Innovative Satellite Bus Structures Using Thermoplastic Materials

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***Abstract*—This paper investigates the use of thermoplastic composites as a sustainable alternative to aluminium in the structural design of small satellite buses. The objective is to reduce reliance on traditional metallic materials while aligning with circular economy principles through recyclability and streamlined manufacturing. The study focuses on key load-bearing components, specifically external panels and primary structural elements, for which thermoplastics offer advantages in mass reduction, end-of-life recovery, and rapid production via thermoforming and over-moulding. A material trade-off analysis was conducted, assessing candidates based on outgassing behaviour, thermal and chemical resistance, manufacturability, and cost. PEEK reinforced with 30% carbon fibre was selected as the optimal material for the final demonstrator. Three preliminary satellite bus configurations were developed and analysed using finite element methods under static, modal, and random vibration loading to assess structural performance. A scaled PETG mock-up was fabricated to evaluate assembly processes and integration constraints. Based on these findings, a mission-specific design for a VLEO Earth observation platform was developed using off-the-shelf components. The final configuration was further analysed for launcher-specific static loads and dynamic response. Future work will focus on fabricating a full-scale demonstrator using advanced thermoforming techniques. The study highlights the potential of thermoplastics to meet aerospace standards while advancing sustainability in satellite design.**

***Index Terms*—Thermoplastics, Satellite bus, Finite element analysis, VLEO, PEEK.**

I. Introduction

The objective of this research is to develop a critical design for a next-generation satellite bus, as part of the New Space Portugal (NSP) project at INEGI. The focus is on exploring the potential of thermoplastic technologies to enable the production of a modular and cost-effective satellite bus. In the preliminary design review of this project, thermoplastic-based manufacturing technologies, particularly thermoforming and over-moulding (available at INEGI), were selected for further exploration. The design process includes the selection of suitable thermoplastic materials to replace conventional aluminium structures, the development of initial structural concepts, and the refinement of a final design compatible with the chosen manufacturing methods.

Although the project is not tied to a specific mission or spacecraft, it is framed as a feasibility study for a future small satellite operating in Very Low Earth Orbit (VLEO). Environmental factors relevant to VLEO—such as atomic oxygen exposure, hypervelocity impacts, and thermal cycling—are key drivers in the material selection and design decisions. This investigation is limited by the following requirements:

A. Mission Requirements

- Mass: The total satellite bus structure mass should be less than 2 kg.
- Material: The structure must primarily use thermoplastic composites to showcase INEGI's capabilities.
- Cost: Minimize costs by reducing the number of mould geometries required for thermoforming.
- Durability: The structure must withstand transport, launch, and environmental loads during the mission.
- Geometry: Design must be optimized for atmospheric drag, with flat surfaces for solar panels and interface compatibility with the launch vehicle.
- Modularity: The bus should support easy geometric modifications.
- Optical System: The platform must accommodate Earth observation subsystems.

B. Technological Requirements (based on INEGI's manufacturing capabilities)

- Maximum part dimension: 500 mm
- Minimum part dimension: 50 mm
- Maximum part thickness: 10 mm
- Minimum part thickness: 1 mm
- Bending radii: Dependent on panel thickness
- Minimum bending angle (V-shapes): 15°
- Minimum edge bending angle (flat panels): 90°

C. General Manufacturing Requirements:

- Rapid prototyping and production are essential.
- Manufacturing processes should be simple and repeatable.
- Moulds and final products should remain cost-effective.
- Weld zones must be accessible from the exterior.
- Overlapping weld zones should be avoided.
- Continuous welds are preferred where possible.
- The structure must be scalable.

This work is part of the New Space Portugal (NSP) project, supported by INEGI.

- The design should minimise the number of required moulds.
- Assembly and disassembly should be straightforward.

II. Structure Design

Current practices in small satellite design typically rely on conventional metallic materials to minimize technical risk and cost [1], [2]. However, with increasing emphasis on sustainability, modularity, and rapid development cycles, alternative approaches are being explored, specifically the development of a satellite bus structure based entirely on thermoplastic composites [3]. This effort leverages INEGI's in-house capabilities in thermoforming and over-moulding to demonstrate a modular, recyclable, and cost-effective platform.

