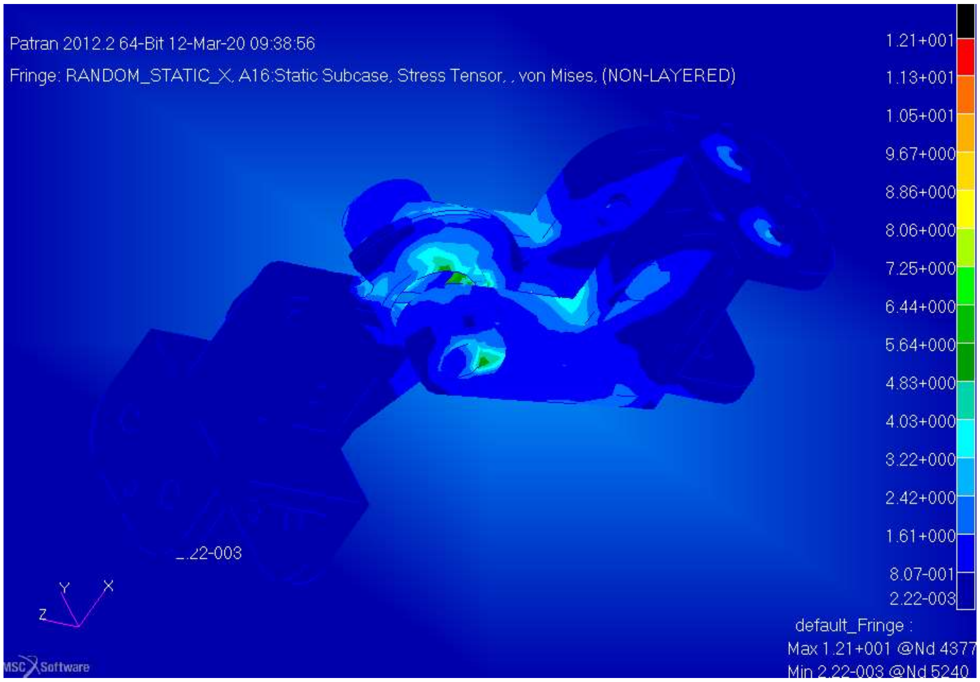


Figure 77: Stress in Deployment Hinge due to Static Load along Z-Direction.

