Transactions on Aerospace Research eISSN 2545-2835, VOL. 272, NO. 3/2023, 40-54
DOI: 10.2478/tar-2023-0016

BASIC PRINCIPLES AND MECHANICAL CONSIDERATIONS OF SATELLITES: A SHORT REVIEW

Reham Reda*, **Yasmeen Ahmed**, **Islam Magdy**, **Hossam Nabil**, **Mennatullah Khamis**, **Ahmed Refaey**, **Nada Eldabaa**, **Manar Abo Elmagd**, **Mohamed Abo Lila**, **Hady Ergawy**, **Alhussein Elgarf** and **Gerges Abed**
Mechanical Design and Production Engineering Division, Mechanical Engineering Department, Faculty of Engineering, Suez University, Suez 43512, Egypt

Abstract

Satellites are used for navigation, communication, oceanography, astronomy, etc.. Satellites come in a diversity of sizes and forms. Depending on the satellite's mission, different subsystems are used. These subsystems are installed inside a housing to protect them from the space environment. This housing, which is also known as the satellite primary structure or mechanical structure, is made of durable materials that can endure severe conditions during launch and in the orbit. The optimisation of satellite mass is crucial right now since satellites are losing mass every day to reduce the cost of manufacturing and launching. This review first introduces an overview of the satellite classifications and subsystems. Then, the different types of mechanical load analysis the satellite subjects itself to are demonstrated. The advanced approaches for promoting the performance of the mechanical structures of satellites are explored, with a spotlight on the effect of the optimisation parameters of isogrid and honeycomb sandwich structures on the mechanical performance of the satellite primary structure. The assembly, integration and testing (AIT) of the small satellite are briefly presented. Finally, the important potential designs to improve the mechanical performance of the satellite primary structure and the challenges of further research are summarised.

Keywords: principles of satellites; primary structure; isogrid structure; honeycomb structure; mechanical load analysis; assembly, integration and testing
Type of the work: review article

1. INTRODUCTION

A machine that is launched into space and orbits the Earth or other planets is referred to as a satellite. Some satellites capture images of the world that aid in weather forecasting and storm tracking. Some satellites photograph distant galaxies, the Sun, black holes, dark matter or other planets. These images aid in the scientific perception of the solar system and the cosmos. Although a satellite has many different subsystems, they all share at least two parts: an antenna and a power source. Information is frequently sent and received from and back to the Earth through the antenna. A solar panel or battery is utilised as the power source. Solar panels turn sunlight into power. Depending on the satellite's mission, other parts are also employed, including cameras and scientific sensors. Each subsystem in a satellite is identified and designed to perform a specific mission. The presence of additional subsystems affects the performance and specification of a single subsystem, which in turn affects the interfaces between subsystems [1].

* Corresponding Author: reham.reda@suezuni.edu.eg
ARTICLE HISTORY **Received** 2023-02-24 **Revised** 2023-06-15 **Accepted** 2023-06-20

Secondary structures, e.g., solar panels, thermal blankets and subsystems, are affixed to the primary structure, which acts as a housing. The primary structure transmits no significant structural loads. Although the integrity of the satellite is often unaffected by the failure of a secondary structure, the mission as a whole may be significantly impacted. When a primary structure fails, the mission fails catastrophically. Therefore, the design and manufacturing of the satellite primary structure are challenging.

When considering small satellite construction, material selection is critical. Both the physical and mechanical requirements must be met. Different subsystems and launch circumstances have an impact on structural design, as do the satellite application and intended environment. The exterior, interior and electronic subsystems of the satellite must be understood in the context of the particular mission environment. The material used to build the satellite must be capable of resisting space radiation and protecting its internal components from damage, be highly resistant to pressure from space travel and possess sufficient strength to withstand potential impacts from smaller asteroids or space debris. Due to the high level of precision required in selecting the appropriate material and the best fabrication technique, manufacturing the satellite is a costly task [2].

When manufacturing satellites and their parts, safety is one of the most crucial considerations. The highest industry standards must be met by each component of a satellite. A satellite cannot be produced or assembled with any potential for human error. Any faults or irregularities, from the smallest inside component to the satellite's surface, could be harmful [3]. The main manufacturing techniques used in space are additive manufacturing and computer numerical control (CNC) machining [4,5]. Using metal additive manufacturing technology for satellite component fabrication promotes weight reduction of parts through extremely effective and lightweight designs. Cost and time reduction, utilising novel materials and unique designs, and consolidation of several parts are also benefits of employing additive manufacturing techniques in space components for performance improvement or risk management [4]. Certainty is one of the most crucial factors when manufacturing parts for satellites or related structures. Whether the inside components or the exterior of the satellite could be severely harmed by any defects or inconsistencies, CNC machining is also considered an excellent production technology for the aerospace industry due to the extremely high level of precision required by this industry. This technique allows for the acquisition of extremely tight tolerances. As a result, it provides outstanding resolutions and aircraft functionability with the use of durable metallic and plastic components [5].

The primary structure of a satellite may have different geometries and configurations. A satellite may have different shapes and dimensions, i.e., cube or hexagonal, depending on its mission, payload and other subsystems. Traditional structure configurations for satellites include skin-frame structures, truss structures, monocoque cylinders and skin-stringers. Advanced structural configurations include isogrid and sandwich structures. The main target in designing the mechanical structure of the satellite is to promote a high strength-to-weight ratio to ensure lightweightness and high stiffness.

The aim of this review is to shine a spotlight on the basic principles of the satellite and the functions and requirements of the different subsystems, while also focusing on the different mechanical considerations of the satellite's primary structure. The mechanical loads and design, finite element analysis (FEA) and testing of the satellite primary structure are discussed. This will provide a thorough understanding of the satellite primary structure's requirements and function and help in attaining the missing points in research concerning the primary satellite structure's design, testing and manufacturing to be considered in future work.

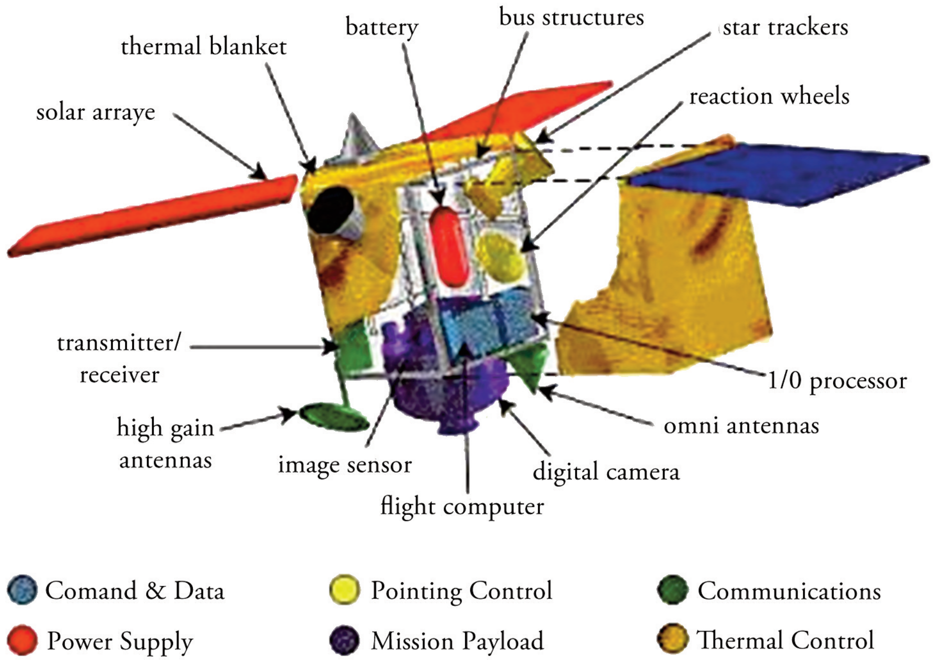
2. SATELLITE CLASSIFICATION AND ESSENTIAL SUBSYSTEMS

Satellites can be classified according to their mission or mass. Satellite missions include remote sensing, navigation, astronomy satellites or space telescopes, weather or meteorological satellites, communications and TV broadcasting satellites and early warning satellites [6–9]. Based on the mass, satellites are categorised into large satellites (exceeding 500 kg), minisatellites (100–500 kg), microsatellites (10–100 kg), nanosatellites (1–10 kg), picosatellites (0.1–1 kg) and femtosatellites (below 100 g) [10–12].

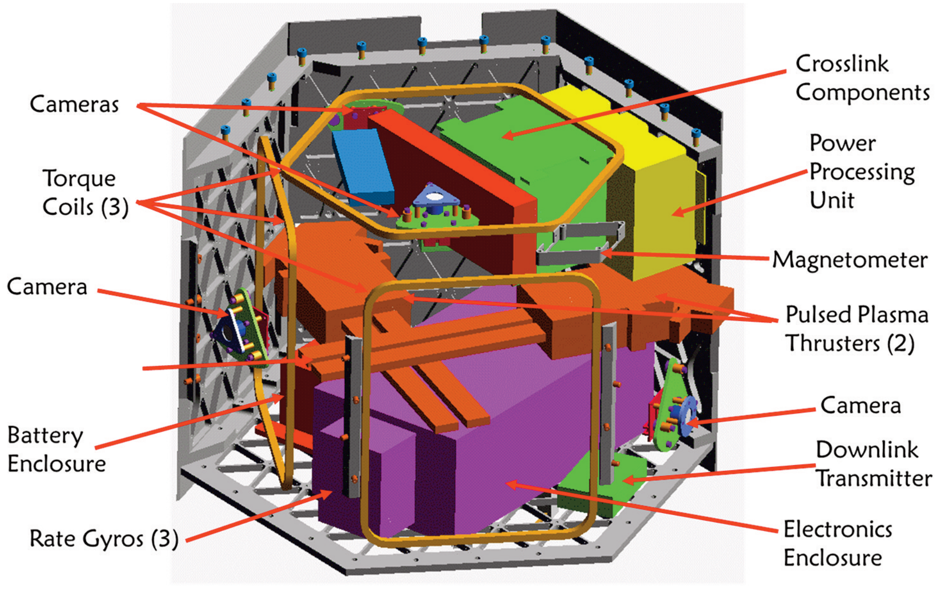
The essential satellite subsystems can be divided into platform subsystems and deployment devices. Regardless of the intended mission of the satellite, a satellite platform typically consists of the satellite bus and the payload. While the payload is responsible for the mission aspect of the satellite, the satellite bus controls the satellite and offers support aids to the payload [13].

