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# NUMERICAL CALCULATION OF THE STRESS-STRAIN STATE OF A COMPOSITE SHELL AIRCRAFT NACELLE STRUCTURE

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***Abstract.*** *This paper presents a numerical analysis of the stress-strain state of a composite shell structure of an aircraft engine nacelle using the finite element method (FEM). The central part of the nacelle, modeled as a cylindrical sandwich shell with carbon fiber reinforced polymer (CFRP) laminates as face sheets and a Nomex honeycomb core, is investigated under internal pressure to evaluate its static strength. Particular attention is paid to the mechanical behavior of orthotropic layers and the application of the Tsai-Wu failure criterion for determining the ultimate states. The analysis showed that the maximum values of the failure index (FI) for all layers remain below 1, while the margin of safety (MS) is positive, which confirms the structural integrity and the presence of a sufficient strength reserve under ultimate loading conditions.*
***Key words:*** *Stress-strain state, finite element method, aircraft nacelle, sandwich composite, numerical modeling, Tsai-Wu criterion.*

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## 1. INTRODUCTION

The increasing use of composite materials in modern aerospace design is driven by their high specific strength and stiffness. In engine nacelle structures, particularly in shell elements, these materials are essential for reducing weight while maintaining reliability. Composite sandwich panels with carbon fiber reinforced polymer (CFRP) face sheets and honeycomb core are widely applied in nacelle design [8]. Considering the complex loading conditions during operation, including static and aerodynamic loads, accurate stress analysis is critically important to ensure the safety and integrity of the structure.

Among the various analytical approaches, the finite element method (FEM) has become the most widely adopted tool for evaluating the behavior of thin-walled composite structures [7, 10, 12, 15]. FEM makes it possible to consider the orthotropic properties of the material, the stacking sequence of laminates, and the interaction between the face sheets and the core, which is essential for sandwich-type constructions. Furthermore, the application of advanced failure criteria, such as the Tsai-Wu criterion, allows not only to predict the stress-strain state but also to assess the reserve of structural strength under realistic operating conditions [6, 9, 11, 12]. This combination of numerical modeling and progressive failure analysis forms the basis for reliable evaluation of the load-bearing capacity of modern nacelle components.

## 2. OBJECTIVES

The main purpose of this study is to analyze the stress-strain state and to evaluate the static strength of the composite shell structure of an aircraft nacelle using the finite element method (FEM). FEM was selected as the principal research tool due to its ability to accurately capture the orthotropic behavior of composite laminates, the geometry of the nacelle shell, and the interaction between face sheets and the honeycomb core. Achieving this goal is important for enhancing the structural efficiency of nacelle components, which directly contributes to weight reduction and operational safety in modern aircraft. To achieve this purpose, the following objectives were formulated:

- to develop a finite element model of the nacelle sandwich shell with CFRP/Nomex configuration;
- to evaluate the distribution of stresses and displacements under the action of internal pressure;
- to verify the static strength of the structure using the Tsai-Wu failure criterion and determine the margin of safety;
- to emphasize the influence of the composite material properties and laminate stacking sequence on the structural efficiency.

## 3. METHODOLOGY

The numerical analysis of the composite nacelle shell was carried out using the Patran/Nastran software package. In Patran, the geometric model was created, the composite layup structure and material properties were defined, while Nastran was employed as the solver. This approach made it possible to perform a realistic assessment of the shell behavior, considering the theory of composite laminates within the finite element method framework [14].

### 3.1. Geometry

The geometry of the nacelle section (Fan Cowl) was modeled as a cylindrical shell composed of three-layer sandwich panels. The model is based on the geometry of the fan cowl structure of the Boeing 777 aircraft. According to industry-standard specifications and manufacturer references, the fan cowl section is approximately 2000 mm in length and 4000 mm in diameter. The composite layup configuration, including the [0/45/0] stacking sequence for CFRP laminates, was chosen based on general industry practices and documented use in similar aerospace sandwich structures. This approach ensures that the numerical model corresponds to real aerospace structures (Fig. 1). The outer and inner face sheets are made of CFRP laminates with the [0/45/0] stacking sequence, while the core is represented by Nomex honeycomb [2, 3].