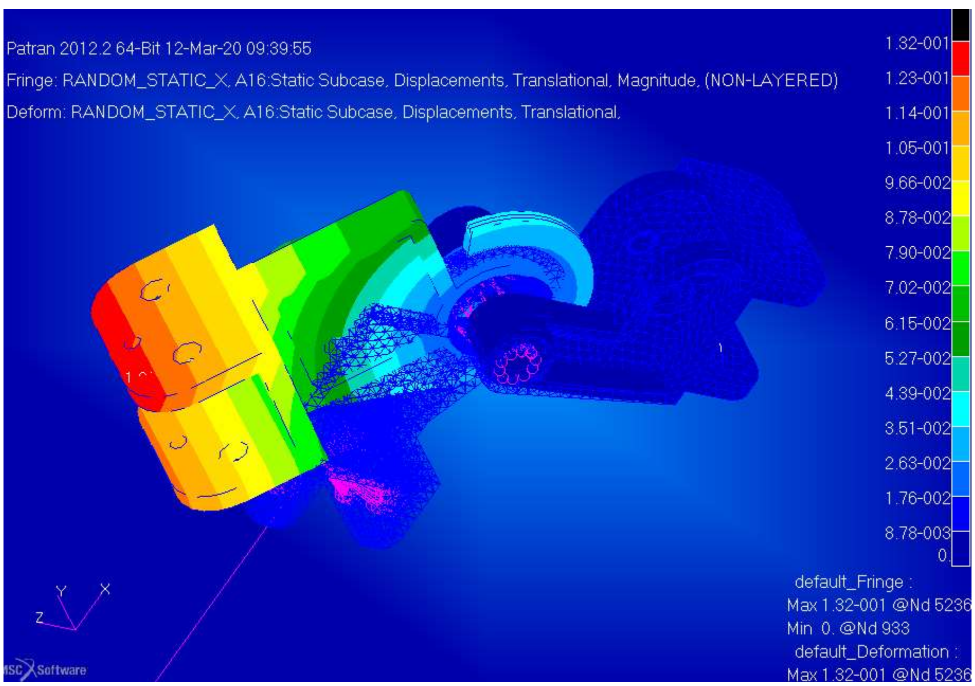


Figure 78: Deformation in Deployment Hinge due to Static Load along Z-Direction.

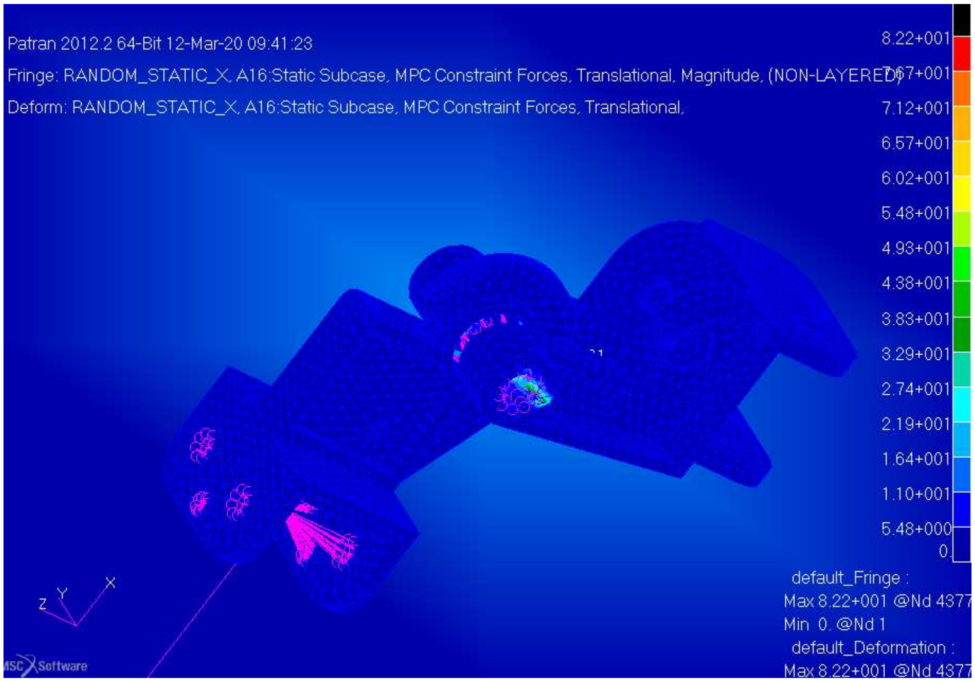


Figure 79: MPC Constraint Forces along Z-Direction.

4.9 Margin of Safety

The margin of safety can be defined as follows:

MS = Sa/Se – 1

Where,

Sa: Material Allowable Stress,

Se: Stress results from analysis.

After performing the structural analyses, the maximum value of Von-Mises stresses generated in the deployment hinge in all three directions is obtained to be 38.3 MPa.

The calculation is shown in Table 20.

Table 20: Margins of Safety.

Subassembly	Max. Stress (Se) MPa	Max. Allowable Stress (Sa), MPa	Margin of Safety (MoS)-
Deployment Hinge	38.3	275	6.18

4.10 Conclusion

The structure design of the Deployment Hinge is satisfying the minimum passing criteria after performing Natural Frequency Analysis, Quasi-Static, Sinusoidal Analysis, Random Analysis and Random Static Analysis.

The margin of safety for all components is greater than zero.

5 TESTING

This chapter describes the activities to be performed to generate a zero ‘g’ environment or gravity suspension system as on the ground. This process of deploying solar arrays using a deployment mechanism is carried out after completing the mechanical environmental testing (Vibration, Thermal Vacuum, Shock, etc) of the satellite.