Typically, a satellite bus includes the following subsystems: propulsion subsystem (PS); electrical power subsystem (EPS); telemetry, tracking and command (TTC) subsystem; attitude and orbit control subsystem (AOCS); command and data handling subsystem (C&DHS); and antenna subsystem, primary structure [1,14]. Minimum weight, low consumption and high reliability are the three main requirements for these subsystems [1]. Figure 1 presents different arrangements of the main satellite subsystems [15,16].

The thrusts needed to provide the required speed alterations to carry out all the motions over the satellite's lifecycle are provided by the PS. Three different propulsion system types are in practice. These include electric and ion propulsion, liquid fuel propulsion and solid fuel propulsion [1]. The propulsion system relies on the idea that by expelling mass with a certain velocity in one direction, thrust is produced in a different direction. In the case of solid and liquid propulsion systems, the creation of a high-pressure gas through the high-temperature breakdown of propellants is required for the ejection of mass at high speed. The high-pressure gas is then forced into a diverging-converging nozzle at supersonic speeds. Ion propulsion involves accelerating the charged plasma of an ionised elemental gas, such as xenon, in a very strong electrical field to create thrust [14].



(A)



(B)

Figure 1. Different arrangements of the main satellite subsystems: (A) remote sensing satellite [15] and (B) Ionospheric Observation Nanosatellite Formation [16].

The EPS's main job is to gather solar energy, convert it to electrical power using solar cell arrays and then deliver that power to other satellite subsystems and batteries. Additionally, the satellite is equipped with batteries that supply standby electricity at eclipse times, in other emergency scenarios, and while the satellite is being launched and the solar panels are not yet deployed.

From the time of launch until the end of the satellite's operational life in space, the TTC subsystem monitors and controls the satellite. The tracking component of the subsystem identifies the satellite's position and tracks its movement using angle, range and velocity data. The telemetry component collects data on the state of different satellite subsystems, encodes it and then transmits it. The command component receives and carries out commands from the remote control to change the platform's configuration, location and velocity.

There are two main tasks carried out by the AOCS. It controls the orbital path, which is necessary to guarantee that the satellite is in the right orbital position to deliver the desired services. Additionally, it promotes attitude control, which is necessary to keep the satellite from tottering in space and to keep the antennas fixed at a fixed location and direction on the surface of the Earth.

The C&DHS or on-board computer (OBC) is considered the satellite's brain and nervous system. It supplies the bus flight software with a platform as it gathers and prepares all space vehicle unit telemetry data, distributes the commands to or from the OBC, decodes them and puts them into action. The EPS and OBC are closely connected to monitor power consumption and availability for managing onboard satellite functions. The OBC is also responsible for receiving, interpreting and carrying out orders from ground operators. It additionally transmits telemetry data packets to the ground station, providing the ground-based operators with a common overview of the health and state of the satellite.

Antennas are used to send and receive signals from ground stations. The primary structure is used for the equipment support, and it should have high rigidity and lightness.

The most crucial component of any satellite is the payload subsystem, which is the component of the satellite that carries the instruments needed to carry out its intended job. Any satellite's payload will vary depending on its mission. To carry out its job, a satellite may contain a single kind of payload or multiple kinds. The transponder, which serves as a receiver, amplifier and transmitter, is the basic payload in the case of a communication satellite. A radiometer is the most crucial payload for a weather forecasting satellite. The payloads of scientific satellites vary based on the mission, including spectrometers, plasma detectors, magnetometers, telescopes and others. The primary payloads aboard a remote sensing satellite are high-resolution cameras, multispectral scanners and thematic mappers. An electro-optical (EO) camera, for instance, can be carried on a remote sensing satellite as part of its payload to take photos of the globe during the day and transform them into electrical signals that can be collected. As an alternative, the camera may have microwave sensors that will allow the payload to detect the radio frequency waves from the planet at various radio frequencies or infrared sensors that will allow the payload to view the Earth after dark. Sometimes, for the payload to carry out its duty, the satellite bus must comply with specific constraints imposed by the payload's operational requirements. The incumbent satellite builders have historically created and implemented a large number of nonstandard interfaces as a result of the reliance between satellite bus and payload subsystems. Because of this, the satellite bus framework in the aerospace industry has been trending towards greater standardisation, which has the potential to significantly reduce costs [13].

Satellite deployment mechanisms include restraints and deployment devices. These mechanisms are able to retain and deploy the parts, i.e., solar panels and antennas. These parts would initially be retained at the facets and travel to the orbit in a compact, folded configuration. Then, the deployment of these parts occurs through certain mechanisms [17,18]. The electric burn wire is one frequent release mechanism (Fig. 2) [17]. In this mechanism, the nichrome burn wire applies a force and a stroke in this release mechanism, which makes use of a compression spring system. When a consistent current is applied to the nichrome wire, it thermally cuts through a Vectran tie-down cable, releasing the deployable that it had been securing [16].

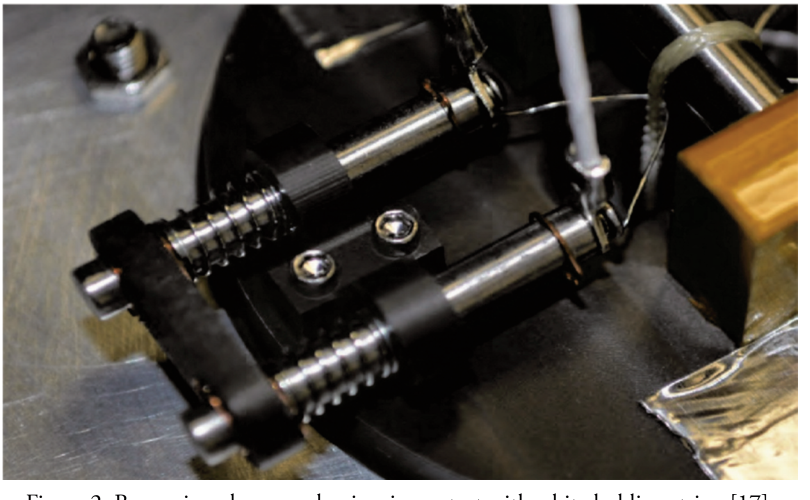
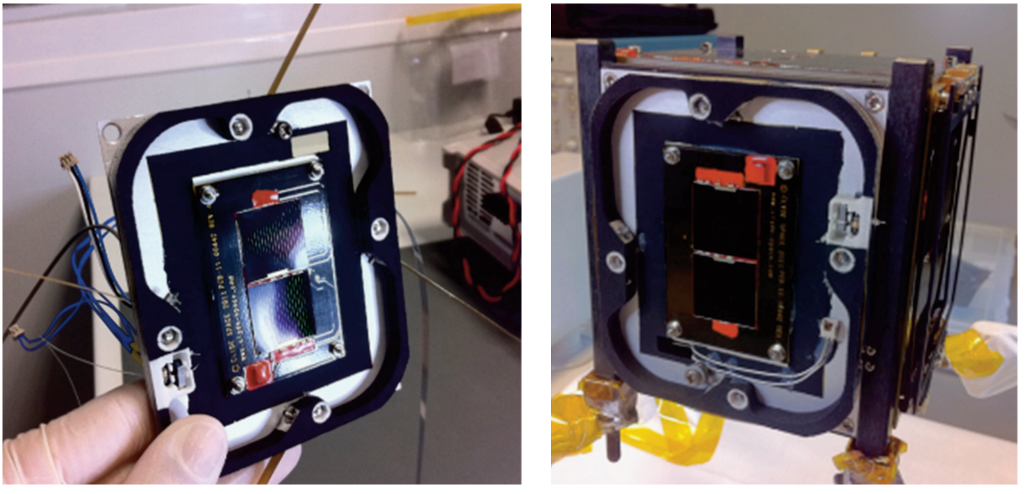


Figure 2. Burn wire release mechanism in contact with white holding string [17].

For deployable antennas, a unibody portion, i.e., the sub-chassis, is constructed onto which all the deployment mechanism's components, including those for retention, burning and opening detection, could be mounted. The antennas are totally encased in a static-dissipative plastic covering that keeps them pressed up against the sub-chassis. Drilling a hole through the film and the antennae underneath it at one end allows access to the burner receptacle, which contains the components for burning and opening detection. Then, a nylon thread is passed through the hole, over the burner resistance and over the detecting switch, which is kept closed by the tight thread. The retention system is then sealed off while being connected to the sub-chassis. The resistance is energised to cut the thread after lift-off and arrival in the orbit have been accomplished, and thus, the antennae are extended (Fig. 3) [18].



(A) (B)

Figure 3. Antenna deployment mechanism: (A) deployed and (B) folded [18].

3. MECHANICAL CONSIDERATIONS OF THE SATELLITE PRIMARY STRUCTURE

3.1. Categories of small satellite structure and geometry

Structure and its mechanisms in a small satellite should be able to attach the satellite to the launch vehicle, mechanically support all other satellite subsystems and provide all the functional capabilities needed in orbit. The satellite faces harsh conditions during its lifetime. The material of the primary structure must meet the requirements of specific stiffness, hardness, toughness, ductility and fatigue strength. To endure the static, dynamic and thermal stresses that arise during launch, deployment and service, the structure and its supporting mechanisms must be designed to satisfy these requirements. Throughout the flight, the structure should shield the payload and other sensitive electronic components from severe distortions, vibrations, temperature fluctuations and unwanted radiation. All of these constraints must be taken into account during the early stages of structural design. The reduction of structural weight in accordance with the necessary reliability level is a significant problem in structural design.

