Structural optimization of a light aircraft composite wing

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Abstract. This paper presents the results of structural optimization of a two-spar composite wing for a light aircraft. Different lay-up structures of power components such as spars, ribs and skin panels was considered. The problem of optimization was solved using Pareto optimization method with two criteria: minimal wing weight and minimal wing deflection under aerodynamic load. The angle of ply orientation and power components thicknesses were considered as the optimization parameters. It was shown that the use of the optimized wing from carbon composite could reduce up to 60% of the wing weight comparing with aluminium one. The stress-strain and buckling analysis of the wing and its power components were carried out in FEMAP software package.

1. Introduction

Currently, polymer composite materials are widely used in many branches of industry. Weight reduction is the greatest advantage of composites usage and is the key factor in using them in aircraft constructions [1 - 5]. When developing a composite structural part, it is necessary to choose reinforcement and matrix of the composite, as well as the structural parameters such as reinforcement orientation angles and ply thicknesses. [8]. This problem can be solved by means of optimization methods. The application of these methods becomes the benefit that allow to decrease as an aircraft weight and a cost as well [1 - 3].

In this study the wing of a light aircraft K-8 was considered as a two-spar structure. Aerodynamic loads acting on the wing were calculated basing on the approaches developed by Zhitomirskii [6] and Neubauer [7].

The obtaining of high mechanical properties of the wing construction was associated with varying of ply orientation in a composite laminate. Four types of composite lay-up structures were considered such as: (a) +45°, -45°, 90°, 0°; (b) -45°, +45°, 0°, -45°; (c) 90°, 0°, 90°; (d) 0°.

The objective of this work was to determine the optimal composite structure providing the minimal weight of the wing. The stress-strain and buckling analysis of the wing was carried out in FEMAP software package using nonlinear static analysis type.

2. Approach to optimization of the composite wing

2.1. Configuration and layout of light aircraft wing
A trapezoidal taper wing shape was considered in this study. Power components of wing consisted of the front spar, rear spar, skin panels and ribs. The front spar was located at the distance of 20% of the wing width from the leading edge while the rear spar – at 70%. The set of transverse force elements – ribs included 10 items placed at the distance of 410.5 mm between each other. The root chord of the wing was 2412 mm while the tip chord 1122 mm.

The main function of the wing skin was to withstand an aerodynamic pressure. The load acted directly on the wing skin, which transmitted it to the ribs. The function of a rib was to maintain the airfoil shape of wing and provide the stability of spars and wing panels. The set of ribs transmitted the aerodynamic load to spars in the proportion to the ribs stiffnesses. The destination of spars was to withstand the wing bending during the flight and thus to reduce the stress in the skin and ribs.

2.2. Simulation models of a wing

Two simulation models were used in this study. At first the analysis of wing frame aimed the determination the optimal laminate structure of spars and ribs was carried out. A frame model of a wing, shown in figure 1(a), was applied for this stage of analysis.

In the second stage, the optimal laminate structure of wing skin was found out by means of a full wing model which shown in figure 1(b).

![Figure 1](image)

Figure 1. (a) Wing frame model; (b) Full wing model

2.3. Ply properties and laminate structure

A laminate carbon composite with monolayer thickness of 0.125mm was chosen for all structural parts of the wing. Properties of the carbon-epoxy ply are shown in table 1.

| Materials properties                  | Value | Unit   |
|--------------------------------------|-------|--------|
| Longitudinal modulus, $E_{11}$       | 181   | GPa    |
| Transverse modulus, $E_{22}$         | 10.3  | GPa    |
| In-plane shear modulus, $G_{12}$     | 7.17  | GPa    |
| Longitudinal tensile strength, $\sigma_{1t}$ | 1500 | MPa    |
| Longitudinal compressive strength, $\sigma_{1c}$ | 1500 | MPa    |
| Transverse tensile strength, $\sigma_{2t}$ | 40   | MPa    |
| Transverse compressive strength, $\sigma_{2c}$ | 246  | MPa    |
| In-plane shear strength, $\tau_{12}$ | 68    | MPa    |
| Major Poisson’s ratio, $\nu_{12}$    | 0.28  |        |
| Density, $\rho$                      | 1600  | kgm$^{-3}$ |

The choice among 4 types of composite laminate structures, which are shown in figure 2 was performed in the study. The thickness of each ply in laminate was the same and equal to 1.25mm.
2.4. Analysis method

The problem of composite wing optimization was considered as a two-criterion optimization with two independent criteria: minimum weight and minimum wing deflection. The optimization parameters were the orientation angles of the plies, as well as the thickness of the spars, ribs and skin.

The optimal wing design was determined by using Pareto optimization method with "ideal center" criterion.

3. Results and Discussion

3.1. Load acting on a wing

Calculation of loads for both frame and full simulation model of the wing was based on the demand that take-off weight of a light aircraft is equal to \( G_0 = 4330 \) \( \text{kg} \) while the wing weight is

\[
G_w = 0.14 \cdot G_0 = 606.2 \text{ kg}.
\]

The uniform pressure corresponded to these weights was applied to the bottom wing skin and considered as load in the analysis of full wing model. It was assumed that a safety factor of a light aircraft was equal to \( f = 1.5 \) and maximum operational overload factor was \( n_{max} = 7.8 \).

