Schneiter et al (1975) successfully designed, developed, manufactured and tested the primary structure of 720 kg Thor Delta communication satellite and launched it. The introduction of advanced materials and consequent design optimization leads to attain a target structural mass of 20 kg of the primary structure. Mass and C.G measurements, static load tests were conducted and correlated to analytical results. Thermal problems were also analyzed and verified by corresponding tests.

Villalaz et al (1977) made a detailed survey on the development, manufacturing and testing activities of satellite structure which led to the development of mass optimized satellite structure. A modal survey test and a vibration test on the satellite structure were carried out in order to investigate its dynamic performance characteristics and to validate the predicted dynamic behaviour with Eigen value and a frequency response analysis.

Patki (1977) briefly presented the design, analysis and testing methods of the Aryabhatta satellite structure. The general requirements and constraints which formed the basis of the design of the structure are explained. The theoretical and experimental studies conducted to evaluate the structural behaviour under various environmental loads expected during ground operations and during flight are also described. The salient features of the fabrication procedures and the details of physical parameters are
presented. The evaluation of the structure is made on the basis of the results of ground tests and the flight performance.

The advisory panel headed by Granger Morgan (1990) aimed at finding ways to reduce the cost of launching spacecraft. This Background Paper reviews four possible approaches to spacecraft design like designing payloads to fit launch vehicles leaving size and weight margins of about 15 percent, allowing payloads to be larger and heavier, allowing satellites to be simpler, and make them lighter and designing micro spacecraft to be launched like artillery shells to reduce the total spacecraft program costs.

Grooms et al (1992) addressed important issues that were closely relevant to spacecraft’s structural design such as ease and cost of manufacturing and inspection, material cost and availability, test and analysis consideration, expanded mission requirements and dynamic characteristics which were entirely different from most designer’s objective function or design criteria in the selection of the structural arrangement.

Srinivas Kodiyalam et al (1993) focused on application of rigorous optimization methods to the design of complete satellite structural subsystems. The paper discusses the development of a software system based on a combination of Finite Element Analysis using MSC NASTRAN, numerical optimization, approximation concepts, and object-oriented technology. The benefits of object-oriented design, in the context of satellite structural optimization, are identified. The application of the software to the actual design of two satellite structures is presented.

Cathleen Grastataro et al (1995) described the use of advanced composites in space application. The design of an all composite spacecraft structure for small satellites, engineering analysis and test results obtained from the development of the spacecraft engineering model are also presented.
Bonde et al (1995) have explained the concept of substructure effective mass for modal identification. They have stated that mode identification of a complex structure is an essential task towards better understanding of its dynamic behavior. Though graphical representation can give some idea of the behavior, it is difficult to comprehend quantitative measures from such mode shape plots as other existing methods of mode identification based on generalized mass principles give an indication only of the global modes. The examples illustrated in this paper show that the concept of substructure effective mass is a clear indication of all the modes—global or local, coupling of various substructures and relative motion of the substructures in any given mode.

Carl Wood et al (1996) detailed the overall design philosophy, manufacturing processes, and initial test results of CATSAT (Cooperative Astrophysical and Technology Satellite). The design utilizes the flat aluminum panels that are screwed and bonded together, resulting in a stiff structure that costs less and requires less than 40 hours to assemble. The horizontal shelves in the structure are supported by vertical panels to transmit forces throughout the structure. Finite element analysis of the structure was performed to verify the strength of the panels and ensured that the resonant frequencies of the structure were met as per specification.

Not performing a complete and comprehensive stress analysis on the spacecraft structural components may lead to an inadequate design with unsafe or inefficient load paths. Without proper stress analysis, the objectives of minimum weight and a balanced design will not be met. The general methodology of performing the stress analysis for structures used in space applications have been described in NASA Reliability Practices PD-AP-1318 (1996).
Probability of failure is increased in flight due to low frequency transient environment. Some workmanship defects in large structures and full-up systems may go undetected. NASA Reliability Practices PT-TE-1406 (1996) explained the assemblies and full-up flight system subjected to sweep sinusoidal vibration.