The satellite structure includes three categories: the primary structure, or main structure; the secondary structure; and flexible appendages. The payload and its accompanying equipment can be attached to the primary structure at several points. When the main structure fails, the satellite completely is destroyed. The secondary structure supports parts such as solar panels and thermal blankets and attaches them to the primary structure. The integrity of the satellite structure is unaffected by the failure of the secondary structure, but if it affects the thermal control or crosses an optical path, it may have significant effects on its mission. Flexible appendages, including antenna reflectors and solar arrays, are the third form of the satellite structure. These structures often have low resonant frequencies that interact with the satellite's dynamic behaviour; therefore, they necessitate careful design [19].

Satellites are made with different primary structure geometries, i.e., cube, rectangle, hexagonal and cylindrical. A satellite may also be made up of one, two, three or more units for more complex missions.

In terms of geometry, the most famous small satellites are cubic satellites (CubeSats) and hexagonal satellites. Figure 4 shows the different geometries and sizes of the small satellites [20–22]. In two ways, CubeSats drastically reduce launch costs. They weigh little, so a rocket needs less fuel to lift them. Additionally, they frequently take off on the same rocket as a bigger satellite. However, the CubeSat design presents some difficulties. The electronics are more radiation-sensitive since they are smaller. CubeSats cannot transport heavy payloads due to their small size. Due to their low price, they are frequently made to operate for only a few weeks, months or years before failing [23]. Hexagonal satellites reduce the integration's inherent difficulties. The satellite could be separated uniformly, thanks to its hexagonal shape [16]. Small structural modifications can free up valuable space for additional subsystems and components.

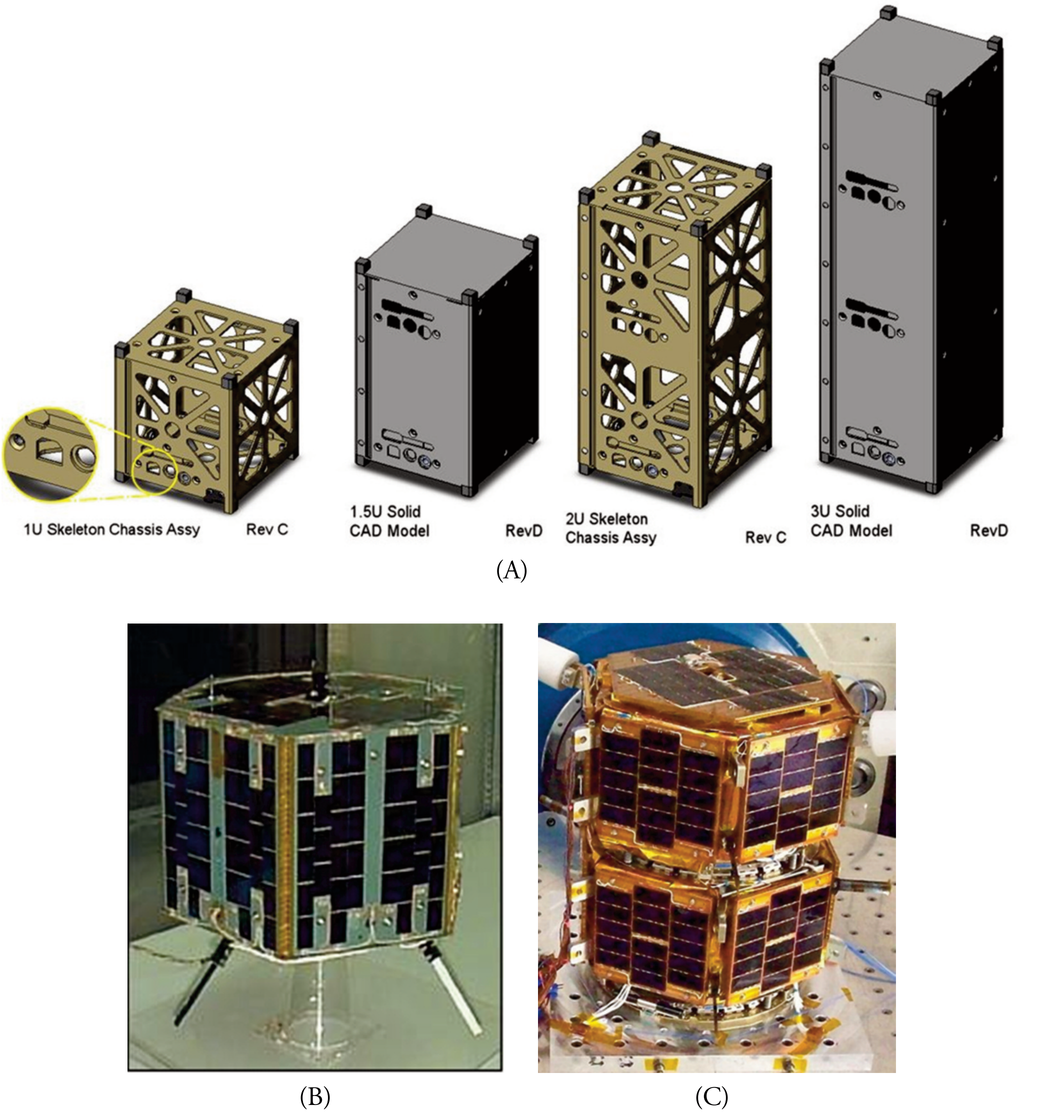


Figure 4. Different geometries and sizes of small satellites: (A) CubeSat standardisation [20] and (B) one-unit hexagonal satellite [21] and (C) two-unit hexagonal satellite [22].

3.2. Structure and mechanical design requirements

When a satellite is created on the Earth, launched into space and then released into space, it is subjected to drastically varied loading conditions, i.e., static loading, dynamic loading and mechanical shocks. Throughout the various stages of a launch, a variety of parameters, including vibrations, thermal expansion and accelerations, can cause plastic deformations and failure. In terms of mechanical design, the satellite must withstand forces acting individually or in combination. A successful satellite mission can be achieved by selecting the right design that can withstand all of these loading scenarios. The loading conditions must be recognised and carefully calculated to design a durable satellite.

In static loading, stresses are created during component assembly, such as pre-stress in bolts. Static loading includes longitudinal and lateral accelerations in a steady state during takeoff and, moreover, thermal loads, such as air friction on the rocket and temperature rise during engine operation. Mechanical shocks occur by means of separation mechanisms that allow the satellite to be released.

Dynamic loading includes low-frequency vibrations, random vibrations and acoustic loads. Low-frequency vibrations, often known as sine vibrations (SVs), are generated by running engines. Random vibrations generated from the mixing of the exhaust with the atmosphere, the turbulence in the boundary layer and engine noise and vibrations during lift-off and flight are transmitted to the satellite as mechanical vibrations of a random nature. Acoustic loads are the noises created by the engine during takeoff and flight, the exhaust with the atmosphere mixing, air rubbing against the rocket and forces acting inside the fairing, the cavity in which the satellite is placed.

Therefore, the mechanical structure of the satellite should be capable of withstanding the mechanical stresses that it must undergo. It should also be simple to manufacture and assemble. Locating the centre of gravity (CG) more towards the geometric centre is what the design must concentrate on. For ease of operation, the number of fasteners (such as screws, nuts, spacers and metal bars) connecting subsystems, solar panels and the main structure should be kept to a minimum. Each variation in payload should be addressed with a unique structure design. The satellite's internal volume should be increased, and its external volume should be modular so that a deployable solar panel can be added. Additionally, during integration, the internal volume of the satellite should be made accessible. The mechanical structure of the satellite should also propose a lightweight solar panel deployment solution, and a maximum solar angle should be obtained after deployment. Passive deployment and locking mechanisms should be present on the solar array. The hinge mechanism's safety factor should be greater than 2.5 to satisfy all design requirements [16,23].

3.3. Materials and properties of the satellite primary structure

One of the crucial stages in the design of a satellite structure is material selection. Since the satellite's weight is a crucial consideration for an on-orbit object. Several materials are used in satellite structural designs. Metals and fibre composites are the two common materials used in satellite applications. When designing a satellite, factors such as specific strength, specific stiffness, thermal characteristics, manufacturability and cost are the criteria for selection in addition to weight.

There are several other material requirements that should be taken into consideration when designing the primary structure of the satellite [24]. At first, the materials should be space-grade and selected from the National Aeronautics and Space Administration (NASA) list of satellite materials. When choosing a material for a structural purpose, thermal conductivity and thermal expansion coefficients are crucial considerations. Thermal insulation or conduction is frequently a secondary function of a satellite's structure, making thermal conductivity crucial. Another crucial factor is the coefficient of thermal expansion. When two materials with different thermal expansion coefficients are employed in the same structure, significant thermal stresses can be produced. Therefore, it could be preferable to reduce thermal expansion for subsystems and payloads. The directionally dependent positive and negative expansion coefficients of composite materials can be used for this purpose. The structure material should also have a thermal expansion coefficient that is comparable to the material used in the deployment mechanism. The material's yield strength ought to be greater than the maximum von Mises stress. Low-density materials should be chosen to reduce mass. The material ought to be simple to manufacture. It is best to choose a material with low out-gassing characteristics in compliance with the out-gassing standard established by NASA.

When designing satellite structures, it is crucial to take manufacturability into account. While being manufactured, certain materials such as beryllium and aluminium–lithium alloys may pose dangerous circumstances. Late alterations frequently take place as the various components are being assembled; therefore, the designer should permit such changes to be made at the assembly level.