These activities mainly include solar array deployment and illumination testing, followed by their removal from the satellite and installation on the simulating walls. Detailed procedures along with the relevant drawings, layouts or sketches (wherever possible) are explained in subsequent sections.

The approach of adopting the available system for the deployment testing rather designing a new one will be evaluated for this research work. There are four types of gravity compensation methods use for the space industry:

- Free Fall Methods
- Buoyancy Methods
- Pneumatic Methods
- Mechanical Methods

Out of these compensation systems, the mechanical methods are the most simple and cost effective for deploying small size solar arrays.

The testing of the deployment mechanism involves the following steps:

- Testing of spring’s stiffness
- Testing of Hinge’s torque
- Deployment test of solar wing on ground

The testing of spring stiffness and torque is performed by using laboratory equipment; however the complete deployment test is performed by mechanical methods.

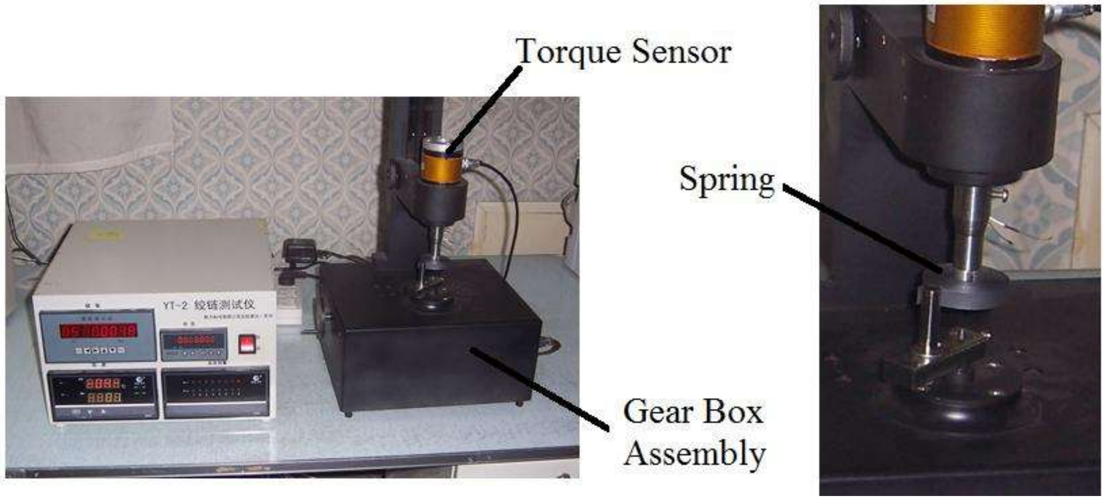


Figure 80: Stiffness measurement equipment [37].