NASA Reliability Practices PT-TE-1413 (1996) explained the principal objective of performing random vibration testing. The random vibration test practice assists in identifying the existing and potential failures in flight hardware so that they can be rectified before the launch.

DeSouza et al (1996) carried out the structural analysis of the Brazilian Scientific Applications Satellite (SACI-1) using the MSC/NASTRAN finite element code for static, normal modes, frequency response, transient response and random response according to the loads given by the launching agency. The results were used to identify whether the structure has fulfilled all the requirements of the mission.

Joel Rademacher et al (1996) explained the challenge in developing a 4.5 kg satellite to perform meaningful science and to fit inside the 0.2m³ Pegasus avionics section to be launched as a piggyback payload on a Pegasus rocket. The cost, size and weight limits were the constraints considered during the design.

Timothy Thompson and Lucie Parietti (1996) of Los Alamos National Laboratory (LANL) in partnership with Composite Optics Incorporated (COI) have successfully designed and tested a composite light weight affordable spacecraft structure. The design parameters were so identified that they could maximize the ratio of payload over structural weight while keeping the manufacturing cost as low as possible.
Cyrus Jilla and Miller (1997) discussed the possible future directions of the satellite design, various design ideas by considering the historical trends of the satellite development.

Michelle Coleman et al (1997) explained the model verification strategy including the selection of modal survey test modes, correlation goals, and the model updating approach based on hardware knowledge, engineering judgment, and analytical methods to produce the final test correlated Finite Element Model. In order to provide the most accurate dynamic flight model for the final verification load cycle, the Cassini Developmental Test Model (DTM) was subjected to a static test. For modal testing, mass mockups were added to the DTM. The objective of the correlation activity was to demonstrate the agreement of the Cassini spacecraft test analytical model (TAM) modes to the test the measured modal survey modes. The resulting updated TAM was then used in the preparation of the flight dynamic model for the final verification of loads cycle.

Manfred Degener et al (1997) presented the modal survey and sine vibration tests including the respective mathematical modeling of PPF/ENVISAT-1, one of the largest European satellites with a launch mass exceeding 8000 kg, with a length of 10 meters. The size and complexity of the structure and very high modal density predicted by the finite element analysis required more than 500 measurement points for the Modal survey and Sine vibration tests.

During the preparation for launch and then during the flight, the spacecraft is exposed to a variety of mechanical, thermal, and electromagnetic environments. The PSLV (Polar Satellite Launch Vehicle) user’s document (1999) provides a description of the environment that the spacecraft is intended to withstand.
Creto Vidal et al (1999) explained the application of design sensitivity analysis for changing the satellite structural design in order to reduce the lowest longitudinal frequency while reducing the structural mass. Making changes to a complex satellite structure, based on a trial-and-error procedure is highly ineffective and time consuming. Fortunately, with the new design sensitivity analysis formulations, it is possible to approach those complex problems in a very systematic way.

Anthony Davenport et al (1999) described an integrated and efficient approach to random analysis using MSC/PATRAN and MSC/NASTRAN. New enhancements available in MSC/PATRAN provided the analyst with an interactive pre and post-processing environment for random analysis. The new features were discussed and the results were presented demonstrating how the new capability was used on two different industrial case studies.

Fernanda Ravetti et al (1999) showed the importance of structural optimization and design sensitivity analysis in the redesign cycles of spacecraft structures and presented all the steps taken and difficulties encountered as they tried to maximize the first natural frequency from the low value of 18.78 Hz obtained with the first trial design to 40 Hz in order to avoid coupling between the rocket excitation modes and the natural vibration modes of the satellite. The upper bound value of mass of the satellite is maintained as 10.5 kg. All the modal and sensitivity analysis as well as optimization steps were performed using MSC NASTRAN. The design variable for the structural optimization steps was composed of thicknesses of the face and core of the sandwich panels.
Fuente and San Millan (1999) explained a method of performing random analysis which is a direct application of a well known result of linear systems theory and that allows exact computation of RMS values of any number of structural responses and that can be post processed as if they were originated in a conventional static analysis.