All structures are thought to contain some degree of microcracking. Every crack tip is a point of stress concentration, and if the local stresses are sufficiently strong, cracks will propagate. The longest crack length that will not spread at a given stress level is known as the critical crack length. Throughout the structure's service life, the designer must make sure that failure due to these faults does not occur. Additionally, before launch, non-destructive testing methods must be utilised to ensure that there are no cracks larger than the critical size. It should be noted, however, that due to the limited mission duration, fatigue is not a dimensioning parameter for most commercial satellites.

The most common metallic materials utilised in the manufacturing of satellites are aluminium alloys [25]. They make up the majority of the components required to construct a satellite. It is the ideal option for the majority of uses due to its high strength-to-weight ratios, high stiffness-to-density ratio, outstanding workability, simplicity of machining, non-magnetism, reasonable cost, high ductility, high corrosion resistance and availability in a wide range of forms. Both steel and aluminium have stiffness-to-weight ratios that are close, but aluminium often has a greater strength-to-weight ratio. Limited yield strength, low hardness and a high coefficient of thermal expansion are drawbacks of aluminium alloys. To improve the material's strength, alloys are often tempered. 6061-T6 and 7075-T7 Al alloys are two typical alloys used in manufacturing. Silicon and magnesium are present in aluminium 6061-T6, which strengthens the alloy after tempering. Zinc and magnesium are included in trace levels in aluminium 7075-T7; this alloy is stronger than 6061-T6, and it is more difficult to machine [25]. Aluminium–lithium (Al–Li) alloys possess the ability to reduce the weight of launch vehicles by as much as 30%. These materials are highly weldable, have a tensile strength over 100 ksi and offer greater cryogenic strength than any other aluminium alloy, which is a crucial factor to take into account for cryogenic fuel tanks.

Applications requiring extremely high stiffness in aerospace use beryllium. Its specific modulus is 6.2 times greater than that of aluminium [26]. Due to its grain alignment, the material is non-isotropic and has low ductility and fracture toughness in the short-grain direction. Because it operates well at cryogenic temperatures and because it has a high thermal conductivity and a low coefficient of thermal expansion, it is frequently utilised in lightweight optics and mirrors. However, beryllium is costly, challenging to manufacture and scarce. Because its powder is a recognised carcinogen when inhaled, beryllium must be processed under supervised conditions. After being machined, the components can be handled safely [27]. This material has a wide range of possible uses due to its stiffness-to-weight ratio, which is six times better than that of titanium or aluminium and has a density of about 60% that of aluminium. Since it is stiffer than other materials, it can be helpful in preventing resonance frequencies that might happen during launch between a satellite and its launcher. It possesses a high elastic modulus (44 Msi), a low magnetic susceptibility and a high yield strength. Since beryllium has a high thermal conductivity, it is a superior material for heat-conducting components and can significantly reduce weight when used in place of aluminium. However, beryllium is twice as brittle as aluminium and exhibits much higher anisotropy and damage sensitivity. One significant drawback is that it has rather poor fracture toughness at cryogenic temperatures; however, developments in beryllium–aluminium alloys may help address this drawback. Because of the toxicity of its dust, it is also exceedingly expensive and requires specialised equipment and tools for milling.

For applications requiring extremely high-strength materials, titanium and its alloys have been used. These materials have excellent corrosion resistance, low coefficients of thermal expansion and high strength-to-weight ratios [25]. Titanium is a non-magnetic material that is frequently utilised in applications where aluminium lacks the necessary strength. It is well suited for low-temperature applications and has a significantly higher yield strength, despite being a little more difficult to machine. It also has a higher stiffness-to-density ratio and is suitable for cryogenic applications as well as cryogenic fuel storage. Titanium performs better than aluminium at high temperatures. However, titanium alloys have low fracture toughness and are hence challenging to machine. The most common titanium alloy used in aerospace applications is Ti-6Al4V; this alloy is used to make missile bodies and wings. Furthermore, the castings used to connect the external fuel tank to the space shuttle and its rockets are among their most well-known applications [24]. Intermetallic titanium aluminides are a type of relatively recent titanium-based material. These low-density materials are strong at temperatures above 700°C and have outstanding oxidation resistance. The main applications are in honeycomb structures and as a composite matrix material. Unfortunately, titanium aluminide reacts badly and fractures when exposed to hydrogen. This limits their usefulness in vehicles that actively cool the airframe with hydrogen.

Fibre composite structures are composed of a matrix (metal or epoxy) and a reinforcement (carbon or graphite). Due to their high specific modulus and distinctive load path, these composite constructions are extremely efficient. These structures are three to five times stiffer than aluminium at 60% of the mass as the flexural shear loads are transferred from the matrix to axial loads on the high-strength fibres [27]. Composites are advantageous for damping unwanted vibrations since their stiffness-to-weight ratios are superior to those of all metals. They enable the construction of structures that will not deform in the climatic extremes of space, thanks to their negative axial coefficient of thermal expansion. They also offer lightweight thermal management and heat sinks as their thermal conductivity is greater than that of copper. For high-temperature applications, including re-entry vehicle skins and metal–matrix, carbon–carbon and ceramic–matrix composites work well since they can tolerate temperatures above 1,370°C without active cooling.

Composite materials still have significant shortcomings. Both efficient oxidation coatings and manufacturing methods for large-scale structures must be developed. The way laminated composites respond to temperature variations is another issue. The differing expansion rates between the fibres and matrix can result in significant internal strains when temperatures change. Temperature differences can cause warping, which is more obvious in the isotropic material. By adding conductive strips, which raise the structure's mass, electrical systems are grounded. The integrity of the fibres can be destroyed, and the composite rendered useless by nicks and dents that can be ignored or fixed in structural metals. Composites are usually deemed too unreliable for use as more than secondary construction in space, where there is little to no inspection and maintenance and where failure of a primary structure can have catastrophic implications. But using composites in secondary structures wisely can still save a lot of material.

4. PRIMARY STRUCTURE CONFIGURATION DESIGNS

4.1. Traditional structural configuration designs

Several criteria are considered when designing the primary structures of the satellite based on the requirements of the mission. The traditional primary structural designs of the satellite include skin-frame structures, truss structures, monocoque cylinders and skin-stringer structures [27].

4.1.1. Skin-frame structures

The skin-frame structural concept mounts outside skin panels with fasteners or rivets, utilising an interior skeletal network of axial and lateral frames. Bending, torsional and axial forces are supported by these frames, while the skin strengthens the structure by sustaining the shear stresses induced by the connections between the inner elements. Despite the thin skin's tendency to cause some structural instabilities, the skin's thickness is occasionally decreased to minimise the total mass. All extra shear loading is transferred to in-plane tension forces at 45° when the skin buckles by shear, and the connections are then required to support these stresses. Excessive deformations occur in the skin through buckling modes, rendering it inappropriate for mounting outside components like solar cells. Adding intermediate elements usually improves the assembly's buckling strength.

4.1.2. Truss structures

For stability, truss members are often fabricated individually and organised in triangle arrays. Extruded tubes composed of metallic or composite materials are used to fabricate the truss members. A stable truss contains no extra members that could generate additional load paths and is clearly recognisable. Trusses are often mass-efficient when the members are arranged into cross-sectional assemblies with a rectangular or triangular cross-section. However, if the cross-section is more circular or hexagonal, they become less effective. Inherent stress concentrations are also produced by the structure's design at interface mounting locations, such as separation systems. Because there are no shear panels, components can be installed both internally and externally, and a payload is easily accessible. However, satellites that use body-mounted solar cells will not benefit from this absence of shear panels.

4.1.3. Monocoque cylinders

Monocoque cylinders are composed of axially symmetric shells without frames or stiffeners. Sandwich or metallic panels with rolled, curved sections are used to make the shells. Usually, the cylindrical form is created by fabricating and joining two or three curved parts. Monocoque cylinders' buckling strength typically sets a limit on their strength. When the loads are evenly spread throughout the structure, the shells perform more efficiently. Fasteners are generally used to mount components to the walls, but it is important to take care not to overload the shell and lead to local failures. A satellite with body-mounted solar cells and reasonably light components can use the monocoque cylinder design.

4.1.4. Skin-stringer structures

Axial and lateral frame components attached to an outer skin are used in the construction of cylindrical skin-stringer systems. These designs resemble skin-frame structures, although this category of structures only includes circular cylinder arrangements. Despite the thin skin's tendency to cause some structural instabilities, the skin is occasionally thinned to reduce mass. Through the diagonal tension phenomenon, the post-buckling behaviour of the skin converts the additional applied shear stresses to torsion. To allow the assembly to operate as a uniform surface, the skin and components must attach consistently. Commonly used connecting techniques include rivets and/or fasteners. Along the stringer assembly, interior parts are often fastened. When applying local loads, this approach is more effective than mounting monocoque cylinder parts. The skin must be sufficiently rigid to allow for the secure installation of exterior components such as solar cells on the body.

4.2. Advanced designs for satellite structure

Several configurations have been developed to replace the traditional structural configuration designs, aiming at satisfying the mechanical design requirements of the satellite primary structure. The current review focuses on the isogrid and honeycomb sandwich structures.

4.2.1. Isogrid structure: the concept and structural optimisation

Isogrid improves a structure's stiffness per weight by utilising an array of equilateral triangular cutouts. The geometric parameters of the isogrid structure are presented in Figure 5 [28]. A metallic panel may be machined to create the isogrid configuration, or fibre composite materials may be used to build it. Since the idea was first developed when NASA released a report titled 'Isogrid design handbook' [29], isogrid structures have been considered. This handbook examines the isogrid structure manufacturing process and mechanical testing. When these structures are subjected to an axial compressive load, the failure may be caused by global buckling if the panel is slenderer than one of the ribs or local buckling if the ribs are slenderer than the panel [29]. The development is still ongoing nowadays, with the majority of research going into composite applications.