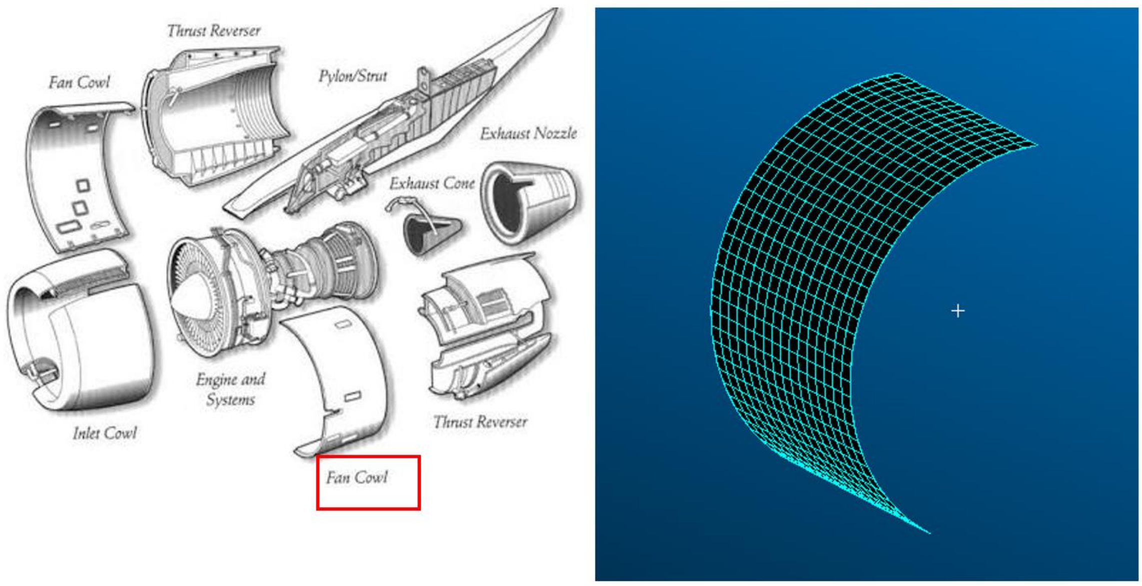


**Figure 1.** Fan Cowl Geometry

### 3.2. Material properties

Based on the previously defined sandwich structure, material properties for the composite layers were specified by creating orthotropic material cards in the Patran environment. The values of elastic modules, shear modules, and Poisson's ratios for the CFRP face sheets and the Nomex honeycomb core were selected from published aerospace composite databases and previous simulation studies [2, 3]. The CFRP laminate stacking sequence [0/45/0] was implemented through ply orientation definitions associated with the shell element properties, with thicknesses corresponding to the specifications of the Boeing 777 nacelle.

**Table 1**

Mechanical properties

| Property | CFRP | Nomex Honeycomb core |
|---|---|---|
| Longitudinal modulus, E1 (MPa) | 135000 | 130 |
| Transverse modulus, E2 (MPa) | 8000 | 130 |
| In-plane shear modulus, G12 (MPa) | 5000 | 50 |
| Out-of-plane shear modulus, G13 = G23 (MPa) | 5000 | 50 |
| Poisson's ratio, ν12 | 0.3 | 0.25 |
| Thickness per ply (mm) | 0.25 | 10 |

The Nomex core was modeled as a homogenized orthotropic material, defined as the central layer of the three-layer sandwich configuration. All material data were directly assigned in Patran as composite properties and verified prior to meshing. The mechanical properties used in the model are summarized in Table 1. These values are consistent with standard data for CFRP and Nomex honeycomb used in the aerospace industry [2, 3].

### 3.3. Verification and Model Validation

From the perspective of the finite element method (FEM), shell elements are ideal for structures in which one dimension (thickness) is significantly smaller than the others, and the deformation occurs primarily in-plane and through bending. The use of quadrilateral elements reduces numerical artifacts and improves the convergence of results, particularly when modeling orthotropic layered materials such as composite laminates [1]. This makes them the preferred choice in aerospace FEM analysis, provided that geometry allows it.

From a theoretical standpoint, quadrilateral shell elements – typically formulated based on Kirchhoff-Love or Reissner-Mindlin plate theories [5] – provide a more accurate interpolation of deformations compared to triangular elements. The stiffness matrix of a shell element is expressed as:

$$K = \int \Omega(B^T D\, B) d\Omega \tag{1}$$

where B is the strain-displacement matrix and D is the material stiffness matrix, which, for orthotropic composite laminates, incorporates layer-wise anisotropy. For quadrilateral elements, the interpolation functions in B allow for better representation of bending and membrane behavior across each element, especially when the mesh quality is high (i.e., aspect ratios near 1:1).