ASAP 5 (Ariane Structure for Auxillary Payload) User’s Manual (2000) provides technical information to assist users in preparing for a mission on this heavy-lift launcher. The Ariane 5 is tailored to the increasingly diversified demand for service including heavier and larger satellites, a wider range of orbits and combined missions.

Sedighi and Mohammadi (2000) studied the static and dynamic structural behavior of a micro-satellite using Finite Element Method by modeling the satellite parametrically using the ANSYS code and based on the data provided by ARIANE-5 launcher. The criteria for safe vibration range are extracted. The required natural frequencies of the satellite have been derived by changing the most effective geometrical parameters such as tray thickness and stringer height. Finally the optimum shape of the satellite has been arrived using the results of the analysis.

Swen Ritzmann et al (2000) conducted the dynamic analysis of the flight configuration of satellite structure to satisfy the high demands which are required to the structure. The test requirements for the qualification were adapted from KOSMOS specifications first, than from PSLV specifications.

Samuel et al (2001) have presented the number of important aspects related to the development of a generic static test facility that have been developed at Indian Space Research Organization laboratory with illustration of tests performed on a typical spacecraft structure. The quasi-static loads experienced by large spacecraft structures during launch are
generally simulated by static loading for qualification purpose and often specific to a spacecraft.

Tom Van Langenhove (2001) illustrated the process of correlation and updating on the Olympus satellite. Instead of building another expensive and time consuming classic modal survey test set-up, the measured data from the low level, medium level and high level qualification tests were used, processed and fed into a conventional modal analysis package in order to estimate the natural frequencies, the damping ratios and the mode shapes. Using this data as reference, the initial Finite Element Model is correlated and fine-tuned towards the resonance frequencies and Modal Assurance Criteria values. The use of LMS/Gateways for the correlation analysis together with LMS/Optimus provides o means to update the FE model, using both the measured resonance frequencies and the MAC values of unity as a reference.

Kurng Chang (2001) had done the force-limited random vibration test along two axes and demonstrated the satisfaction of qualification requirement of design and verification of the integration of the assembled flight spacecraft for the transmitted vibration which are encountered during the launch.

Bernelli-Zazzera et al (2001) had done a preliminary satellite structural design considering the need to accommodate all the components based on the requirements of electronic subsystems as well as the inertia characteristics.

Angelos Tsinas and Christopher welch (2001) examined the current approach to design of volume production in satellites and the implementation of volume production method.
Frank Reganato (2002) demonstrated the structural/mechanical integrity of the assembly when subjected to the qualification PSD random vibration excitation.

ISO/CD 16454/ISO TC 20/SC 14/WG 1(2002) clearly described the need for structural stress analysis of satellite structure and the importance of anticipating all the possible failure modes and design the structure against all of them.

Gram et al (2002) explained the importance of optimal use of space in small satellites as there is a limited space for placing equipment.

The Finite Element Analysis for two different satellites were carried out by John Middendorf (2003) using PATRAN/NASTRAN programs to look at the peak vibration response, which made the design greatly simplified so as to have identical dimensions and mass.

Kari Marjoniemi et al (2004) analyzed a wide collection of existing spacecraft, identified the commonalities and the design rules were developed. Low cost solutions, short development cycles, structural concepts allowing maximum freedom in payload accommodation, and structural applicability for several launchers were the key principles that were followed based on this.