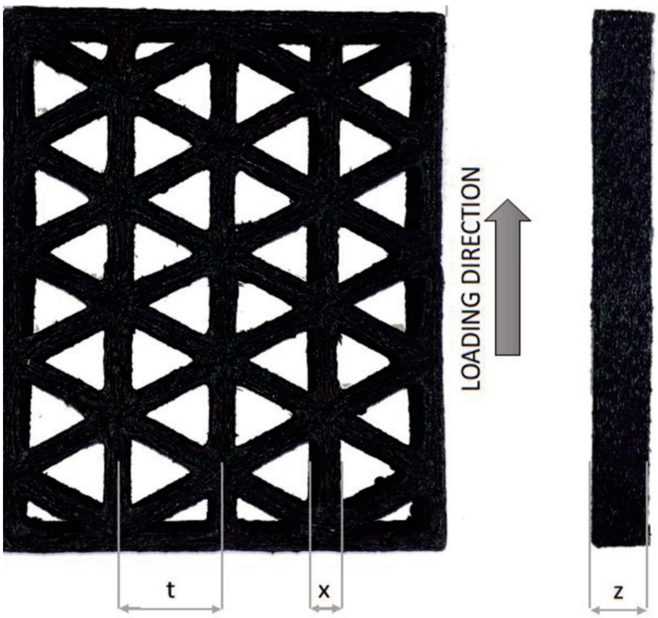


Figure 5. Geometric parameters of the isogrid structure, where t is the rib height, x is the rib width and z is the rib thickness [28].

Since composite materials have high specific characteristics, their employment in the aircraft sector is a well-established practice [30]. The structures may be subjected to significant axial compressive loads in some applications, such as propellant tanks for rockets, fuselages and boosters, which could result in the composite panels failing due to buckling processes [31]. Composite stiffening ribs can be utilised to add stiffness and strength to prevent this failure. These stiffeners are often created as isogrid structures or equilateral triangles and attached to the laminates [32].

A few research studies have focused on the design optimisation of isogrid structures to enhance their particular strength and stiffness. Finite element simulation was used by Zheng et al. [30] to optimise a stiffened cylinder with a height of 6 m and a diameter of 4 m. They stated that due to the quadratic relationship between critical force and thickness, the rib thickness should be greater than the rib width.

Finite element simulation was also used by Dawood et al. [33] to investigate how the isogrid technique changed the structure's natural frequencies. Due to the lower mass of the isogrid structure, the analysis results showed that the natural frequencies estimated for the isogrid case were greater than those for the non-isogrid case.

By using additive manufacturing, Forcellese et al. [34] created isogrid panels in short carbon fibre-reinforced polyamide with varying rib thickness and rib height values. Through compression experiments performed at room temperature, they examined how geometric parameters such as rib thickness and rib height affected the isogrid panels' compressive strength and buckling behaviour. They found that increasing rib width causes a rise in the peak load and specific maximum load, while increasing rib height causes a decrease in the maximum load and specific maximum load. They also observed that the global buckling failure mode caused the isogrid panel to fail during compression testing, indicating that the structure's slenderness is higher than that of one of the ribs.

The buckling and environmental performances of composite structures produced by 3D printing were also studied by Forcellese et al. [35]. In their comparison, solid, dried solid, isogrid with 3 mm rib width and dried isogrid with 5 mm rib width were all employed. They found that the lowest environmental loads and lowest buckling resistance are exhibited by isogrid with 3 mm rib width. The results for the solid and the isogrid with dried panels that have 5 mm rib width in terms of buckling resistance are quite close. The latter, however, has worse environmental burdens than the former.

The effects of rib width and rib thickness on the strength and specific strength of polyamide reinforced with 20% of the weight of short carbon fibre were examined by Ciccarelli et al. [28]. They found that an increase in rib width causes a drop in specific strength and an increase in overall strength, while an increase in rib thickness causes an increase in the overall strength and specific strength, though this impact is less pronounced when specific strength is considered. They also noticed that depending on the geometric parameters, isogrid structures can fail under local or global buckling; the configurations that result in the highest strength are different from those that result in the highest specific strength, and isogrid structures that experience a local buckling failure mode exhibit lower specific strength than anticipated.

4.2.2. Honeycomb sandwich structure: the concept and structural optimisation

A sandwich structure is made up of two thin face sheets connected to both sides of a lightweight honeycomb core. Sandwich structures are designed so that the core can withstand normal flexural shears, while the outer face sheets can support axial loads, bending moments and in-plane shears [27]. Due to the heterogeneous nature of the core/face sheet assembly, sandwich structures are prone to failure due to high local stress concentrations. The point loads from connections must consequently be distributed during component mounting using potted inserts. Aluminium or composite panels made of graphite and epoxy are frequently used to manufacture sandwich panel face sheets. Typically, an aluminium honeycomb structure is used to manufacture the core. The lightest solution for situations involving compressive or bending loading is a honeycomb sandwich panel (Fig. 6) [36]. The geometric parameters of the honeycomb sandwich structure are shown in Figure 7 [37]. The honeycomb geometry is anisotropic, and the longitudinal direction exhibits increased stiffness [38]. However, when put together in a sandwich arrangement, the core behaves almost isotopically for in-plane loads. The mounting requirements for potted inserts and the thermal inefficiency are drawbacks of using honeycomb cores. These inefficiencies are due to the limited thermal conductivity of the adhesive layers utilised in fabrication, which makes the honeycomb structure unsuitable for optical and mirror aircraft applications.

Thin strips are shaped into honeycomb cells to create honeycomb sandwich cores. It is quite challenging to mill honeycomb core due to several of the required characteristics that make it appropriate for a wide range of applications. The intricate procedures required by the honeycomb composites' unique shape make them different from other CNC machining processes. In addition, honeycomb cutting frequently necessitates substantial postprocessing to eliminate partially released flags along the machined edge of the core walls. Specialised cutting tools that are more expensive than regular CNC machine parts are frequently needed for honeycomb machining. Additionally, the complex honeycomb machining process with fewer errors can be achieved with a skilled machinist partner [39]. Applying the right and properly maintained tools is a crucial aspect of successful honeycomb machining. To create the intricate hexagonal structure required in the honeycomb core panels, all cutting tools for honeycomb CNC machining must be manufactured using hard materials and kept at a high level of sharpness. For honeycomb machining, polycrystalline diamond (PCD)-tipped milling cutters, PCD-tipped turning tools, flute cutters and 5-axis CNC machines are the best options.

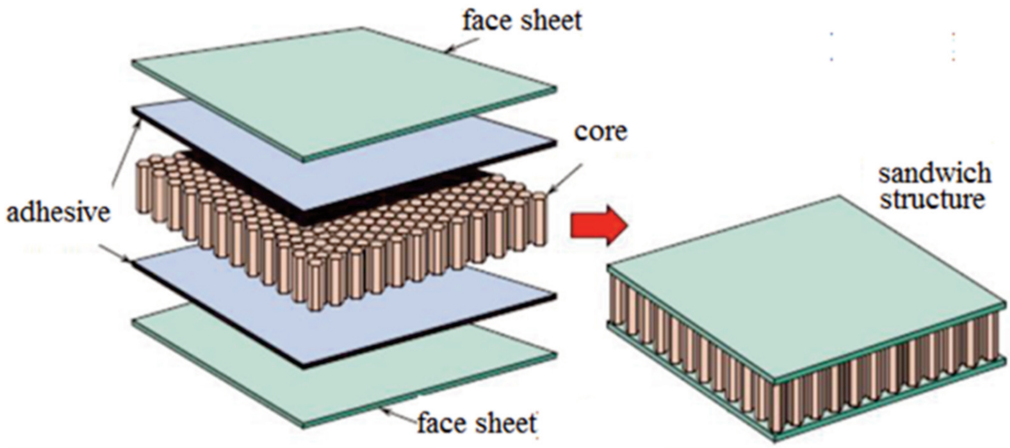


Figure 6. Honeycomb sandwich panel structure [36].

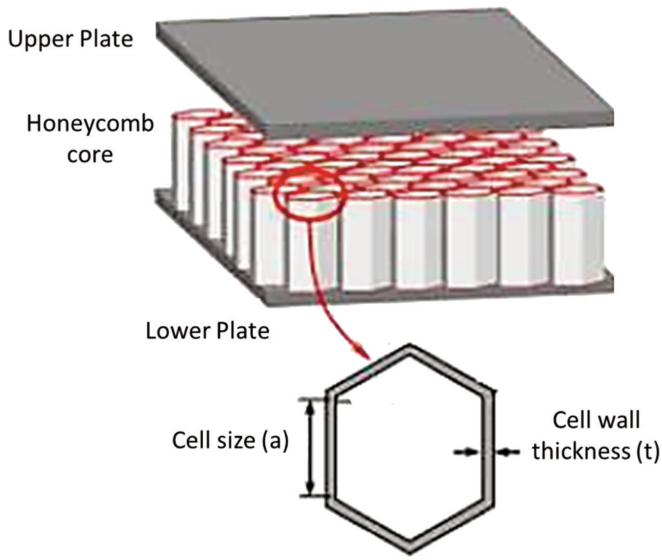


Figure 7. Geometric parameters of the honeycomb sandwich structure [37].

The crushing strength of the kraft paper honeycomb with varying cell wall thickness, cell size and density values under compression loading was numerically examined by Abd Kadir et al. [40]. They found that crushing strength increases as cell size decreases, with 10-mm cells showing the highest strength. Meanwhile, the honeycomb has better crushing strength, thanks to its 0.4-mm-thick cell walls. Additionally, denser materials have greater crushing strength.

The effect of honeycomb design parameters on the forced vibration behaviour of aircraft sandwiches with honeycomb cores was investigated experimentally and numerically by Sadiq et al. [37]. To examine how honeycomb structure parameters affect the transient response of sandwich structures, core height, cell size and cell wall thickness were considered. They found that raising the cell wall thickness from 0.8 mm to 1.5 mm leads to an increase in the maximum transient response, while increasing the core height from 5 mm to 25 mm leads to a decrease in the maximum transient response. Additionally, they found that increasing the cell size had a detrimental impact on the response's maximum transient. According to their findings, variation in the core height has a larger influence than the other factors, although the cell wall thickness has a lesser influence.