A load in the form of linear distributed force applied to the lower contour of the spars was used in the analysis of wing frame model. This distributed force was statically equivalent to the pressure. It was calculated by following formulas:

\[
q_y = n_{max} \cdot f \cdot G_0 \cdot b(z)/S
\]

(1)

\[
q_u = n_{max} \cdot f \cdot G_w \cdot b(z)/S
\]

(2)

\[
q_s = q_y - q_u
\]

(3)

\[
P_z = q_s / b(z)
\]

(4)

\[
q_s = \sum_{i=1}^{a} q_i = q_1 + q_2
\]

(5)

\[
q_1 = H_1^2 q_z \left(H_1^2 + H_2^2\right)
\]

(6)

\[
q_2 = q_z - q_1
\]

(7)

where \( z \) - length of wing span; \( b(z) \) - length of wing chord; \( q_y \) - distributed lift force acting on the wing; \( q_u \) - distributed structural load of the wing; \( q_s \) - resulting aerodynamic load acting on the wing; \( P_z \) - resulting distributed pressure acting on the wing; \( q_1, q_2 \) - aerodynamic loads acting on the front and rear spars; \( H_1, H_2 \) - heights of front and rear spars.
3.2. Optimal thicknesses of wing frame parts and laminate structure

There were 20 variants of the wing frame with different thickness of the spars and ribs analyzed. The stress-strain analysis was carried out under the variable linear distributed loads acting along the wing spars. The model of plies arrangement $+45^\circ$, $-45^\circ$, $90^\circ$, $0^\circ$, $90^\circ$, $-45^\circ$, $+45^\circ$ was took in each case. The analysis results are shown in Table 2.

Table 2. Results of design parameters for wing frame

| No. | Thickness of front spar (mm) | Thickness of rear spar (mm) | Thickness of ribs (mm) | Deflection (mm) | Weight of wing frame (kg) |
|-----|-----------------------------|-----------------------------|------------------------|-----------------|--------------------------|
| 1   | 4.625                       | 4.625                       | 2.375                  | 42.39           | 15.25                    |
| 2   | 4.625                       | 4.625                       | 3.375                  | 41.68           | 18.41                    |
| 3   | 4.625                       | 4.625                       | 4.625                  | 40.57           | 22.34                    |
| 4   | 4.625                       | 3.375                       | 5.625                  | 56.67           | 24.65                    |
| 5   | 3.375                       | 4.625                       | 5.625                  | 37.04           | 24.33                    |
| 6   | 4.625                       | 4.625                       | 5.625                  | 39.44           | 25.53                    |
| 7   | 5.625                       | 4.625                       | 2.375                  | 43.17           | 16.21                    |
| 8   | **4.625**                   | **5.625**                   | **2.375**             | **34.25**       | **15.95**                |
| 9   | 5.625                       | 5.625                       | 2.375                  | 35.29           | 16.92                    |
| 10  | 5.625                       | 4.625                       | 3.375                  | 42.41           | 19.37                    |
| 11  | 4.625                       | 5.625                       | 3.375                  | 33.86           | 19.12                    |
| 12  | 5.625                       | 5.625                       | 3.375                  | 34.83           | 20.08                    |
| 13  | 5.625                       | 4.625                       | 4.625                  | 41.15           | 23.33                    |
| 14  | 4.625                       | 5.625                       | 4.625                  | 33.22           | 23.07                    |
| 15  | 5.625                       | 5.625                       | 4.625                  | 34.04           | 24.04                    |
| 16  | 5.625                       | 3.375                       | 5.625                  | 56.25           | 25.62                    |
| 17  | 3.375                       | 5.625                       | 5.625                  | 29.96           | 25.04                    |
| 18  | 5.625                       | 4.625                       | 5.625                  | 39.87           | 26.50                    |
| 19  | 4.625                       | 5.625                       | 5.625                  | 32.57           | 26.24                    |
| 20  | 5.625                       | 5.625                       | 5.625                  | 33.23           | 27.21                    |

