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# ANALYSIS OF THE STRESS-DEFORMED STATE OF THE AIRCRAFT FLAP

**Ua** Закрилки літака відіграють важливу роль в експлуатації літака. Вони використовуються для збільшення підйомної сили літака на заданій швидкості, що дозволяє здійснювати повільніший зліт і посадку. Збільшуючи розвал крила, закрилки забезпечують додаткову підйомну силу, що особливо корисно під час зльоту та посадки, коли літаку потрібно підтримувати контроль на низьких швидкостях. Крім того, закрилки також допомагають збільшити опір літака, що дозволяє уповільнити його під час посадки. Закрилки мають складну геометричну форму і тому для оцінки їх напружено-деформованого стану, необхідно застосовувати чисельні методи, насамперед метод скінченних елементів. В роботі на основі метода скінченних елементів були визначенні напруження при реальних експлуатаційних навантаженнях. Запропонована методика дає можливість здійснити як перевірочні розрахунки при проектування, так і оптимізацію маси закрилка літака.

**En** Aircraft flaps play an important role in the operation of the aircraft. They are used to increase the lifting force of the aircraft at a given speed, which allows for slower take-off and landing. By increasing the camber of the wing, the flaps provide additional lift, which is especially useful during takeoff and landing when the aircraft needs to maintain control at low speeds. In addition, flaps also help increase the drag of the aircraft, which slows it down during landing. The flaps have a complex geometric shape, and therefore, to assess their stress-strain state, it is necessary to use numerical methods, primarily the finite element method. In the work based on the method of finite elements, there were determinations of stress under real operating loads. The proposed method makes it possible to carry out both verification calculations during design and optimization of the mass of the aircraft flap.

## Introduction

The flaps are designed to improve the take-off and landing characteristics of the aircraft. Aircraft flaps play a critical role in aircraft operation. They are mainly used to increase the lift of the aircraft at a given speed, which allows for slower take-offs and landings. By increasing the camber of the wing, flaps can provide additional lift, which is especially useful during takeoff and landing

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when the aircraft needs to maintain control at low speeds. In addition, flaps also help increase the drag of the aircraft, which helps slow it down during landing. This is critical for the aircraft to land safely on the runway. Therefore, the analysis of the stress-strain state of this structure is a very urgent task. It should be noted that the analytical calculation of flap structures is not always effective. This is due to the complex geometry of the structure, load conditions and boundary conditions of the boundary value problem that arises during the calculation [1 - 5]. Therefore, numerical methods for calculating the stress-strain state of an aircraft flap are preferred in the work, namely the finite element method [6 - 10].

### Selection of materials for structural elements

The flap is attached to the wing with supports. The main link of the flap consists of a caisson, nose and tail parts. The caisson consists of two spars, strips, upper and lower panels and ribs. The tail part is made of a frame and a lower technological panel. The frame consists of upper and lower panels and ribs. All frame parts are made of composite materials. With the help of the graphic editor Solid Works, geometric 3-D models of the flap claddingwere obtained.

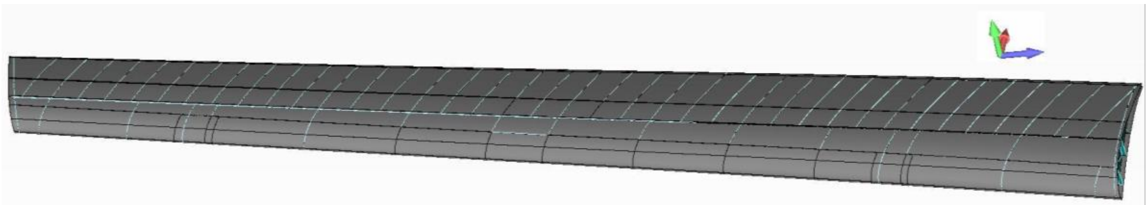

Fig. 1. 3-D model of the flap

Fig. 1 shows a 3-D model of a closed plane, created using the Solid Works program. The choice of materials for the structural elements of the flap is determined by operating conditions and strength characteristics. The spar and front ribs are made of dural-aluminum alloy (D16T). The thickness of the spar walls is taken as average δ=2,5mm. The rear ribs are made of polymer composite material UOL900. The thickness of the ribsis δ=2,5mm.