A. Material Selection

A trade-off analysis using the VIKOR multi-criteria decision-making method was conducted to identify the most suitable thermoplastic material for the satellite bus structure. Six candidate materials (PEEK, PSU, PPS, PESU, PC, and PEI) were initially evaluated based on key parameters, including maximum and minimum service temperatures, glass transition temperature, Total Mass Loss (TML), and Collected Volatile Condensable Materials (CVCM). Following a preliminary screening, the selection was narrowed to four materials: PEEK, PEI, PESU, and PPS. A weighted decision matrix incorporating 15 material properties (outlined in Table 1) was used to perform the VIKOR analysis. Among the evaluated options, PEEK emerged as the top-performing material due to its superior thermal stability, mechanical properties, and outgassing performance, with PPS ranking second. Although PEEK is associated with higher material costs, its performance benefits can be justified for mission-specific applications. PPS remains a viable, cost-effective alternative with broad processing compatibility. To streamline fabrication and simplify design, it was deemed advantageous to manufacture the structure from a single fibre-reinforced thermoplastic. Given the inherently low thermal and electrical conductivity of polymers, the incorporation of carbon fibres is essential to enhance functional performance. Based on this rationale and further evaluation, PEEK reinforced with 30% carbon fibre (VICTREX AE 250 UDT) was selected for the final demonstrator.

B. Preliminary Designs

The core design philosophy guiding the preliminary development was to create a versatile and cost-effective satellite bus structure using thermoplastic composites. The aim was to optimise both structural performance and manufacturability using INEGI's thermoforming and over-moulding capabilities. Figure 1 illustrates the conceptual design featuring an octagonal cross-section, selected for its balance between structural efficiency and ease of panel integration. Taking into account manufacturing constraints and assembly considerations, particularly related to the thermoforming of large flat surfaces and the integration of over-moulded joints, three distinct design

TABLE I: Properties used in VIKOR method decision making matrix versus their suggested weights.

Property	Suggested Description	Weight
Density	Critical (affects mass and launch cost)	15%
Tensile Modulus	Very important (affects stiffness and structural integrity)	12%
Compression Modulus	Important (affects compressive strength)	8%
Tensile Strength	Very important (resists mechanical stresses)	10%
Thermal Expansion	Very important (minimizes thermal stresses)	10%
Glass Temperature	Important (maintains material stability at high temperatures)	7%
Max Operating Temp.	Critical (ensures material operates well in space)	12%
Fracture Toughness	Important (resists cracking or breaking under stress)	8%
TML	Important (minimizes material loss in vacuum)	6%
CVCM	Very important (reduces risk of outgassing)	6%
Price	Important (budget constraint)	6%

concepts for the side panels were developed (Figure 2). These configurations explore various strategies for geometry simplification, modularity, and manufacturing feasibility within the scope of the selected thermoplastic material [4].

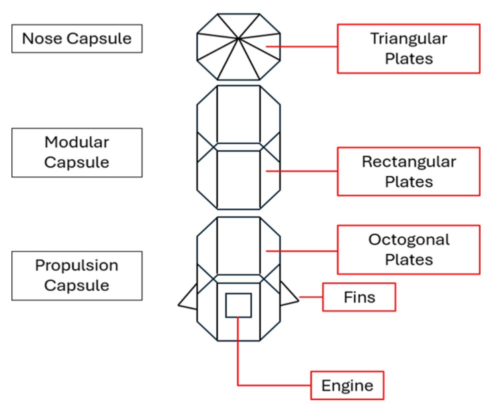


Fig. 1: The conceptual modular design

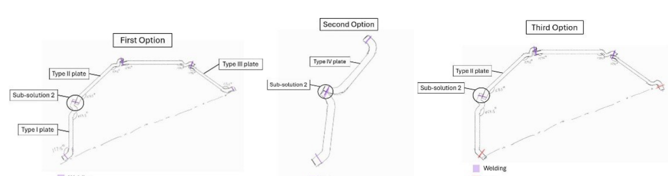
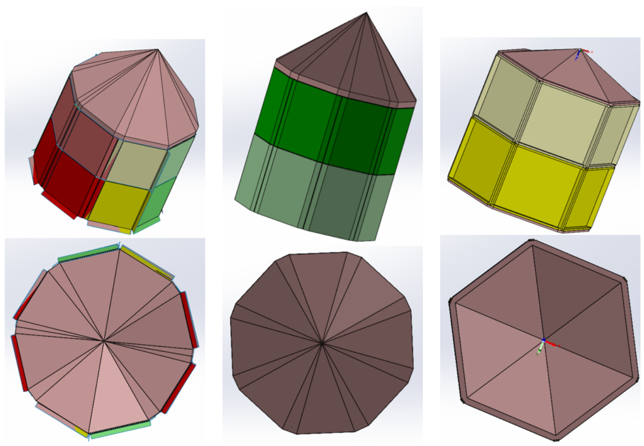