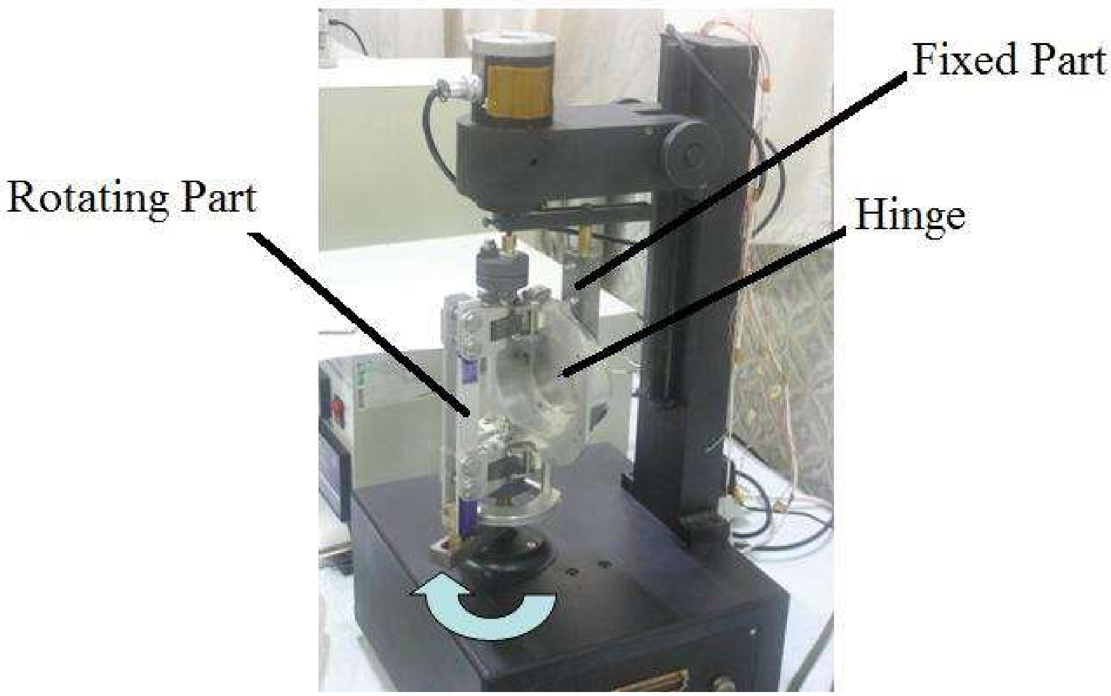


Figure 81: Torque measurement equipment [37].

5.1 Mechanical Suspension Method

Due to the microgravity conditions in space the deployment of the array in space has two degrees of freedom, therefore the mechanical suspension system provides the most suitable technique for on-ground simulation and it is simple, robust, user friendly and cost effective. In this system of suspension, wires, cords, pulleys, trolleys, rails or springs are used to create a '0' g environment. Gravity is compensated in most cases by more or less discrete forces which occur due to discrete suspension points. A schematic view for the system is shown in Figure 82 [38].