Moreover, numerical studies demonstrate that quadrilateral elements reduce numerical locking and improve solution convergence in thin-walled shell problems. This makes them particularly suitable for modeling nacelle structures manufactured from thin-walled composite laminates.

### 3.4. Strength Verification Using the Tsai-Wu Criterion

#### 3.4.1. Formulation at the ply level (lamina)

The verification is carried out at the level of each face-sheet ply in its local axes 1–2 (1 – along the fibers, 2 – transverse to the fibers). The required input data for each ply includes:

- Elastic constants $E_1$ $E_2$ $G_{12}$ $v_{12}$ (for UD/tape, identical for all plies, differing only by the orientation angle Θ).
- Ply thickness $t_k$ and orientation angle $\Theta_k$.
- Strength allowables $X_t$ $X_c$ (tension/compression along 1), $Y_t$ $Y_c$ (tension/compression along 2), and S (in-plane shear 1–2).

#### 3.4.2. Computation of ply stresses

The global membrane forces $\{N_x\ N_y\ N_{xy}\}$ and moments $\{M_x\ M_y\ M_{xy}\}$ are related to the mid-plane strains $\{\varepsilon_0\}$ $\{k\}$ through the laminate stiffness matrices:

$$\begin{bmatrix} \{N\} \\ \{M\} \end{bmatrix} = \begin{bmatrix} A & B \\ B & D \end{bmatrix} \begin{bmatrix} \{\varepsilon^0\} \\ \{k\} \end{bmatrix} \tag{2}$$

The strain state of a ply located at a distance $z_k$ from the mid-plane is expressed as:

$$\{\varepsilon\}_{global}^{(k)} = \{\varepsilon^0\} + z_k\{k\} \tag{3}$$

The transformation into local ply axes is performed as:

$$\{\varepsilon\}_{12}^{(k)} = T_\varepsilon(-\theta_k)\{\varepsilon\}_{global}^{(k)} \tag{4}$$

The local ply stresses are given by:

$$\{\sigma\}_{12}^{(k)} = Q\{\varepsilon\}_{12}^{k} \tag{5}$$

$$Q = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix} \tag{6}$$

where

$$\upsilon_{21} = \upsilon_{12}\frac{E_2}{E_1}, \quad \Delta = 1 - \nu_{12}\nu_{21}, \quad Q_{11} = \frac{E_1}{\Delta},\ Q_{22} = \frac{E_2}{\Delta}, \qquad Q_{12} = \frac{\nu_{12}E_2}{\Delta}, \quad Q_{66} = G_{12}$$

#### 3.4.3. Tsai-Wu failure criterion for an orthotropic lamina

The failure index (FI) in the local axes 1-2 of the ply is expressed as:

$$FI = F_1\sigma_1 + F_2\sigma_2 + F_{11}\sigma_1^2 + F_{22}\sigma_2^2 + F_{66}\tau_{12}^2 + 2F_{12}\sigma_1\sigma_2 \tag{7}$$

with the failure condition defined as FI < 1. The coefficients are determined from the material allowables as follows:

$$F_{11} = \frac{1}{X_t X_c},\ F_{22} = \frac{1}{Y_t Y_c},\ F_{66} = \frac{1}{S^2} \tag{8}$$

$$F_1 = \frac{1}{X_t} + \frac{1}{X_c}, \qquad F_2 = \frac{1}{Y_t} + \frac{1}{Y_c} \tag{9}$$

Here, $\sigma_1$, $\sigma_2$ and $\tau_{12}$ are the local ply stresses in the 1–2 axes.

The relevance of applying the Tsai-Wu criterion to composite structural analysis is confirmed by modern research [6, 11, 12, 13] in which an orthotropic damage model based on a three-dimensional formulation of the criterion was proposed [6, 11].

### 3.5. Boundary Conditions and Applied Load

The boundary conditions were defined to simulate the realistic support of the nacelle section. The upper edge was supported at four discrete hinge-type points corresponding to the actual mounting system. At each of these points, all translational degrees of freedom ($U_X$, $U_Y$, $U_Z$) and two rotational degrees of freedom (about the X and Y axes) were constrained, leaving only the rotation about the Z-axis ($R_Z$) free, in accordance with hinge support behavior (Fig. 2).