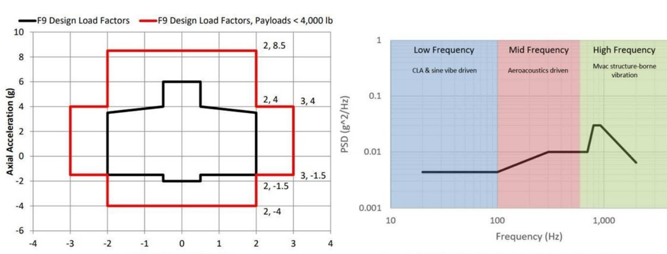
Fig. 2: Three preliminary designs for side panels with welding or bolt connections.

Based on the proposed panel configurations, three structural models of the satellite bus were developed (Figure 3) and evaluated through finite element analysis. Each model was assessed under static loading conditions, free vibration (modal analysis), and random vibration, simulating the mechanical environment of launch (Figure 4). At the stage of preliminary design, environmental loadings of Falcon 9 were used. For modal analysis, specific criteria were defined to ensure dynamic compatibility with typical launch vehicles: the primary lateral natural frequency was required to exceed 10 Hz, the primary axial frequency above 25 Hz, and all secondary structure resonances above 35 Hz to avoid dynamic coupling with the launcher. To meet these requirements, the panel thicknesses were iteratively optimised to ensure compliance with predefined failure criteria, including minimum allowable natural frequencies and stress limits.



(a) (b) (c)

Fig. 3: Three preliminary models for satellite bus structure



(a) (b)

Fig. 4: G-loads acting on a payload and random vibrations at a Falcon launch.

The performance and design efficiency of the three structural models are summarised in Table 2, which compares key parameters including total mass, number of moulds required for fabrication, and total external surface area—an important factor for solar array integration. This comparative analysis facilitates a balanced trade-off between structural performance, manufacturability, and functional system requirements. Additionally, the table provides the dimensions of the internal cylindrical section enclosed within each model, offering a consistent reference for estimating available internal volume—particularly relevant for the accommodation of propulsion units and modular payload components. It is also noteworthy that, if Model 2 were constructed using a 1.5 mm thick aluminium shell instead of the selected thermoplastic composite, the total mass would increase to approximately 1.96 kg, representing a 33% increase compared to the PEEK-carbon fibre configuration. This highlights the significant mass-saving advantage of the PEEK material used while maintaining structural viability.

TABLE II: Comparison of the preliminary models

Properties	**Model (a)**	**Model (b)**	**Model (c)**
Cylinder Dimension (mm)	D: 306 H: 246	D: 295 H: 241	D: 204 H: 398
Mass (kg)	2.07	1.47	1.79
# of moulds required	5	3	3
Area (m^2)	0.505	0.485	0.440

Among the three evaluated concepts, Model 2 demonstrated the highest fundamental natural frequency while maintaining the lowest overall mass, making it the most promising candidate for further development. As a result, Model 2 was selected for the critical design phase.