Aluminium honeycombs' mechanical responses to in-plane and out-of-plane compression were investigated by Li et al. [41]. They noticed that the walls of aluminium honeycombs were thickening. Some significant observations about the quasi-static and dynamic crushing responses of honeycomb structures were established by Thomas and Tiwari [42]. The face-sheet material and thickness have a significant impact on the core's ability to absorb energy during dynamic projectile and impulsive impacts. The performance of the honeycomb core under static stresses applied in various directions was influenced by geometrical factors such as cell size, node length, cell wall thickness and cell configuration. Hexagon, triangle, square and circular honeycomb cell configurations were chosen based on minimal material requirements and maximum strength standards. The ideal configuration for maximum strength and the lightest material was hexagonal honeycomb.

5. ANALYSIS OF SATELLITE MECHANICAL STRUCTURE

The primary structure of satellites is subjected to different types of loads, such as static, dynamic, transient and shock loads. There are various forms of mechanical analysis required to ensure the validity and integrity of a satellite structure. These analyses are carried out using finite element modelling (FEM) and include modal analysis, random vibration analysis and harmonic response analysis.

5.1. FEM

In all manufacturing processes, the first step should be the FEM of a product. FEM is a numerical method used by different simulation programs to solve partial differential equations in the mathematical model that defines the engineering model. One of the typical fields that employ FEM is structural analysis. The basic procedure of FEM is as follows:

1. Defining the material and engineering model
2. Meshing is simply the discretisation of the engineering model into small areas, which comes with a few advantages such as (a) complex geometry can be made simpler and (b) if a model has different material properties in different areas, this can be dealt with by discretisation. Mesh optimisation is an important step in building FEM; in some cases, a very fine mesh is required due to the model's complexity, while in other cases, a very fine mesh is not really essential and would only lengthen the solving process without improving the results.
3. Setting the boundary conditions: the manual for the launch vehicle or certain previously released NASA flight data can be used to determine the boundary conditions of the FEM. A complete knowledge of the physical phenomena that the satellite encounters at each step from manufacture to life in space is essential to accurately determine the boundary conditions and to check and validate the FEM model. As a result, the computer analysis results will be more accurate.
4. After creating a mesh and defining the boundary conditions, the program starts to calculate the solution using some approximation of the partial differential equations.

Verification and validation are important steps in any FEA. The boundary conditions used in vibration analyses should be correct to produce accurate results, such as the contact behaviour of different bodies in the structure and the load on the structure from fasteners, and where a mesh refinement should be placed to help optimise the analysis time. So, verification is needed at every step to see if the setup of the simulation is designed as per the specified requirements. Validation at the end of the FEA is made by comparing the analysis results with the actual physical tests.

5.2. Mechanical load analysis of the satellite structure

Both static and dynamic mechanical loads are present in a satellite. Dynamic loads change throughout time, whereas static loads are stable or unchangeable. These loads may also be external or self-contained. Transient and shock loads also occur.

5.2.1. Static loading analysis

Static loads, which include static external and static self-contained loads, can be produced from a variety of sources. The external loads that are quasi-static loads are due to the structural modules' and the supported components' inertia from steady acceleration or gravity. A steady-state acceleration remains constant as time changes, and that comes from gravity when handling the satellite. The mechanical preloads known as static self-contained loads include thermo-elastic loads produced by temperature variations and purposeful internal loads created during assembly [43]. Static load tests are intended to simulate the launcher, motor and spin-induced static loads imposed by static and dynamic accelerations. Due to launcher acceleration and aerodynamics, satellites are subjected to simultaneous static and dynamic loads throughout the launch process. A satellite must be designed to endure these loads using FEA. For a better understanding of the structural dynamics of the component, more transient analysis methods have been required but at the expense of runtime and large data files. So, a typical method in design is to combine static and dynamic loads into an equivalent static load known as a quasi-static load to validate the satellite's structural design. Whiffle-tree tests, centrifuge tests and acceleration with vibration test systems are the three methods of static testing that are employed [44]. One of those tests involves applying quasi-static loads to the component, utilising vibration test devices to simulate the loads the launcher adaptor exerts on the structure.

5.2.2. Dynamic loading analysis

Different sources such as dynamic external loads, dynamic self-contained loads and thermo-mechanical loads can generate dynamic loads on a satellite structure. Engine power, sound pressure and wind gusts are some examples of dynamic external loads that might occur during launch, as well as time-varying forces brought on by vibration while being transported to the launch site. Dynamic self-contained loads can be imposed by mass loading a vibrating satellite while it is being environmentally tested or while it is in space after the force that created the excitation has been withdrawn. On-orbit loads that result from thermal cycles as a result of the on-orbit environment's experience are considered of the thermo-mechanical load type. Other dynamic loads are brought about by satellite manoeuvres during the orbital or attitude correction stages [43].

5.2.2.1. Modal analysis

Modal analysis examines a structure's dynamic properties in the frequency domain, utilising a variety of sensors, data transmission devices and a computer to display and analyse the output data. This can be performed by placing the satellite on a shaker, then monitoring and recording its vibrations. Modal analysis identifies various periods at which a satellite structure will naturally resonate and provides the boundary conditions for the vibration analysis to ensure that the satellite does not fail due to resonance.

The basic data required for the modal analysis include defining the following: (1) the material of which every part is made, (2) the connections between various parts in the structure, (3) the launch vehicle conditions to determine the loads and fixtures for the boundary conditions setup and (4) the analysis settings, which can come from an experienced engineer [45].

5.2.2.2. Random vibration analysis

A vibrating table is used in the random vibration test to stimulate the test component. Typically, a stationary random procedure is used to determine the random frequency spectrum, utilising data from a launch vehicle manual and load coupling analysis [16]. The design of the primary and secondary structures of satellites should be designed to ensure that their fundamental and lateral frequencies satisfy the minimum prescribed values of the launch vehicle. Using frequency-domain analysis, one can perform a random vibration response with the assumption of having vibration loads in all directions.

Avoiding resonance is the main issue with random vibration analysis. The results of the modal analysis, which determine the natural frequencies and mode shapes, are employed as the boundary conditions of the random vibration analysis. Power spectral density (PSD) must also be provided in addition to the results of the modal analysis to perform the random vibration analysis.

The PSD value equals the limit of the root mean square of a random variable. A PSD spectrum is a statistical indicator of how a structure responds to random dynamic loading conditions. In the PSD spectrum, the PSD value, which might take the form of a displacement PSD, velocity PSD, acceleration PSD or force PSD, is plotted as a function of frequency. According to mathematics, the variance, or square of the standard deviation of the response, is equal to the area under a PSD against a frequency curve.

A random vibration study can be carried out as a single-point or multi-point analysis, just like a response spectrum analysis. In a single-point random vibration analysis, a PSD spectrum can be specified at several model positions. A multi-point random vibration analysis allows for the specification of various PSD spectra at various model positions. The random vibration analysis is carried out to verify that the instantaneous deformation and stress amplitudes are caused by random, unexpected loads. The satellite material's yield stress or a specific deformation value could serve as the limit for the analysis. The analysis's specified stress input data are in the form of qualifying PSD values in the units of g^2/Hz, which match input values for frequency [45].

5.2.2.3. SV analysis

To simulate the low-frequency launch environment, SV testing entails exposing the test object to a sweeping sine input over a specified frequency range (usually 5–100 Hz). On a structural model, this test method is utilised for a variety of reasons, but mostly for flight articles. The SV levels are determined from measured flight data or from coupled load analyses' interface acceleration levels (CLA). For the test, a constant time interval per bandwidth is commonly excited at a logarithmic sweep rate (e.g., 2 or 4 octaves/min), with the goal of simulating sustained sine and transient events that happen during launch. By using best practices, risks can be reduced [46].

5.2.2.4. Harmonic response analysis

Harmonic analysis is used to make sure that the vibration amplitude will not go beyond the predetermined limit (in metres) along the excitation frequencies or vibration modes. With regard to the launch vehicle, two main directions of analysis are conducted. The first is along the longitudinal axis, while the second is along the transverse axis. The launch vehicle manual provides the imposed loading conditions [45]. It provides the steady-state response of a linear structure to harmonically time-varying loads. These loads are imposed on a satellite by a launch vehicle. The harmonic analysis aids in determining the proper torque to apply while fastening screws and other fasteners to the satellite to prevent them from falling loose or breaking during launch [47]. From the launch vehicle manual, the acceleration values in the two directions as well as the lowest and maximum frequency in hertz can be extracted and used as limits for the harmonic analysis.

5.2.2.5. Acoustic loads

Launchers and their payloads are subjected to intense acoustic loads produced by boosters during takeoff operations. The overall sound pressure level (OSPL) achieved typically ranges between 140 dB and 160 dB. Even though it is challenging to predict the fluctuations in the sound pressure level precisely, measurements should be made to better understand the acoustic loads. However, some techniques can help the design team estimate these loads along the vehicle during the early stages of satellite or payload development. This will allow the exact demand for payload protection to be assessed and solutions to lower acoustic energy within the fairing to be taken into consideration. NASA researchers developed a method that can roughly estimate the acoustic loads rockets experience during takeoff using a combination of empirical relationships obtained from experimental data and mathematical models. The use of basic acoustic theory is also an option [48].

5.2.3. Transient and shock loads

A single degree of freedom (SDOF) system's reaction to a shock or any other transient acceleration is known as the shock response spectrum (SRS). A graphical representation is used to represent it. Shock loading also happens as a result of the launch vehicle's acceleration, but to put it into perspective, imagine yourself sitting in a bus that suddenly accelerates, pushing you backwards. You experience only a very small portion of what occurs on the launch vehicle while you are on the bus. The launch vehicle experiences vibration and increased acceleration. The acceleration of the satellite is affected by vibration frequency. The same things that need to be analysed for static analysis also need to be analysed for shock analysis; it must make sure that the maximum stress exerted on the parts is not greater than the material's yield strength considering a safety factor [47].