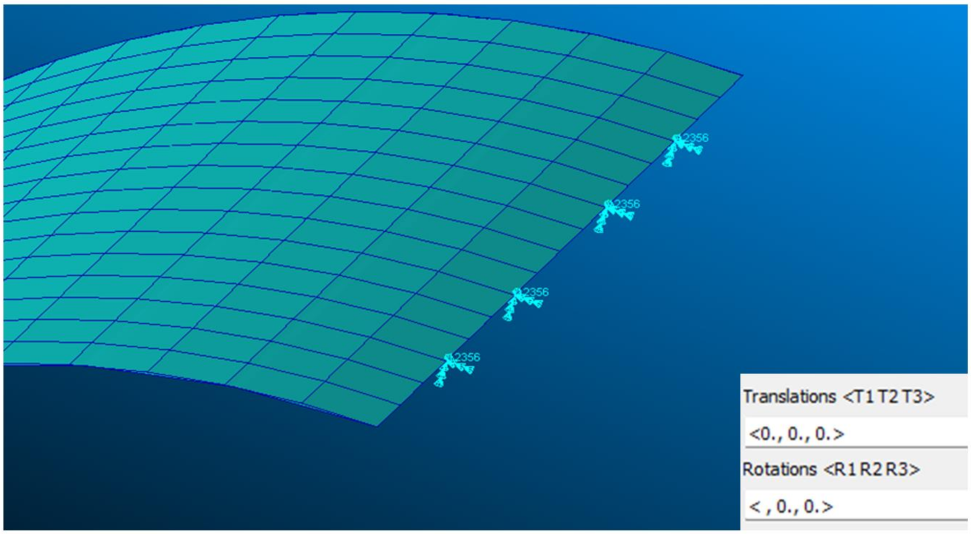


**Figure 2.** Boundary conditions for the upper edge of the Fan Cowl

The lower edge was modeled as fully fixed, with all translational and rotational degrees of freedom constrained. This corresponds to the rigid fastening provided by latches, which are used in the actual operation of nacelles (Fig. 3).

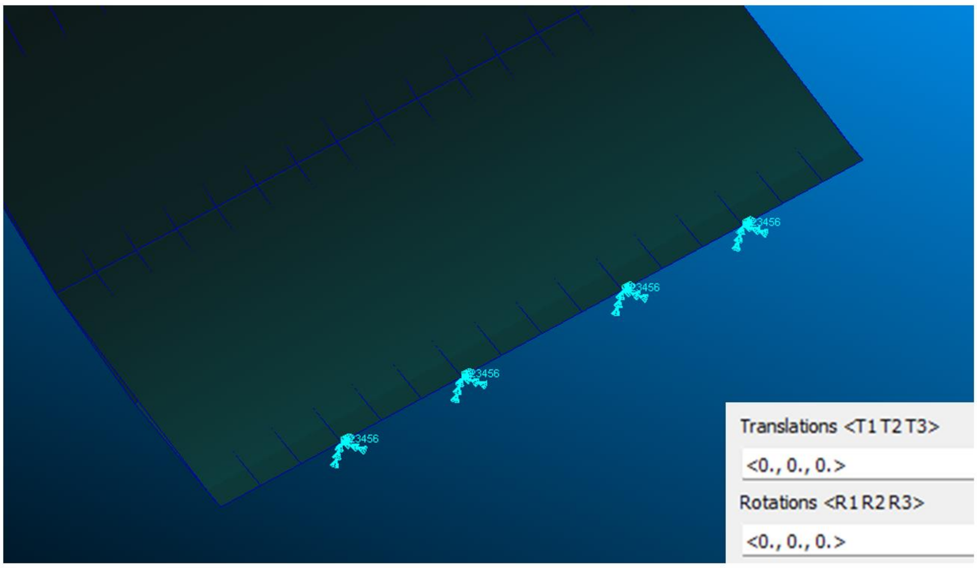


**Figure 3.** Boundary conditions for the lower edge of the Fan Cowl

The internal pressure was uniformly applied to the inner surface of the shell, simulating the aerodynamic loading. According to typical loading scenarios for fan cowls of wide-body aircraft such as the Boeing 777, the internal pressure was set to 0.035MPa, representing the pressure differential during takeoff and climb [4].

This value is consistent with the regulatory load levels for nacelle panels provided in FAA advisory materials and Boeing structural design references [4, 5].

Prior to exporting the model to the Nastran solver for stress-strain analysis, the model was checked for mesh quality, continuity of connections, and correctness of boundary conditions. Visualization of the model is presented in Fig. 4.