C. Mock-up

To make the proposed design functional, it was necessary to define an assembly sequence for the structure. To validate the design and its assembly, a small-scale mock-up (Figure 5) was fabricated using PETG material. This prototype allowed researchers to test key assembly and disassembly processes, providing valuable insights into the practicality of using such designs in satellite manufacturing. The mock-up also highlighted the potential for rapid prototyping and iteration, a significant advantage over traditional manufacturing methods. The proposed assembly steps are as follows:

Fig. 5: PETG mock-up produced through 3D printing

1) Join the lateral plates into two sub-assemblies: This step involves welding four lateral plates together to form a sub-assembly.
2) Mount the modular lateral plates onto the sub-assemblies: If the modular capsule is utilized, the lateral plates from the modular capsule must be connected to the sub-assemblies of the propulsion capsule (assembled in the previous step). This step will extend the length

of the sub-assemblies, but they will remain separate. Welding is employed in this step.
3) Connect the nose to one of the sub-assemblies: At the top of the sub-assembly, connect the nose using welding.
4) Join the two sub-assemblies: The two parts are connected together according to the chosen assembly concept.
5) Connect the bottom: This involves attaching the bottom to the assembled structure using the appropriate joining method.

D. Satellite Components

In this stage, additional mission-specific components — sourced from commercially available off-the-shelf (COTS) systems — were integrated into the structural model based on the defined requirements of a representative Earth observation mission in VLEO. Following this integration, additional simulations were performed to re-evaluate the structure under static loading condition and free vibration, ensuring continued compliance with launch and operational criteria. To ensure the satellite met its mission's demanding performance standards, every component was chosen through a rigorous evaluation process that balanced strict mass limitations with essential functionality. Factors such as weight, energy consumption, durability, and each component's role in critical operations, including attitude stabilization, power management, thermal regulation, and communication were carefully considered. Lightweight, high-performance materials and multi-functional designs were prioritized [4] so that the satellite not only minimized launch costs but also maintained reliability over extended mission lifetimes. Every part — from the structural bus and power system to the onboard computer and propulsion components — had to justify its inclusion by contributing indispensably to the mission objectives. Table III presents the components considered in the detailed critical model as well as their representative dimensions and masses.

TABLE III: Satellite component specifications

Component	Dimensions (mm)	Mass (g)
HR Camera Module	97.2 × 55 × 55	277
OBC	90 × 95 × 30	34
Battery System	85 × 92 × 40	500
Power Unit	93 × 93 × 8.75	250
Comm. Transceiver	83 × 57 × 16	125
Magnetometer	33 × 20 × 11.3	12
Sun Sensor	22 × 15 × 5.3	4
Magnetorquer (×3)	10.5 × 10.5 × 92	28
Reaction Wheels (×3)	43.5 × 43.5 × 24	137
GPS Antenna	50 × 50 × 12.5	29
Startracker	55 × 65 × 70	475
GPS Receiver	9.7 × 10.1 × 4	20
Propulsion System	140 × 120 × 98.6	3900
Interface Board	100 × 70 × 15	100

E. Critical Design

To accommodate all required components, the preliminary design underwent a major modification, with the addition of a floor to house the propulsion system. The critical design, shown with a simplified representation of the internal components and with certain plates hidden for clarity, is presented in Figure 6. The final dimensions of the satellite bus are a height of 414 mm and a base with an inner radius of 97 mm for the octagonal cross-section.

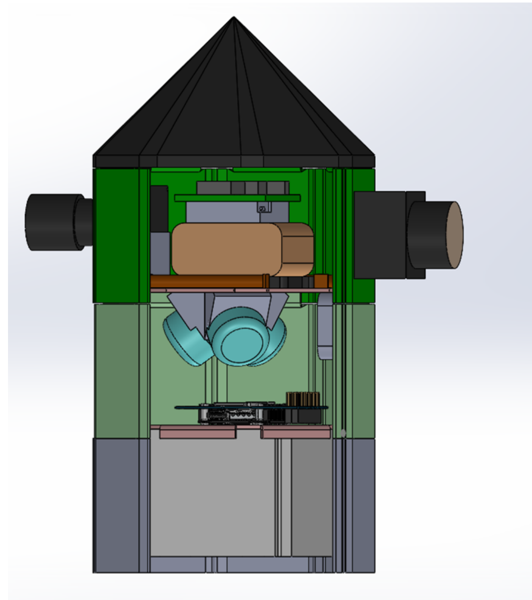


Fig. 6: Critical Design CAD

III. Finite Element Analysis

A total of five simulations were conducted to validate the satellite bus model using ANSYS software [5], in accordance with the launch load conditions specified in the Ariane 6 User's Manual for Multi-Launch Services [6]. The main reason for changing the environmental load conditions from Falcon to Ariane was the latter's tighter vibration thresholds and more demanding launch environment. Four of these simulations addressed different combinations of acceleration loads, while the fifth focused on identifying the fundamental frequencies of the structure.