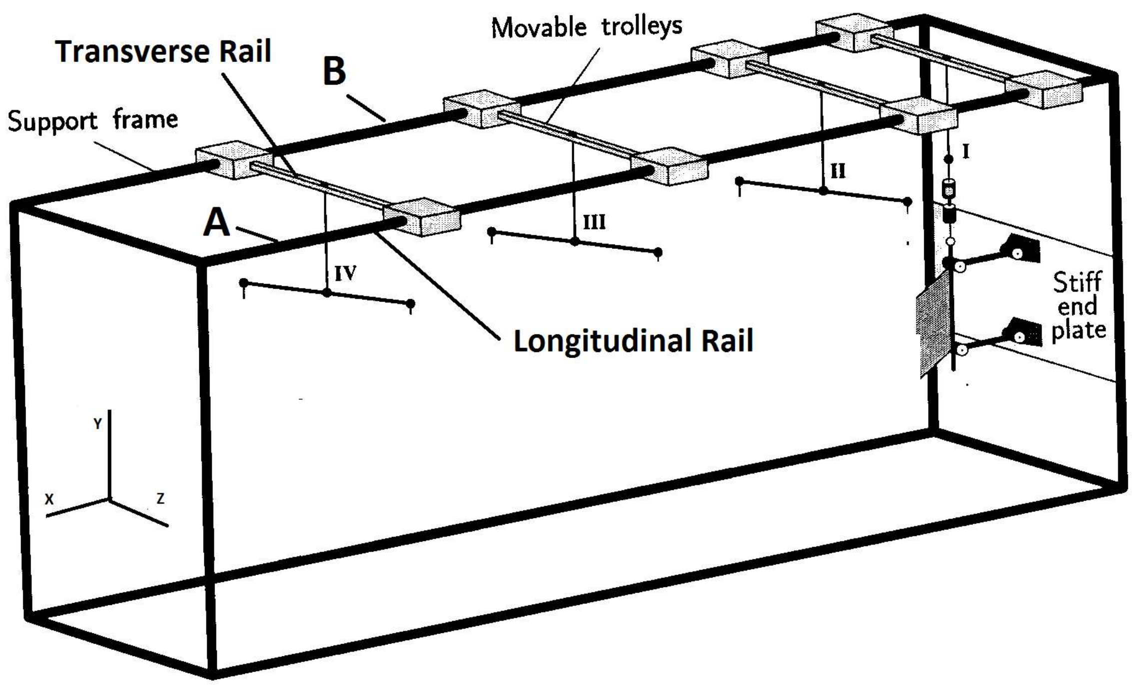


Figure 82: Gravity suspension system for 'Zero g' simulation.

The process of testing the deployment mechanism has shown in Figure 83.

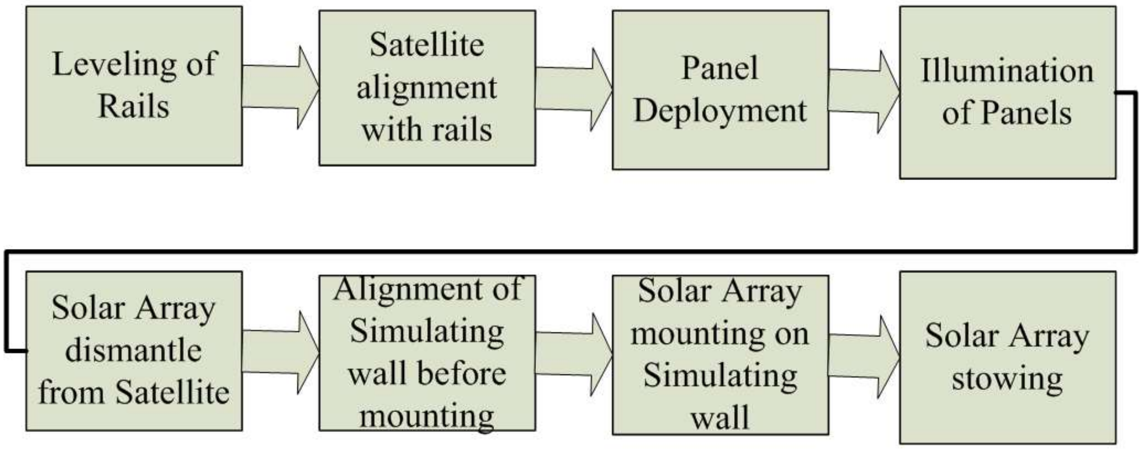


Figure 83: Process of deployment testing of solar arrays.

The generation of friction force between the supporting frame (rail) and moveable trolley, when the panel is pushed outward by the spring, is the main disadvantage of this system. It can be reduced by as much as possible by using bearings. The magnitude of the friction force depends on the weight of moveable trolley and the acceleration force of the panel. The actual value of the both dynamic and static friction is determined for simulation. The procedure involves rolling of a known weight along the rail at various speeds and calculating its pulling force; this pulling force will be equal to the friction force.

5.2 Adjustment Process

The process requires the following tooling:

- Lifter
- Theodolites
- Hardware Tool box (Spanner, Screw drive etc)
- Transverse Rail
- Hanging Steel Rod
- Target Mirror (pasted on the transverse rails and steel rod

The level adjustment of transverse rails during deployment is very important to achieving zero-gravity conditions for the solar array. This level adjustment is achieved before installation of the solar array. The methodology involves first aligning the longitudinal rail exactly horizontal (X Axis) using a theodolite. As the height of the frame structure is very large in comparison with the theodolite, a bar is hanged at the longitudinal rail with a target at its end. This target is than viewed through the eyepiece of theodolite to note its position. This is move along the longitudinal direction to get second reading, the difference between two consecutive reading should not be more than 0.2mm.

If we move the suspended bar along the X-axis, then the value measured along the Y-axis needs to be same at every point measured along the X-axis. Once the rail A has achieved the required straightness, the rail B can be made parallel to it.