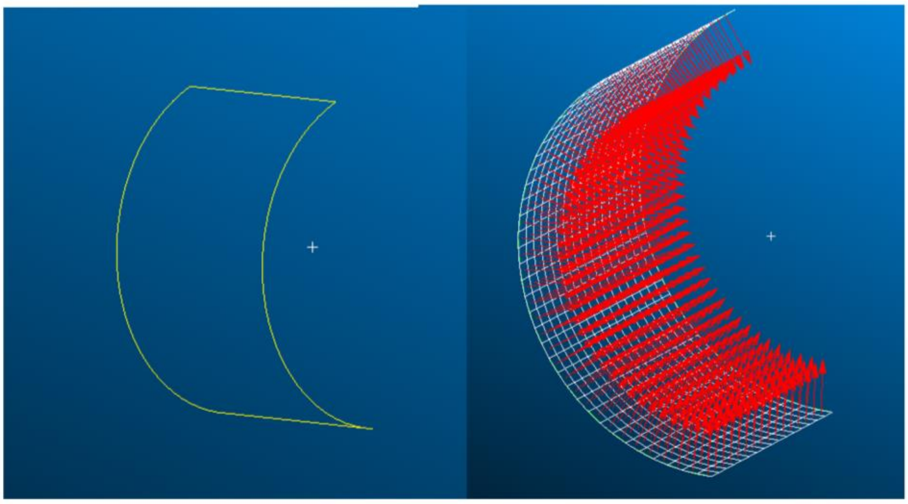

**Figure 4.** Verification of element normals and boundaries

## 4. RESULTS

As a result of the analysis in the Patran/Nastran environment, the failure indices (FI) were obtained for the plies of the composite sandwich shell according to the Tsai-Wu criterion.

- The FI distribution maps (Fig. 5) show that the maximum values are concentrated in the plies with a [45] fiber orientation, which are the most sensitive to combined stress states [15].
- All calculated values satisfy the condition FI < 1, which indicates the absence of failure under the applied load (internal pressure of 0.035MPa).
- The computed margins of safety (MS) for all monitored elements (Fig. 6) are positive. The minimum value of MS=+0.25 was observed in the [45] oriented plies; however, it remains above zero, ensuring a sufficient strength reserve.