Fig. 2 shows the model of the flap covering, taking into account the composite structure of the material.

The casing of the caisson is made of D16T and has a thickness of δ=8mm. The back paneling is made of KM UOL900 and has a thickness of δ=3mm. The front cladding is made of UT250 carbon fiber and has a thickness of δ=3mm.

The main properties of the materials used in the working calculations are listed in Tab. 1.

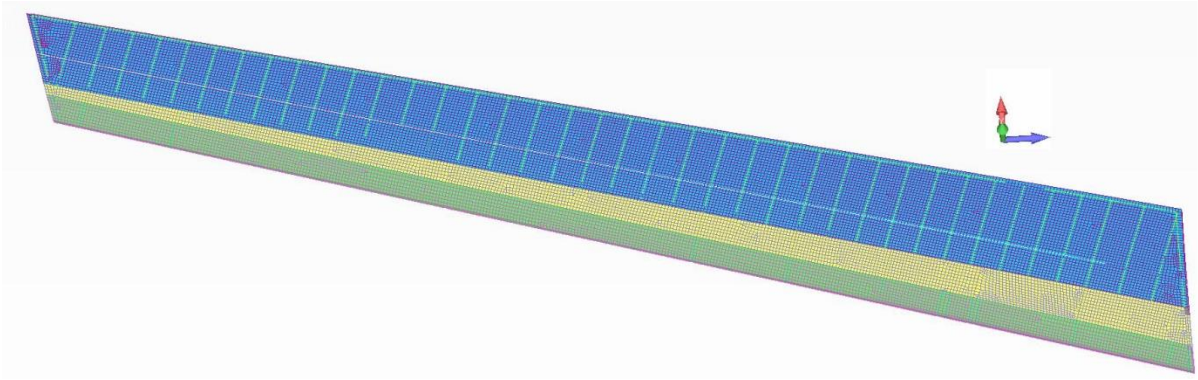

Fig. 2. Model of composite flap cladding

***Table* 1.**

The main properties of the materials

| name | $\sigma_B$ kgf / mm$^2$ | E kgf / mm$^2$ | G kgf / mm$^2$ | μ |
|---|---|---|---|---|
| D16T | 50 | 7200 | 2700 | 0,3 |
| WAL300 | 120 | 4900 | 1200 | 0,19 |
| UT900 | 55 | 2000 | 1000 | 0,28 |

The load mode of the flap was chosen as follows: loads $P_{n\ оз}^{P}$, $P_{\tau\ оз}^{P}$ lie in a plane parallel to the front wall; loads $P_{n\ оз}^{P}$ is perpendicular to the proper chord of the main link; loads $P_{\tau\ оз}^{P}$ is perpendicular to the own chord of the main link – the load is directed along the own chords of the deflector and the main link, respectively, the positive direction $P_{z\ D}^{P}$ - is back from the nose point.

$P_{n\ оз}^{P}$=424 kgf; .. $P_{\tau\ оз}^{P}$=-264 kgf; .. $P_{z\ D}^{P}$=–814 kgf; .. $\overline{x}_{D\,оз}$=0,3.

The distribution of the load along the span of the flap is taken in proportion to the chords.

### Calculation of the stress-deformed state of the flap

Consider the following boundary value problem

– the equation of equilibrium as a special case of the equation of motion, in a generalized form:

$$\nabla^{n}\sigma_{mn}=0 \tag{1}$$

– geometric (for small deformations), in a generalized form:

$$\varepsilon_{ij} = \frac{1}{2}\left(\nabla^i U_j + \nabla^j U_i\right) \tag{2}$$

– as well as deformation compatibility equations:

$$\varepsilon_{ij} = \varepsilon_{ij}^e \tag{3}$$

– physical equations

$$\varepsilon_{ij}^e = C_{ijmn}\sigma_{mn} \tag{4}$$

where $C_{ijmn}$ is the modulus of elasticity tensor.