5.3 Failure Mode Effective Analysis

The classification of failures for the mechanism has been tabulated below [23].

Table 21: Failure Classification.

Level	Final Effects	Explanation
I	Catastrophic	Cause the loss of satellites
II	Critical	Cause the satellite system damaged, or cause the missions failed
III	Marginal	Cause the satellite missions degraded or delayed
IV	Negligible	Not pertain to any category above

Table 22: Failure Mode Effective Analysis (FMEA).

S No.	Description	Failure Mode	Possible Causes	Failure Effect	SPF	Criticality Level	Observable symptoms	Preventive Compensatory provisions	Remarks / Recommendations
1.	Hinge drive spring	Spring breaks	Inadequate material and manufacturing process	Failure of solar panel deployment	Y	I	Indication signal of deployment	1. Mounting two hinges at each hinge axis 2.Controlling material	For deployment of solar panels
2.	Hinge bearing	Rotation of bearing is not agile	Contamination	Increase in deployment duration	N	III	Indication signal of deployment	Controlling material and manufacturing process	For deployment of solar panels

3.	Latching Device	Hinge is not latched	Failure of latching unit	Failure of the latching device can cause the detainment of solar	N	II	Indication signal of deployment	Controlling material and manufacturing process	Latching of solar panels after deployment
4.	Hinge Cold Welding	Locking Mechanism Failure	Contact of metals for longer duration at very low	Failure of solar panel deployment	Y	I	Indication signal of deployment	layer of molybdenum disulfide grease should be used to	For deployment of solar panels

6 CONCLUSIONS AND FUTURE WORK

The objective of this thesis was to propose a solar array deployment mechanism for a small satellite. In this thesis a comprehensive study regarding need for small satellites in future has been presented in Chapter 1. There is a great potential for small satellites in future. The space industry is focusing on "Small and Cheap" mission concepts. So the importance of deployable solar arrays will increase in near future, as the power requirements for small satellites are increasing, which can only be fulfilled by deployable arrays.

In this work I represent the functional and design requirements for a solar array deployment mechanism, its causes of failures with mitigation technique. The detailed design of latching, holding and release mechanisms for solar arrays is not represented in this thesis as the designing requires more details regarding interface of each subsystem of satellite. Together with this the thermal design and analysis of the deployment mechanism is not performed. I recommend the following suggestions for future work.

6.1 Thermal Design and Analysis

The operating temperature range for a LEO orbit is -100ºC to 100ºC. For thermal design and analysis of the deployment mechanism, the following guidelines should be followed:

- Use a simplified CAD model by removing minor details like threads and holes without compromising on the thermal path.
- Bolts, washers, nuts, external connectors, cut-outs and rounds can be removed as they have no influence on thermal analysis.
- Radiation heat transfer shall be considered from all surfaces of the equipment and avoid any conductive path from mechanism to satellite body and during sun pointing there is possibility of heat transfer from mechanism structure into satellite, which may increase the temperature of units inside the satellite.
- Analysis to be performed for Hot and Cold case. Use following values of flux for performing analysis.

Case	Hot case	Cold case
Solar Flux	1322 w/m^2	1414 w/m^2
IR	218 w/m^2	244 w/m^2

6.2 Holding and Release Mechanism

The solar panels are placed in the stowed position in launcher as stated in Chapter 1. To hold the panels with satellite body a holding and releasing mechanism is required. I proposed a holding mechanism using Nichrome wire with following design suggestions:

- The materials used must all be space proven and in common use. Also the materials must be compatible with each other.
- The temperature of the heating Nichrome wire must be above the degradation temperature of nylon, which is 250 degrees centigrade.
- The springs which will be used for the spring pins to move should have enough stiffness to penetrate through the nylon rope easily.

6.3 Locking Mechanism

There is a requirement for the mechanism to lock after achieving the desired orientation, in my case 90^o to satellite body. I propose a compression spring mechanism with a guided pin which can lock the panels after achieving the desired locking point. The above suggestion should all be incorporated in the proposed design.