Joana Maria Pereira Morgado (2004) designed a structural configuration such as to allow the satellite as a whole to fulfill the mission objectives as well as to allow each subsystem to perform their individual function. The main parameters that were considered when defining the position of each subsystem were their dimensions, mass and electrical and magnetic interferences with other subsystems. The impact caused by the position of each subsystem to the harness was also considered.
Joel Quincieu (2004) designed USUsat 2 satellite structure specifically for a modular platform. To provide a design platform that will reduce the cost, shorten lead-time and shorten the design and manufacturing processes, the micro satellite bus design should be reusable and the design must be based on a flexible architecture or configuration that can integrate the bus and the payload and that can be reproduced with a minimum amount of modification (modularity) and cost.

Yosi Sadkin (2004) performed the Quasi-Static (QS) test of a satellite on a shaker by a controlled Sine sweep vibration because of its simplicity and cost and time savings consideration. It also enabled using the same specimen for all tests, QS as well as sine vibration, acoustic and shock tests, with only slight modifications.

Sairajan et al (2005) in their work used the mathematical optimization tool, Optistruct to arrive at the design of a composite base structure of a typical spacecraft. The work involved replacement of an existing metallic structure with a composite part with improved mass and other properties without affecting interface and overall dynamic behavior. The optimized design resulted in weight saving of approximately 15% of mass with respect to initial design without compromising on interface requirements and structural performances.

Antonios Vavouliotis et al (2005) in their work address the structural analysis of the ejection system of Young Engineers Satellite 2 (YES2) in order to gain the confidence to the resized mechanical design. The structural analysis procedure followed can be divided in two steps. Firstly the forces acting on the ejection mechanism due to the random vibration and transient loads during the launch phase were derived from a modal analysis of FE model of MASS & FOTINO satellite part. The second step covers the FE modeling of the complex ejection mechanism and then the calculation of the
developing stresses on the each components of the mechanism under the action of those forces that were calculated during the first step. Finally the safety margins were calculated.

MR SAT Tethered satellite (2005) developed in University of Missouri-Rolla discussed the essentiality of a structural design with sufficient capacity to carry all the necessary components for the success of a spacecraft’s mission and also to limit the mass and size of the spacecraft in order to lower the manufacturing costs for placing it in the orbit.

The thermal control of micro-satellite presents with unique challenges to the thermal engineer since the mass, power and volume available are all very limited. Regardless of these problems and the extreme environments and changing power conditions (internally and externally) of the satellites the temperature in the subsystems must still be maintained within the specified limits. HAMSAT is a micro-satellite for providing satellite-based Amateur Radio services to the national as well as the international community of Amateur Radio Operators (HAMs). Badari Narayana and Venkata Reddy (2005) presented and discussed the thermal design and on-orbit temperature characteristics of HAMSAT. A purely passive thermal control system without heaters is used to control the temperature of the spacecraft.

Dafu Xu et al (2006) explained the complexity of the design of micro-satellite structure and necessity of modeling and analysis of micro-satellite for structure design, which not only verify rationality of the structure design but also contribute to optimization design. Utilizing the FEM software MSC/Patran and MSC/Nastran, which is popularly used in international aerospace field, dynamics analysis was carried out for Sat X, a micro-satellite.
Xiaomin Zhang et al (2006) have presented about the development of the first micro-satellite in the DFH Satellite Company Ltd. The first micro-satellite as a basic type is named as HummerSat-1. HummerSat-1 is three-axis stabilized with orbit control capability. Information and power control are implemented through an on-board network, Ga-As solar cell and Li-ion battery are adopted to obtain and storage power, S-band TT&C and data transmission works are used. The payload of HummerSat-1 has a weight of 60 kg and power consumption of 200 W. DFH would be aiming to produce product spectrum of 80–200 kg for different missions, including earth remote sensing, communication, science exploration, etc.

Small satellites are formed by subsystems and each subsystem is designed by the experts on that technical area. Suat ONTAC (2006) considered satellite structure as one of the important subsystem which is exposed to different environmental conditions and hence different approaches have to be considered while designing, analyzing, testing and verifying the satellite structure. The analysis, verification and test stages of a small satellite in terms of mechanical design have been explained.