Since the structural integrity of the bus is the primary focus of this study, internal components were simplified and represented by point masses. This approach preserves their influence on load distribution while significantly reducing the computational cost of the simulations.

The first four simulations performed static analyses under quasi static load (QSL) conditions representative of the launch environment, during which the spacecraft experiences both static and dynamic forces. As outlined in the Ariane 6 manual [6], spacecraft that meet the required frequency thresholds must be designed to withstand specific QSL values. In these analyses, the satellite was fixed at its base to simulate the launch phase, during which acceleration-induced loads are applied. For a satellite of this size and configuration, the structure is required to withstand accelerations up to 10 g.

The fifth simulation aimed to determine the fundamental natural frequencies of the satellite bus to ensure compliance with dynamic decoupling criteria and launcher requirements.

IV. Results and Discussion

A. Static Analysis

The coordinate system shown in Figure 7, as defined in the Launcher Manual [6], is used.

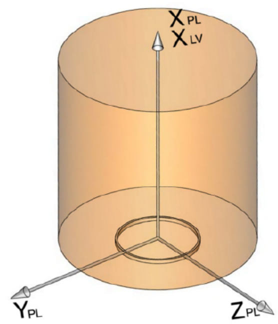
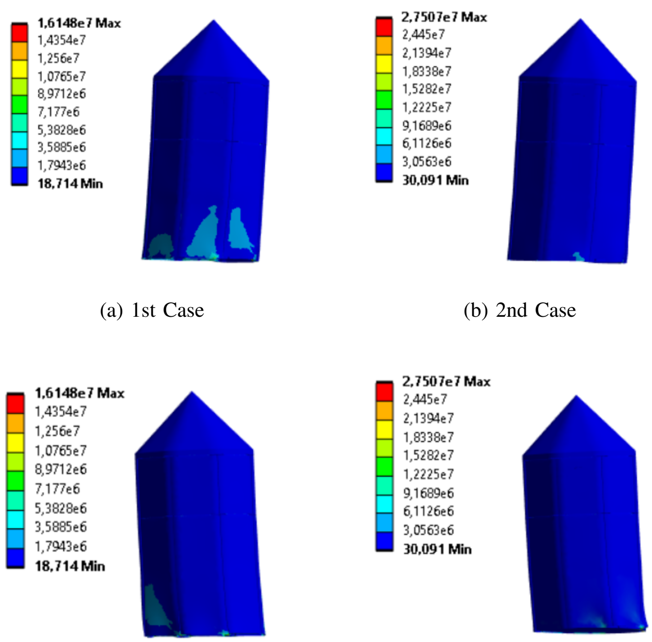


Fig. 7: Payload axis for a forward mounted payload [6]

The static load cases applied to the satellite, along with the resulting maximum von Mises stress and displacement values, are summarized in Table IV.



(a) 1st Case (b) 2nd Case

(c) 3rd Case (d) 4th Case

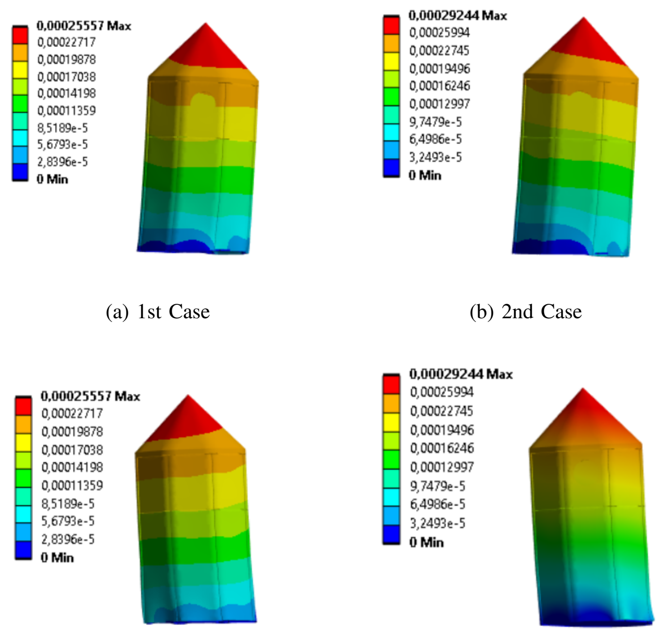
Fig. 8: Maximum stress in static analysis in the satellite structure for all the load cases