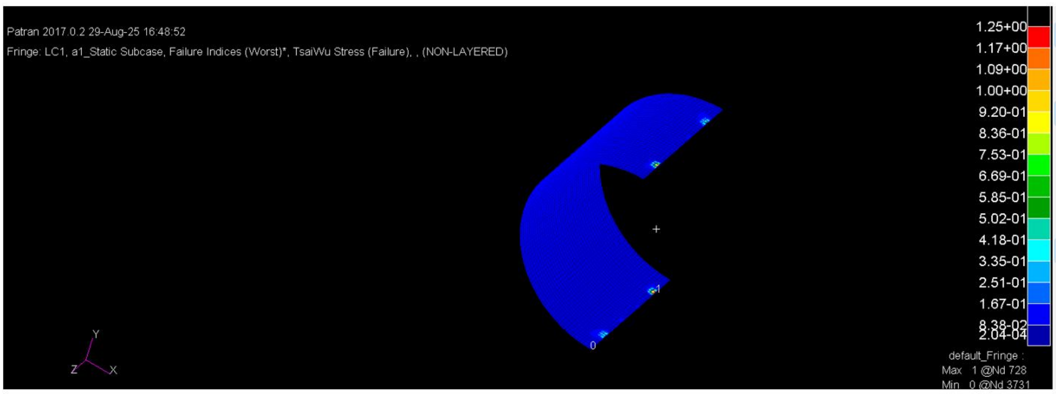


**Figure 5.** Failure indices for the analyzed panel

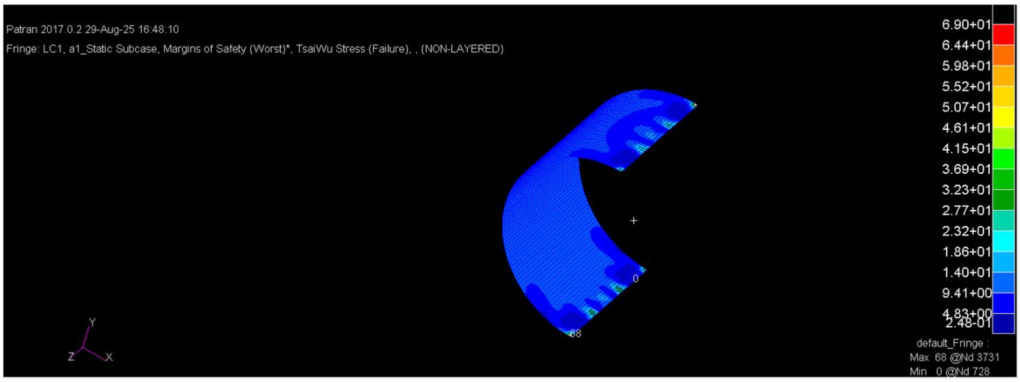


**Figure 6.** Margin of safety for the analyzed panel

Thus, the analysis demonstrated that the nacelle sandwich shell with [0/45/0] layup provides the required strength reserve according to the Tsai-Wu criterion under operational loading conditions.

## 5. CONCLUSIONS

1. The developed finite element model of the nacelle sandwich shell with CFRP/Nomex composite material accurately represents its stress-strain state under internal pressure. The use of quadrilateral shell elements allowed for both convergence and accuracy of the calculations.
2. The stress distribution in the face sheet plies confirmed that the highest loads are carried by the [0] and [45] oriented plies, which corresponds to the expected behavior of an orthotropic laminate under combined membrane and bending loads.
3. The static strength verification using the Tsai-Wu criterion showed that for all plies the failure index (FI) values remain below unity, while the margin of safety (MS) is positive. This indicates the presence of a strength reserve and the structural integrity of the nacelle under the operational internal pressure of 0.035MPa.
4. The most loaded plies were those oriented at [45], however, even for them, the minimum margin of safety remained above zero, ensuring safe operation of the shell.
5. The obtained results confirm the feasibility of using the selected stacking sequence and materials for the nacelle structure. The proposed methodology can be applied for further analysis and optimization of composite elements, particularly when considering the effects of vibrational loads in future studies.

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**УДК 539.3**

# ЧИСЕЛЬНИЙ РОЗРАХУНОК НАПРУЖЕНО-ДЕФОРМОВАНОГО СТАНУ ОБОЛОНКОВОЇ КОНСТРУКЦІЇ ГОНДОЛИ ЛІТАКА

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***Резюме.*** *Представлено результати чисельного аналізу напружено-деформованого стану (НДС) композитної оболонкової конструкції гондоли авіаційного двигуна із застосуванням методу скінченних елементів (МСЕ). Модель центральної частини гондоли побудовано у вигляді циліндричної сендвіч-оболонки з обшивками з вуглепластикових ламінатів (CFRP) та осердям з сот Nomex. Отримано розподіли напружень і переміщень при дії внутрішнього тиску, що імітує експлуатаційне навантаження, та показано, що найбільш навантаженими є шари з орієнтацією волокон ±45°, які чутливі до комбінованого напруженого стану. Виконано перевірку міцності на рівні окремих шарів за критерієм Цай–Ву, що враховує ортотропні властивості матеріалу. Отримано, що для всіх шарів значення індексу руйнування (FI) залишаються меншими за одиницю, а мінімальні значення запасу міцності (MS) є додатними, що свідчить про працездатність конструкції при внутрішньому тиску 0,035 МПа. Звідси випливає, що розглянута конфігурація укладки шарів [0/45/0]° забезпечує необхідний рівень статичної міцності та відповідає вимогам надійності для умов експлуатації. Показано, що застосування оболонкових чотирикутних скінченних елементів забезпечує адекватну точність і збіжність розрахунку, а врахування ортотропних характеристик шарів дозволяє коректно прогнозувати роботу композитних сендвіч-панелей у складі авіаційних конструкцій. Результати дослідження підтверджують доцільність використання обраної методики та обраних матеріалів для побудови моделей гондол, а також можуть бути застосовані при подальшій оптимізації конструкцій і врахуванні вібраційних навантажень у перспективних дослідженнях. Практична значущість роботи полягає у створенні бази для подальшого використання МСЕ-моделей при проектуванні й сертифікаційних перевірках авіаційних композитних компонентів. Методологічна новизна дослідження полягає у поєднанні аналізу НДС із критерієм Цай-Ву для комплексного оцінювання працездатності, що може бути розширене на інші типи оболонкових елементів літальних апаратів.*

***Ключові слова:*** *напружено-деформований стан, метод скінченних елементів, гондола літака, сендвіч-композит, чисельне моделювання, критерій Цай-Ву.*