Boundary conditions on $S_U$ and $S_P$ are additionally involved:

$$U_i\big|_{S_U} = \hat{U}_i; \tag{5}$$

$$\sigma_{mn}\nu_n\big|_{S_P} = \hat{P}_m \tag{6}$$

To solve the boundary value problem, it is convenient to have its variationalstatement, which is what we use.

As a result, it is possible to obtain the following functional with respect to variations of displacements and associated deformations,

$$F = \int_{\Omega} \sigma_{mn}\delta\varepsilon_{mn}d\Omega - \int_{S_P} \hat{P}_m \delta U_m dS\ . \tag{7}$$

In the Finite Element Method, Hooke's linear law is written in the form

$$\{\sigma\} = [D]\{\varepsilon^e\}, \tag{8}$$

where [$D$] is the matrix of elastic moduli.

In the case of elastic isotropy of the material matrix

$$[D] = 2G \cdot \begin{pmatrix} a & b & b & 0 & 0 & 0 \\ b & a & b & 0 & 0 & 0 \\ b & b & a & 0 & 0 & 0 \\ 0 & 0 & 0 & c & 0 & 0 \\ 0 & 0 & 0 & 0 & c & 0 \\ 0 & 0 & 0 & 0 & 0 & c \end{pmatrix}, \tag{9}$$

where $2G = E/(1+\mu)$; $a = (1-\mu)/(1-2\mu)$; $b = \mu/(1-2\mu)$; $c$ = 0,5; $E$ is the Young's modulus; μ is the Poisson’s ratio.

All deformations are elastic, therefore

$$\{\varepsilon\} = \{\varepsilon^e\} \tag{10}$$

Considering (10) and (9), let's write the expression (8) in the form:

$$\{\sigma\} = [D]\{\varepsilon\} \tag{11}$$

$$\{\varepsilon\} = [\mathrm{B}]\{q\}_{\mathrm{e}} \tag{12}$$

$\{q\}_{\mathrm{e}} = \{\left(q^1, q^2, q^3\right)_1, \ldots, \left(q^1, q^2, q^3\right)_{\mathrm{M}}\}^{\mathrm{T}} = \{q_1, q_2, \ldots, q_{3M}\}^{\mathrm{T}}$ is the vector of displacements in the nodes of the finite element; [$B$] is the differentiation matrix by global coordinates; [$B$] elated only to the type of finite elements and the global coordinate system.

Functional (7), taking into account the possibility of superposition of works on finite element, is written as follows:

$$F = \sum_{e} \int_{\Omega^e} [B]^T \{\delta q\}_{\mathrm{e}}^{\mathrm{T}} [D][B]\{q\}_{\mathrm{e}} d\Omega - \sum_{e} \int_{S_P^e} [\varphi]^T \{\delta q\}_{\mathrm{e}}^{\mathrm{T}} \{\hat{p}\} d\mathrm{S}, \tag{13}$$

where the load vectors are marked $\{\hat{p}\} = \{\hat{p}_1, \hat{p}_2, \hat{p}_3\}^{\mathrm{T}}$; $S_P^e$ there is a finite element side facing $S_p$ of the body; sign $\sum\limits_{e}$ means the addition of all finite elements containing the considered degree of freedom of the node.