The maximum von Mises stress in the structure under all four loading cases (represented in Figure 8) remains well below PEEK's tensile and compressive yield strengths—approximately 100 MPa and 138 MPa, respectively—by a margin of nearly five times [7], [8]. It is worth noting that the final model, to be manufactured at INEGI, will utilize PEEK reinforced with 30% carbon fiber, offering significantly higher mechanical performance.

The displacements in the satellite under various loadings are presented in Figure 9. With a maximum displacement of approximately 0.3 mm relative to the satellite's largest dimension (414 mm), the design demonstrates sufficient safety.

TABLE IV: Static load cases: acceleration, maximum stress, and displacement

Case	Acceleration	Max Stress [MPa]	Max Displacement [mm]
1	X: -10 g, Y: 10 g, Z: 10 g	16.00	0.255
2	X: 10 g, Y: 10 g, Z: 10 g	27.50	0.290
3	X: 10 g, Y: -10 g, Z: -10 g	16.14	0.256
4	X: -10 g, Y: -10 g, Z: -10 g	27.50	0.290



(a) 1st Case (b) 2nd Case

(c) 3rd Case (d) 4th Case

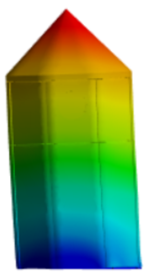
Fig. 9: Maximum displacement in static analysis in the satellite structure for all the load cases

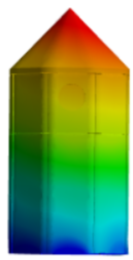
B. Modal Analysis

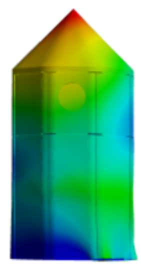
The first 10 vibration modes of the satellite structure were simulated, with the corresponding natural frequencies presented in Table V. According to the Launcher Manual [6], for a satellite of the dimensions considered in this project, the fundamental frequency should exceed 115 Hz, which is consistent with the values obtained here. The deformations in the satellite structure for the first 6 modes of vibration are presented in Figure 10. It seems that the first bending modes in different directions of the satellite are lower than the first overall axial mode. Then it is followed by the 1st torsional mode, 1st bending at the 3rd deck from the bottom, and later breathing modes in the conical section of the satellite.

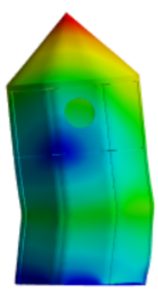
TABLE V: First 10 natural frequencies of the satellite structure

Mode	**Frequency [Hz]**
1st bending (z-direction)	146.46
1st bending (y-direction)	151.26
2nd bending (y-direction)	335.20
2nd bending (z-direction)	365.18
1st overall axial mode	424.37
1st torsion	464.33
1st bending (at 3rd deck from bottom)	778.24
1st bending (at the conical section)	918.67
1st breathing (at the conical section)	981.30
2nd breathing (at the conical section)	1036.70


(a) 1st mode


(b) 2nd mode


(c) 3rd mode


(d) 4th mode


(e) 5th mode

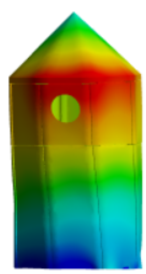

(f) 6th mode

Fig. 10: Deformations in the satellite structure in each vibration mode

V. Conclusion

This research proposed a satellite design utilizing thermoplastic PEEK material, selected through decision-making matrices considering material behavior in space, market factors, technology readiness levels, and manufacturing processes. The material choice supports thermoforming and over-moulding techniques, enabling low-cost, modular design and fabrication of small satellites. The design process included an initial concept followed by a critical design phase, incorporating COTS components for an Earth observation mission in VLEO. The critical design was simulated to ensure compatibility with European launch vehicle loading envelopes. Future work will involve manufacturing a demonstrator at INEGI to advance technological development, focusing on sustainability and circular economy principles.

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