APPENDIX A: MEAN EFFECTIVE MASS FRICTION

S. No.	Frequency	T1	T2	T3
		Fraction		
1	247.41	6.55E-04	5.27E-04	9.57E-01
2	673.53	4.05E-08	9.57E-01	5.28E-04
3	2123.5	9.84E-01	2.67E-08	4.49E-04
4	2248.2	1.44E-05	2.38E-02	1.55E-05
5	2711.8	5.65E-06	2.37E-03	4.08E-04
6	5803.5	6.79E-04	1.17E-04	1.83E-02
7	7272.5	1.53E-05	2.09E-03	5.24E-04
8	7911.7	4.70E-06	3.88E-05	2.92E-05
9	8525.0	1.99E-03	1.17E-05	4.35E-03
10	10487.0	2.44E-05	1.20E-03	4.29E-05

Material Properties

Material	Modulus of Elasticity (GPa]	Poisson Ratio	Density [kg/m^3]	CTE[10^{-6} m/m.K]	Yield Point (MPa)
Al-6061-T6	68.9	0.33	2700	23.6	276

7 REFERENCES

[1] D. M. Allen, "A survey of next generation solar arrays (for spacecraft electric power)," *AIAA, Aerospace Sciences Meeting & Exhibit, 35th, Reno, NV,,* pp. 6-9, 1997.

[2] J. P. Laboratory, Solar Array Design Handbock, California: California Institute of Technology, 1976.

[3] P. A. Joness and B. R. Spence, "Spacecraft Solar Array Technology Trends," in *IEEE Aero Space Conference*, USA, 1998.

[4] Europeon Space Agency, [Online]. Available: http://esapub.esrin.esa.it/bulletin/bullet82/mes82.htm.. [Accessed June 2019].

[5] H. R. Ali-Akbari, "Design of a Satellite Solar Panel Deployment Mechanism Using the Brushed DC Motor as Rotational Speed Damper," *International Journal of Aerospace and Mechanical Engineering,* vol. 12, no. 2, pp. 345-350, 2018.

[6] A. W. Berqquist, J. M. Jensen, B. J. Santies and A. T. Vaughan, "Structures, mechanisms, launch vehicle," Aerospace and Ocean Engineering Department, Virginia Tech, Blacksburg, VA, 2001.

[7] 11 July 2019. [Online]. Available: https://www.machinedesign.com/springs/what-are-differences-between-springs.

[8] P.S.Valsange, "Design Of Helical Coil Compression Spring" A Review," *International Journal of Engineering Research and Applications,* vol. 12, no. 6, pp. 513-522, 2012.

[9] Pacific Scientific Material Energy Company, [Online]. Available: https://psemc.com/products/pyrotechnic-cutter/. [Accessed 15 June 2019].

[10] "ECSS-E-30 Part 3A: Mechanisms"; ESA".

[11 S.Kabeer, "Space Application," University of Cape Town, Cape Town, 2018.

]

[12] G. Richardson, "Small Sayellite Trends 2009-2013," in *29th Annual AIAA/USU Conference on Small Satellites*, California, 2015.

[13] "M Design," HaWK Solar Array, [Online]. Available: https://mmadesignllc.com/products/solar-arrays/. [Accessed 30 June 2019].

[14] B. Lal, "Trends in Small Satellite Technology and the Role of the NASA Small Spacecraft Technology Program," Science and Technology Policy Institute (IDA), 28 March 2017. [Online]. Available: https://www.nasa.gov/sites/default/files/atoms/files/nac_march2017_blal_ida_sstp_tagged.pdf. [Accessed 22 June 2019].

[15] C. O. Quinn, "Setting the standard: Launch units for the SmallSat Era," Aerospace Corporation, May 2018. [Online]. Available: https://aerospace.org/paper/setting-standard-launch-units-smallsat-era. [Accessed 22 June 2019].