The size of the small satellites results in small surface areas which often translate into thermal and power constraints. A small satellite may not have enough surface area for radiators and/or solar panels. The radiators are used to release the internal heat during hot environments, and solar panels create necessary power for the heaters during cold environments. Because of the surface area and power limitations, a passive thermal design was then selected for the Formation Autonomy Spacecraft with Thrust, Relative Navigation, Attitude, and Crosslink Program (FASTRAC) twin satellites. Millan Diaz-Aguado et al (2006) performed the thermal cycling and thermal analysis of FASTRAC. The thermal cycling was done in Chamber-N at Johnson Space Center, Texas, considering extreme cases of hot and cold
scenarios. The thermal analysis was conducted using Finite Elements (FE), and the results were compared to the test data and validated.

Gasser Farouk Abdelal et al (2006) executed thermal analysis of small satellite structure by applying Finite Difference Method (IDEAS) and temperature profile for satellite components case is evaluated. Then the thermal fatigue analysis is performed by applying Finite Element Analysis (ANSYS) to calculate the resultant damage due to on-orbit cyclic stresses, and structure deformations at the payload and ADCS equipments seats.

The duty of a satellite thermal control system is to maintain the temperatures of all satellite components within their allowable operational temperature limits, throughout the satellite mission. The thermal control methods are classified as passive and active. Farhani et al (2007) presented the thermal design and analysis of a small satellite operating in Low Earth Orbit (LEO), having a passive thermal control system, and using thermal control hardware such as space qualified paints.

Modal analysis of satellite system is the basic requirement for the satellite structural design. Zhengfeng Bai et al (2008) modeled and analyzed a small satellite, of which the monolithic structure is a honeycomb sandwich plate, by using the Finite Element Analysis software PATRAN/NASTRAN. The results indicate that the Finite Element Method is a convenient and useful tool for modal analysis, which can calculate the natural frequencies and predict the modal shapes. The results of the analysis can accommodate bases for structural design and optimum design of the small satellite structure. In addition, the numerical simulation results can be supplied for modal analysis during the testing.
It is observed from the literature that the satellite structure is exposed to different environmental conditions and hence different approaches have to be considered while designing, analyzing, testing and verifying the satellite structure. The satellite structural design is highly dependent on the mission requirements and the launch vehicle characteristics. The design platform for the satellite structure has to reduce the cost, shorten the lead-time and shorten the design and manufacturing processes, the micro satellite bus design should be reusable and the design must be based on a flexible architecture or configuration that can integrate the bus and the payload and that can be reproduced with a minimum amount of modification and cost. Satellite structural designs also use several different materials and are chosen based on their properties, cost, and complexity.

It is also evident from the literature that the distribution of subsystem components within the satellite is critical because it determines the Center of Gravity and the Moments of Inertia about the satellite's principle axes. With the use of Finite Element Analysis (FEA) software packages it is possible to model structures mathematically in a great detail, and to examine their behavior under all the possible static and dynamic load conditions. The structural testing required to qualify an assembly for launch is often accomplished by subjecting a prototype to static and vibration loads in excess of those anticipated for flight and the actual flight unit is subjected to near flight levels. The thermal design for a satellite is to maintain the temperature of all the subsystems within their specified range of operation. Also, for evaluation of satellite thermal design it is necessary to obtain temperature distribution of all satellite components, using the test or analysis.

In the present study, a micro-satellite structure of dimension $569\times569\times600$ mm$^3$ and mass 35 kg was designed to accommodate 40 subsystems. The satellite was carried as a piggy-back payload in PSLV- C2
launcher; the theoretical and experimental studies were conducted to evaluate the structural behavior of the micro-satellite structure under various environmental loading conditions expected during the ground operations and during flight.