Let’s mark:

$$[K]_e = \int_{\Omega^e} [B]^T [D][B] d\Omega, \tag{14}$$

$$\{P\}_{\mathrm{e}} = \int_{S_P^e} [\varphi]^T \{\hat{p}\} d\mathrm{S}. \tag{15}$$

Then

$$F = \sum_{e} \{\delta q\}_{\mathrm{e}}^{\mathrm{T}} \left([K]_e \{q\}_{\mathrm{e}} - \{P\}_{\mathrm{e}}\right). \tag{16}$$

Using the minimization condition and taking into account the kinematic boundary conditions, we obtain a system of algebraic equations in the form

$$[K]\{q\} = \{P\}. \tag{17}$$

As a calculation object, we use the 3-D model of the constructed flap, (Fig. 2). The generated finite elements grid has 42,543 nodes and 45,328 elements.

Let's consider the results of the flap calculation using the finite element method.

Fig. 3 shows the calculation of the stress-strain state of the upper and lower flap cladding according to Mises.

As we can see from Fig. 4, the maximum stresses occur in the supports and the place of maximum deflection of the flap. VAT in the zone of maximum flap deflection.

## Conclusions

Taking into account the complexity of the structure of the flap of a transport aircraft, its load and boundary conditions, the work presents a methodology for calculating the stress-strain state of this structure based on the finite element method. This made it possible to accurately model the load, geometry and boundary conditions of the corresponding boundary value problem. Stresses under real operational loads, which are necessary to solve important tasks related to the operation of the aircraft as a whole, were determined.

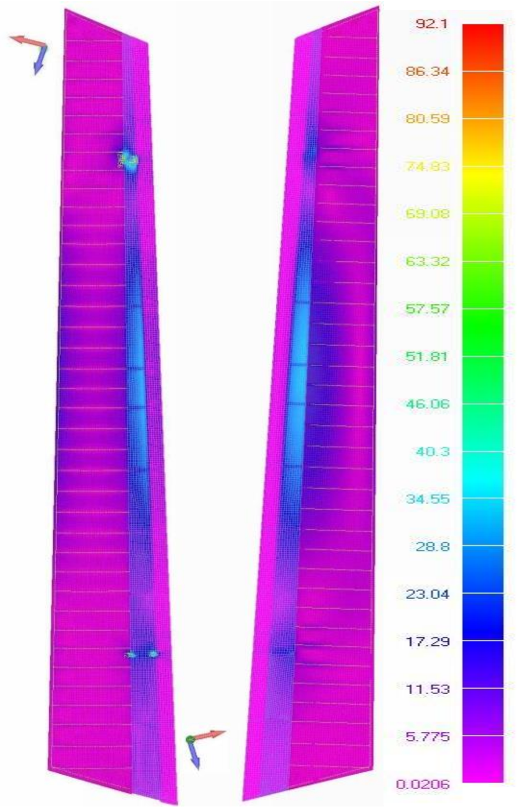


Fig. 3. Stress-deformed state of the upper and lower cladding of the flap according to Mises

The proposed calculation method also makes it possible to optimize the mass of the aircraft flap.

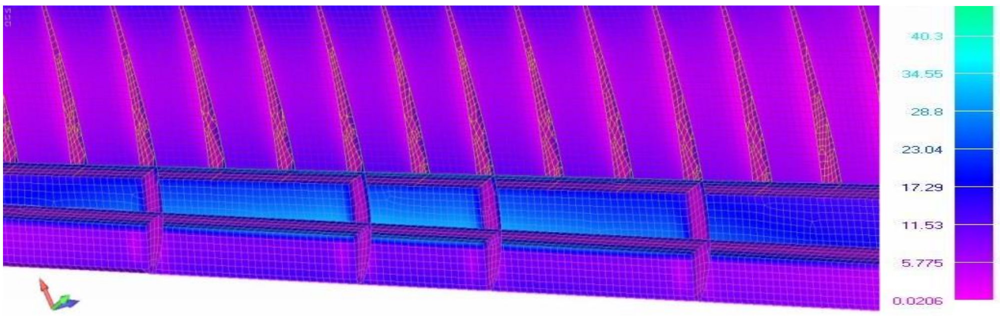


Fig. 4. Stress-deformed state in the zone of maximum flap deflection according to Mises

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