[16] N. United, "United Nations Treaties and Principles on Outer Space, UNITED NATIONS PUBLICATION,ISBN 92-1-100900-," 1975.

[17] T. Wekerle, J. B. Pessoa, L. E. Verqueiro and L. d. Costaro, "Status and Trends of Smallsats and their Launch Vehicle, An updated review," *J. Aerosp. Technol. Manag., São José dos Campos,* vol. 7, no. 3, pp. 269-286, 2017.

[18] "Small Satellite Market 5th Edition," Northern Sky Research, Dec 2018. [Online]. Available: https://www.nsr.com/research/small-satellite-markets-5th-edition/. [Accessed 23 June 2019].

[19] S. Works, "Nano-Microsatellite-Market-Forecast-9th-Edition-2019," Space Work Enterprises Inc (SEI), 2019. [Online]. Available: https://www.spaceworks.aero/wp-content/uploads/Nano-Microsatellite-Market-Forecast-9th-Edition-2019.pdf. [Accessed 23 June 2019].

[20] S.Kabeer, "Space Mission Design and Analysis," University Of Cape Town, Cape Town, 2018.

[21] K. Vipavetz, "Systems Engineering Lessons Learned from Solar Array Structures and Mechanisms Deployment," NASA Langley Research Centre, Virginia, 2013.

[22] J. H. S. Jean-Francois Castet, "Satellite and satellite subsystems reliability: Statistical data analysis and modelling," *Reliability Engineering and System Safety,* vol. 94, no. 11, pp. 1718-1728, 2009.

[23] Y. Yang, L. Xie, S. Yan and J. Wu, "Failure Analysis of Deployment Mechanism of a Satellite Solar Array," in *9th International Conference on Reliability, Maintainability and Safety*, Guiyang, 2011.

[24] J. C. Yang, K. K and d. Groh, "Materials Issues in Space Environmnet," pp. 12-20, Jan 2010.

[25] M. M. Finckenor, K. K and d. Groh, "Space Environmental Effects," NASA ISS Program Science Office, Texas.

[26] B. Bhushan, Modern Tribology Handbook, Washington DC: CRC Press, 2001.

[27] C. Lafleur, "Spacecraft Encyclopedia," 2004. [Online]. Available: http://claudelafleur.qc.ca/Scfam-failures.html. [Accessed August 2019].

[28] E. B. Ron van der Ven, "ARA MK4 Solar Array Development," in *EDP Sciences*, 2007.

[29] P. Horn, "Design Construction and Validation of articulated solar panel for cubesat," Lulea Univerity of Technology, 2010.

[30] H. Wu, C. Wang and P. Liu, "Reliability Analysis of Deployment Mechanism of Solar Arrays," *Applied Mechanics and Materials,* vol. 42, pp. 139-142, 2011.

[31] W. Ley, K. Wittmann and W. Hallman, Handbook of Space Technology, New York: John Wiley & Sons, Ltd, 2009.

[32] R. M. Fowler, L. L. Howell and S. P. Magleby, "Compliant space mechanisms: a new frontier for compliant mechanisms," *Mechanical Science,* vol. 2, no. 2, pp. 206-211, 2011.

[33] P. Hohn, "Design, Construction and Validation of an articulated Solar Panel for Cubesat," Lulea University of Technology , Kiruna, 2010.

[34] Ju Won Jeong at al, "A novel tape spring hinge mechanism for quasi-static deployment of a satellite deployable using shape memory alloy," *Review of Scientific Instruments,* vol. 2, no. 85, 2014.

[35] H. Carlson, Spring Designer's Handbook, New York: Marcel Deckee Incorporation, 1978.

[36] R.C.Hibbler, Mechanics of Material, New York: Prentice Hall, 2011.

[37] Y. Qiaolong, "Mechanisms of Solar Wings," Chinese Academy of Science and Technology , Beijing, 2010.

[38] D. Schultheiß, "Gravity Compensation of Deployable Solar Arrays for Small Satellite," Univerity of Cambridge, Cambridge, 2013.