6. SATELLITE ASSEMBLY, INTEGRATION AND TESTING (AIT)

AIT is a formalised, sequential and documented process for integrating, testing and certifying that the system's requirements and specifications are met. A well-crafted test procedure that is constructed to specify how each requirement is going to be verified is essential for a successful AIT process. The test procedure should reveal unexpected interactions between the subsystems, failure mechanisms and remedies, poor construction and infant mortality (burn-in or wear-in tests). These tests are carried out to simulate the real operational environment and scenarios. The launch and on-orbit environments are modelled in the vibration test and thermal vacuum test (TVT), respectively [49,50]. When realistic test methods are not practical, further testing for payloads, electromagnetic compatibility (EMC), and attitude determination and control subsystem (ADCS) may need to employ simulators or modelling. Discrepancy reports are issued in response to deviations or anomalies to pinpoint the issue and allow a fix [51]. The satellite is then prepared for shipping to the launch site, where it is functionally tested once more to make sure there is no damage [50]. The most popular tests after the satellite assembly are summarised in the following text.

6.1. Mass property measurement

The measurement of the mass properties (MPs) of the entire assembly in its launch configuration will be one of the last steps in the construction of the satellite. MP measurement for a single item involves several steps [52]. Choosing the instrument(s) to be utilised, specifying the coordinate system to be used and its relationship to the satellite coordinate system, comparing the MP tolerances to the item's dimensional tolerances, choosing or designing locating fixtures, and reporting the results are some of these stages. In this measurement, the satellite's mass, CG and moment of inertia (MOI) about its roll axis are measured to validate that these quantities fall within the defined tolerances. If not, some ballast may be added to make small adjustments. Any significant variations that may have occurred during component fabrication or assembly cannot be remedied anymore. Performance is affected when these variations are large [53].

6.2. Environmental testing

The main goals of environmental testing are to provide basic knowledge and indicators that serve as representations of the influence of the environment under test. Typically, it is carried out to simulate the hard launch conditions and the space environment while the satellite is in orbit. The satellite and its components are subjected to a variety of strict tests, which include thermal, vibrational and acoustic testing. This testing simulates the various environmental conditions and mechanical stresses that a satellite could experience during its lifetime [50].

6.2.1. Thermal test

The satellite will be thermally stressed during the thermal tests to (1) reveal latent faults by exhibiting three cycles between hot and cold temperatures without any failures, (2) ensure that the satellite can operate successfully over a wide temperature range and during temperature changes and (3) collect temperature data about the steady-state temperature to evaluate part, board and assembly gradients.

Both the ground station (or electrical ground support equipment, EGSE) and the dynamic simulator are connected to the satellite that is within the thermal chamber during the thermal test. Electronic stimuli required for functional testing are provided by the ground station.

A thermal balance test is performed first. The thermal chamber should be capable of regulating temperatures between 40°C and +55°C at a rate of at least 3°C/min. During the test, chamber temperature gradients should not increase over 4°C/s. Temperatures at chamber and satellite test points should be monitored and recorded by a data acquisition system. The ground station should gather data from the satellite telemetry. When the chamber is within 2°C of the required setpoint, environmental stability is attained.

6.2.2. Vibration test

The satellite is normally qualified for the launch environment using vibration tests. These tests generally consist of sine, sine-burst and random test cycles. The choice of the facility where the test will be conducted must be made beforehand. The capabilities of the vibration table and the number of instrumentation channels available are the two main factors to be taken into account when selecting a test facility. To fasten the satellite to the vibration table, an interface fitting must be made. The interface fitting should have a very high resonance frequency (> 1,000 Hz). The bolt pattern for bolting to the table would be provided by the testing facility. Theoretically, a test may be carried out with no instrumentation at all; as long as the satellite survives the test environment without malfunctioning, the test is a success. Even though there would be a big risk, it would be acceptable, especially during random vibration testing.

The satellite is normally not powered during tests, and it is in the launch configuration without thermal blankets. The satellite is covered by a lightweight bag that enables accelerometer attachment. Lifting equipment that is not necessary for flight configuration is taken out. The stowed configuration of solar panels is installed and locked.

When the vibration test is performed on multiple axes, there will be one that is most crucial. It is possible to learn more about the structure before applying the critical test loads by beginning with the non-crucial axis. For each axis, the first and last test cycles are signature cycles, either sine (5–2,000 Hz) or random (20–2,000 Hz). In each axis, the sine, sine-burst and finally random vibrations are tested. It is simple to compare the sine and sine-burst tests to the predictions from the pretest FEM study. This enables the engineer to confirm that the instrumentation is operating properly. All satellite modes across the whole frequency range are stimulated during the random test. The test results give the first indication of the model's accuracy. Certain modes might couple, resulting in a higher response than expected, while the results of the sine and sine-burst tests are often what is expected.

The launch vehicle acts as a flexible constraint when the satellite is attached to it during launch. When one of the satellite modes takes off, the interface loads with the launch vehicle rise, although some of these loads may be reduced by the flexibility of the satellite. Depending on measured loads (from prior launches) or coupled load analysis, static design loads are determined. These account for this dampening. Therefore, from a strength perspective, the satellite just has to be able to withstand 1.25 times the design loads. The sine-burst test is considered a strength test, while random testing is considered a workmanship test, not a structural or strength test. This is important to take into account when performing the random vibration test. On the other side, the satellite is fastened to a vibration table that offers no damping during vibration testing. Therefore, the response may be substantially greater than what would be observed during flight if one of the satellite modes takes off during testing. This phenomenon allows for the limitation of input levels during random testing.

6.2.3. Bake out test

This optional bakeout test's goal is to outgas satellite components to reduce the possibility of condensable volatile contamination upon their flight. Before bakeout, the satellite undergoes thermal cycling at ambient pressure to reduce the possibility of condensable volatile contamination upon its flight. Before baking, the satellite undergoes thermal cycling at ambient pressure. Integration of the satellite into the thermal vacuum chamber (TVAC) requires confirmation of the chamber's cleanliness. The thermal vacuum bakeout chamber is where the spacecraft and its TVAC ground support equipment (GSE) are installed. During the test, the chamber is kept at +40°C and less than 1×10^{-5} torr of pressure. The data acquisition system continuously monitors satellite temperatures, compares them to alerts and records temperatures in real time until the satellite has reached bakeout.

6.2.4. TVT

Satellites or their components are commonly tested in a TVAC in a space environment simulation. The TVAC has a controlled radiative thermal environment. Typically, the thermal environment is created by applying thermal lamps for high temperatures or by flowing liquids or fluids through thermal shrouds for cold temperatures. Temperatures are recorded real time by the data acquisition system, which continuously checks and compares them with satellite temperature alarms. The purpose of TVAC tests is to validate the thermal design by exposing the satellite and payload to thermal test environments that conservatively simulate the hot and cold conditions of flight, gathering steady-state and transient data to correlate the thermal models, and operating the satellite at temperatures above those anticipated in the orbit.

7. CHALLENGES AND OPPORTUNITIES

Different materials are determined by NASA to be suitable for space missions. The most often used space-grade aluminium alloys are 6061-T6 and 7075-T7. Despite having different qualities, no research has been carried out to compare the two alloys' suitability as materials for satellite primary structures. Furthermore, more investigation is required to determine the ideal structural configuration for these two materials. The isogrid and honeycomb structures are the most recent configurations that achieve great stiffness-to-weight ratios. More research is required to compare and optimise the shape parameters of these two designs. Additionally, it is necessary to optimise the manufacturing process and its variables.

8. CONCLUDING REMARKS

This article attempted to present a general view of satellite concepts in relation to how they are categorised according to their mass and intended usage. A light is also shed on the satellite's main subsystems and their functions. This review's focus is drawn to the mechanical design aspects affecting the performance of the satellite primary structure, such as the materials, geometry and configuration, as well as the mechanical loads and their analyses.

Aluminium alloys are the most popular in the satellite primary structure industry. In addition to having the proper mechanical properties, these alloys have a lot of other benefits, thanks to their being readily available, affordable and simple to manufacture. Aluminium primary structures for satellites can be produced using different configurations that tend to promote a high stiffness-to-weight ratio. The most promising structural configurations are isogrid and honeycomb sandwich structures. High-precision manufacturing processes, such as CNC milling and additive manufacturing, can be used to produce satellite primary structures with different configurations and dimensions.

In conclusion, it can be argued that the primary structure of the satellite experiences a variety of loads over the period of its lifetime, necessitating additional study to promote its material, geometry, configuration and mechanical design analysis.

REFERENCES

[1] Maral, Gerard and Bousquet, Michel. *Satellite Communications Systems*, Chapter 10. 5th edn. Wiley Online Library (2009): pp. 261–285, DOI 10.1002/9781119259411.ch10.

[2] Available at: https://axessnet.com/en/satellite-manufacturing-considerations/ (2022).

[3] Available at: https://www.rapiddirect.com/knowledge-base/how-cnc-machining-is-used-in-aerospace/ (2022).

[4] Blakey-Milner, Byron, Gradl, Paul, Snedden, Glen, Brooks, Michael, Pitot, Jean, Lopez, Elena, Leary, Martin, Berto, Filippo, and du Plessis, Anton. "Metal Additive Manufacturing in Aerospace: A Review." *Materials and Design* Vol. 209 (2021): p. 110008.

[5] Available at: https://etherealmachines.com/everything-you-need-to-know-about-cnc-machining-in-aerospace/ (2022).

[6] Available at: https://www.thomasnet.com/articles/plant-facility-equipment/remote-sensing-satellite/ (2022).

[7] Boudriga, Noureddine. *Security of Mobile Communication*. 1st Edition, Taylor & Francis Group (2009), DOI 10.1201/9780849379420.