The optimal thicknesses of the wing frame parts were defined by using two criteria: minimal deflection and minimal weight in Pareto multicriteria optimization. The alternative variants of wing frame thicknesses are shown in figure 4 as points in criteria plane (weight – deflection). Structural parameters for the optimal wing frame have been determined as follows: for front spar – 4.625mm, rear spar – 5.625mm and ribs – 2.375mm.
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Table 3. Deflection of the wing frame vs type of spars and ribs laminate structure

| No. | Spars laminate structure | Ribs laminate structure | Deflection (mm) |
|-----|--------------------------|-------------------------|-----------------|
| 1   | +45°,-45°,90°,0°,90°,-45°,+45° | +45°,-45°,90°,0°,90°,-45°,+45° | 34.25 |
| 2   | +45°,-45°,90°,0°,90°,-45°,+45° | +45°,-45°,0°,-45°,+45° | 34.27 |
| 3   | +45°,-45°,90°,0°,90°,-45°,+45° | 90°,0°,90° | 34.34 |
| 4   | +45°,-45°,90°,0°,90°,-45°,+45° | 0° | 34.41 |
| 5   | +45°,-45°,0°,-45°,+45° | +45°,-45°,90°,0°,90°,-45°,+45° | 25.67 |
| 6   | +45°,-45°,0°,-45°,+45° | +45°,-45°,0°,-45°,+45° | 25.74 |
| 7   | +45°,-45°,0°,-45°,+45° | 90°,0°,90° | 25.70 |
| 8   | +45°,-45°,0°,-45°,+45° | 0° | 25.89 |
| 9   | 90°,0°,90° | +45°,-45°,90°,0°,90°,-45°,+45° | 28.38 |
| 10  | 90°,0°,90° | +45°,-45°,0°,-45°,+45° | 28.36 |
| 11  | 90°,0°,90° | 90°,0°,90° | 28.46 |
| 12  | 90°,0°,90° | 0° | 28.43 |
| 13  | 0° | +45°,-45°,90°,0°,90°,-45°,+45° | 16.78 |
| 14  | 0° | +45°,-45°,0°,-45°,+45° | 16.77 |
| 15  | 0° | 90°,0°,90° | 16.80 |
| 16  | 0° | 0° | 16.81 |

Figure 4. The alternative variants of wing frame and "ideal center"

At the next step of work 16 types of laminate structures for the wing frame were considered. The results of this step of study are presented in table 3. The unidirectional arrangement of plies (0°) for spars and (+45°,-45°,0°,-45°,+45°) for ribs were selected as the optimal laminate structures.

Table 4. Deflection of the wing frame vs skin laminate structure

| No. | Spars laminate structure | Ribs laminate structure | Deflection (mm) |
|-----|--------------------------|-------------------------|-----------------|
| 1   | +45°,-45°,90°,0°,90°,-45°,+45° | +45°,-45°,90°,0°,90°,-45°,+45° | 84.65 |
| 2   | +45°,-45°,90°,0°,-45°,+45° | +45°,-45°,0°,+45° | 67.41 |
| 3   | 90°,0°,90° | 90°,0°,90° | 78.72 |
| 4   | 0° | 0° | 58.04 |

3.3. Optimal laminate structure for wing skin

Optimal laminate structure for wing skin was obtained from consideration of 4 types of laminate structures. The thickness of skin was taken of 4.625 mm. The results presented in table 4 showed that the optimal laminate structure for composite skin, corresponding to minimal deflection, was unidirectional (0°) plies arrangement.
3.4. Check with safety factors for the design of optimal composite wing
Verification of the optimal wing model was carried out at the last step of the study. The maximal acting aerodynamic load was equal to 0.01192MPa. Static analysis showed that maximal stress in the composite parts of wing did not exceed the ultimate strength of the composite ply. According to nonlinear static analysis of the wing under the stepwise increasing load the safety factor was obtained as high as 3.3 (figure 5).

![Figure 5. Nonlinear wing deflection vs stepwise increasing load](image)

4. Conclusions
The main results of the study may be formulated as follows:
- The approach to design of composite wing based on multicriterial optimization was proposed;
- Optimal composite laminate structures and optimal thicknesses for composite parts of wing were obtained;
- Verification of designed composite wing was carried out by both static and buckling analysis.

References
[1] Qun Z, Yunliang D, Haibo J 2011 A Layout Optimization Method of Composite Wing Structures Based on Carrying Efficiency Criterion Chinese Journal of Aeronautics 24 425–433.
[2] Shabeer K P, Murtaza M A 2013 Optimization of aircraft wing with composite material. International Journal of Innovative Research in Science Engineering and Technology 2 2471–2477.
[3] Liu Q, Mulani S, Kapani R K 2014 Global/Local Multidisciplinary Design Optimization of Subsonic Wing Proceedings of the 10th AIAA Multidisciplinary Design Optimization Conference 13-17.
[4] Wang Y, Ouyang X, Yin H, Yu X 2016 Structural-Optimization Strategy for Composite Wing Based on Equivalent Finite Element Model. Journal of Aircraft 53 351–359.
[5] Tatarnikov O, Gaigarova M 2012 Three-Level Design of Composite Structures Proceedings of the 2nd International Conference on ACMTAA 108–112
[6] Zhitomirskii G I 2005 Konstruktsiya samoletov [The design of the aircraft] Moscow: Mashinostroenie 406.
[7] Neubauer Dr M, Günther G 2000 Aircraft Loads RTO AVT Lecture Series 1–4 15–17.
[8] Nisha Mary K et al 2014 Ply Orientation of Carbon Fiber Reinforced Aircraft Wing – A Parametric Study Int. Journal of Engineering Research and Applications 4 53–55.