[8] History and Timeline of the ISS, Center for the Advancement of Science in Space. Press Release Point NASA, (2021).

[9] Columbia Electronic Encyclopedia 6th Edition. Columbia University Press; 2020. https://resolver.ebscohost.com/Redirect/PRL?EPPackageLocationID=1615.576175.44397200&epcustomerid=s2947694.

[10] Available at: https://news.viasat.com/blog/scn/how-big-is-that-satellite-a-primer-on-satellite-categories, (2022).

[11] Elizabeth Buchen, Nano/Microsatellite Market Assessment. Space Works Enterprises, Inc., (2014), https://digitalcommons.usu.edu/cgi/viewcontent.cgi?article=3018&context=smallsat.

[12] *Small is Beautiful: US Military Explores use of Microsatellites*. Defense Industry Daily UTC by Defense Industry Daily staff (2011), https://www.defenseindustrydaily.com/Small-Is-Beautiful-US-Military-Explores-Use-of-Microsatellites-06720/.

[13] Nguyen, Tien M. *Satellite Systems: Design, Modeling, Simulation and Analysis*. IntechOpen (2021), DOI 10.5772/intechopen.73789.

[14] Maini, Anil K. and Agrawal, Varsha. *Satellite Technology Principles and Applications*, Chapter 4. 3rd edn. Wiley Online Library (2014), DOI 10.1002/9780470057902.

[15] Lu, Yu, Miller, Alice A., Hoffmann, Ruth, and Johnso, Christopher W. "Towards the Automated Verification of Weibull Distributions for System Failure Rates". *Conference Paper*, 2016. DOI 10.1007/978-3-319-45943-1_6.

[16] Stevens, Craig L. "Design, Analysis, Fabrication, and Testing of a Nanosatellite Structure." Masters Theses. 2002.

[17] Available at: http://www.satmagazine.com/story.php?number=1998655929, (2022).

[18] Vilna, José Antonio, Agelet, Fernando Aguado, Estévez, Miguel López, and Muiño, Alberto González. "Flight Results: Reliability and Lifetime of the Polymeric 3D-Printed Antenna Deployment Mechanism Installed on Xatcobeo & Humsat-D." *Acta Astronautica* Vol. 107 (2015): pp. 290–300.

[19] Jean-noel, Giraudbit. Structural Design of Aerospace Structures, SUP' Aero hand-out, E.N.S.A.E Toulouse France, (1983).

[20] Available at: https://www.americaspace.com/2014/12/04/nasas-cube-quest-challenge-aims-to-send-nanosatellites-to-lunar-orbit-and-beyond-on-first-sls-flight/, (2022).

[21] Available at: https://www.eoportal.org/satellite-missions/sapphire-stanford#sapphire-microsatellite, (2022).

[22] Available at: https://phys.org/news/2010-11-hand-built-satellite-students.html, (2022).

[23] Available at: https://www.nanosats.eu/, (2022).

[24] Wertz, James R. and Larson, Wiley J. *Space Mission Analysis and Design*. Microcosm Press and Kluwer Academic, Torrance, CA (1992).

[25] Mangonon, Pat L. *The Principles of Materials Selection for Engineering Design*. Prentice Hall, Upper Saddle River, NJ (1999).

[26] Conkey, Shelley B., Lee, Chiachung, Chaykovsky, Steve, Content, David, Morell, Armando, and Budinoff, Jason. "Optimum Design of a Lightweight Mirror using Aluminum Foam or Honeycomb Sandwich Construction – A Case Study for the GLAS Telescope." *Current Developments in Lens Design and Optical Systems Engineering; Proceedings of the Conference*: pp. 325–332. San Diego, CA, Vol. 4093, 2000.

[27] Sarafin, Thomas P. and Larson, Wiley J. *Spacecraft Structures and Mechanisms – From Concept to Launch*. Microcosm Press and Kluwer Academic Publishers, Torrance, CA (1995).

[28] Ciccarelli, Daniele, Forcellese, Archimede, Greco, Luciano, Mancia, Tommaso, Pieralisi, Massimiliano, Simoncini, Michela, and Vita, Alessio. "Buckling Behavior of 3D Printed Composite Isogrid Structures." *Procedia CIRP* Vol. 99 (2021): pp. 375–380.

[29] Meyer, R.R., Harwood, O.P., Harmon, M.B., and Orlando, J.I. *Isogrid design handbook*, (1973), https://ntrs.nasa.gov/citations/19730000395.

[30] Zheng, Q., Jiang, D., Huang, C., Shang, X., and Ju, S. "Analysis of Failure Loads and Optimal Design of Composite Lattice Cylinder under Axial Compression." *Composite Structures* Vol. 131 (2015): pp. 885–894.

[31] Holmes, M. "Carbon Composites Continue to Find New Markets." *Reinforced Plastics* Vol. 61 (2017): pp. 36–40.

[32] Rahimi, G.H., Zandi, M., and Rasouli, S.F. "Analysis of the Effect of Stiffener Profile on Buckling Strength in Composite Isogrid Stiffened Shell Under Axial Loading". *Aerospace Science and Technology* Vol. 24 (2013): pp. 198–203.

[33] Dawood, Sarmad Daood Salman, Inayatullah, Othman B., and Samin, Razali B. "Computational Study of the Effect of using Open Isogrids on the Natural Frequencies of a Small Satellite Structure." *Acta Astronautica* Vol. 106 (2015): pp. 120–138.

[34] Forcellese, Archimede, di Pompeo, Valerio, Simoncini, Michela, and Vita, Alessio. "Manufacturing of Isogrid Composite Structures by 3D Printing". *Procedia Manufacturing* Vol. 47 (2020): pp. 1096–1100.

[35] Forcellese, Archimede, Marconi, Marco, Simoncini, Michela, and Vita, Alessio. "Environmental and Buckling Performance Analysis of 3D Printed Composite Isogrid Structures. *Procedia CIRP* Vol. 98 (2021): pp. 458–463.

[36] Sivák, Peter, Delyová, Ingrid, and Diabelková, Patrícia. "Analysis of Sandwich Structures by the FEM." *American Journal of Mechanical Engineering* Vol. 5 No. 6 (2017): pp. 243–246. DOI 10.12691/ajme-5-6-2.

[37] Sadiq, Sadiq Emad, Jweeg, Muhsin J., and Bakhy, Sadeq H. "The Effects of Honeycomb Parameters on Transient Response of an Aircraft Sandwich Panel Structure." *IOP Conference Series: Materials Science and Engineering* Vol. 928 (2020): p. 022126.

[38] Harris, E. Nathan, and Morgenthaler, Daniel R. "Design and Testing of Multifunctional Structure Concept for Spacecraft. *The 41st Structures, Structural Dynamics, and Materials Conference and Exhibit*. Atlanta, GA. AIAA/ASME/ASCE/AHS/ASC, 2000.

[39] Available at: https://www.uplandfab.com/blog/challenges-cnc-machininghoneycomb core, (2022).

[40] Abd Kadir, Nurdina, Aminanda, Yulfian, Dawood, Mohd. Sultan Ibrahim Shaik, and Mohktar, Hanan. "Numerical Analysis of Kraft Paper Honeycomb Subjected to Uniform Compression Loading." *IOP Conference Series: Journal of Physics: Conference Series* Vol. 914 (2017): p. 012004.

[41] Li, Z.G., Wang, J.M., Li, M., and Chen, F.Y. "Theoretical Analysis and Numerical Simulation of Square Honeycombs." *Journal of Physics: Conference Series* Vol. 1777 (2021): p. 012042.

[42] Thomas, Tiju, and Tiwari, Gaurav. "Crushing Behavior of Honeycomb Structure: A Review." *International Journal of Crashworthiness* Vol. 24 No. 5 (2019): pp. 555–579.

[43] Abdelal, Gasser F., Abuelfoutouh, Nader, and Gad, Ahmed H. *Finite Element Analysis for Satellite Structures*. Springer-Verlag, London (2013), DOI 10.1007/978-1-4471-4637-7.

[44] Yilmaz, Fatih, Haktanir, Ozan, and Uygur, Ahmet. "Quasi-Static Structural Test of Satellites." *Conference Paper*, 2015, DOI: 10.1109/RAST.2015.7208381.

[45] Raviprasad, Srikanth, and Nayak, Nagaraj. "Dynamic Analysis and Verification of Structurally Optimized Nano-Satellite Systems." *Journal of Aerospace Science and Technology* Vol. 1 (2015): pp. 78–90.

[46] NASA Engineering and Safety Center Technical Bulletin No. 15-03, Best Practices for Use of Sine Vibration Testing.

[47] Available at: https://www.aero.iitb.ac.in/satelliteWiki/index.php/Analysis, (2022).

[48] Souto, Carlos, and Pirk, Rogerio. "Acoustic Loads on a Sounding Rocket Lift-Off".19th Brazilian Congress of Mechanical Engineering, Proceedings of COBEM, 2007.

[49] Pisacane, V. *Fundamentals of Space Systems*. 2nd edn. Oxford University Press, New York (2005).

[50] Sebestyen, George, Fujikawa, Steve, Galassi, Nicholas, and Chuchra, Alex. *Low Earth Orbit Satellite Design*. Microcosm Press and Springe (2018), DOI 10.1007/978-3-319-68315-7.

[51] Baghal, Lisa. "Assembly, Integration, and Test Methods for Operationally Responsive Space Satellites." Master Thesis, Corpus ID: 108725233. 2010.

[52] Boynton, R. and Wiener, K. *Mass Properties Measurement Handbook*, Society of Allied Weight Engineers (SAWE). Space Electronics, Inc., 57th Annual Conference, Wichita, Kansas, (1998).

[53] Wiener, Kurt. "The Role of Mass Properties Measurement in the Space Mission". *Conference Paper*, Society of Allied Weight Engineers, Inc., 2005.