SOME LAMINATE DEFORMATIONS FEATURES

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The present stage of aviation technology development is characterized by polymer-based composite materials wide implementation into load-bearing airframe structure. The increase of composite materials use in aircraft structures is shown in Fig. 1.

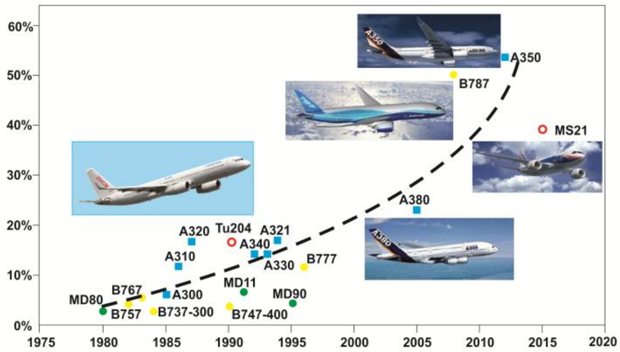

Fig. 1 - The increase of composite materials use in aircraft structure

Carbon fiber reinforced layered composites with epoxy matrix are commonly used for this purpose. The basic laminate unit is a monolayer - unidirectional composite. Monolayers with different orientations are composed to form the laminate - an anisotropic material. Due to the principal difference between composite and isotropic materials, isotropic failure criteria are not applicable to laminates. There are special layered composites failure criteria: Hill- Mises, Tsai - Wu [1] Hoffman , as well as Hashin [2] and Pak [3] failure criteria that enable estimating separately the strength of fiber and matrix. Using these criteria allows determining the monolayer with the lowest safety factor; this monolayer determines the laminate primary failure safety factor. However, this estimation doesn't determine the total strength of laminate. In fact, laminate with destructed matrix in several layers can withstand the load up to the moment when the fiber failure process begins. This fact must be taken into account for a more accurate laminate carrying capacity determination.

The failure criteria, mentioned above, determine the strength of a layer and the results are confirmed experimentally. However, the situation changes when the strength of laminate with different layers orientations is estimated. A failure criteria comparative analysis was made for various combinations of shear and tension (compression) loads. A significant discrepancy in predicted laminate strength is observed, especially in combined shear and tensile loads cases (Fig 2.). In such a situation, we have to rely on the criterion that gives the lowest safety factor that is to rely on Hill, Hofmann, Tsi-wu criteria.

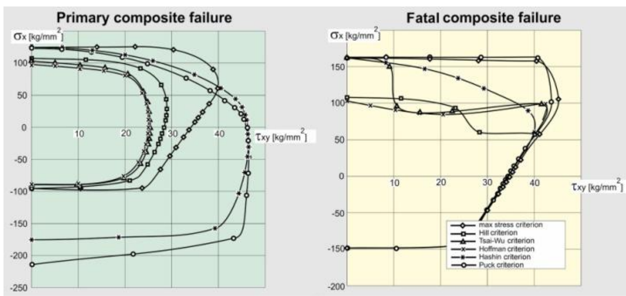

Fig. 2 - Laminate strength diagrams for various failure criteria

An iterative algorithm was used in laminate strength calculation, that allowed to analyze the matrix strength (primary failure) as well as the fiber strength (fatal failure). The account for layer degradation was made by nulling the matrix stiffness of a layer with the lowest safety factor for a given layer failure criterion. There were two criteria for laminate failure. They are: the fiber safety factor is to be lower than the matrix safety factor and the laminate matrix is destructed completely. This algorithm application makes it possible to detect the matrix cumulative failure effect for some laminates (Fig. 3). This effect consists in the whole matrix chain reaction failure which can happen after matrix failure in any ply. The load in this process is constant (method 2 Fig 3). Previously used algorithms had not found it out (method 1 Fig 3).

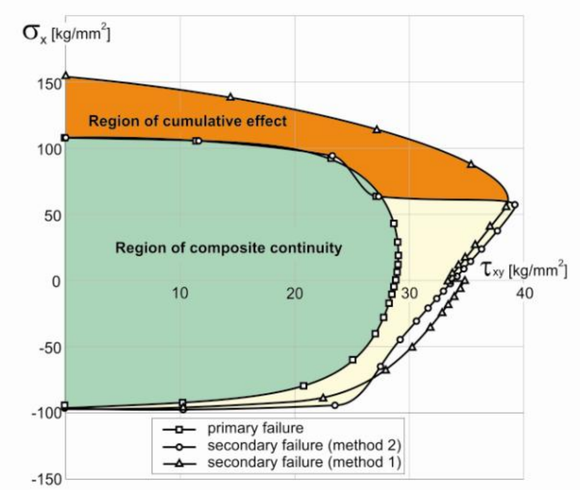

Fig. 3 - Laminate matrix destruction cumulative effect

High numerical results dispersion of estimated laminate strength requires the more careful failure criteria selection and the results analysis. The experimental and numerical strength research results of laminates from various materials and with various layers combinations make it possible to give the following recommendations:

- In the presence of experimental data, the criterion that most closely agrees with the experimental results for simple load cases (tension, compression, shear) should be chosen.
- the mean value of primary and fatal failure stresses should be taken for data processing. In this case, a satisfactory agreement between the numerical and experimental results is observed.

The laminate deformation has a certain specificity, which should be taken into account in some cases when estimating the strength. Thus, the edge effects may take place at the laminate free edges, and effect the structure strength. [4,5]. The edge effects appear due to differences of Poisson ratios in layers integrated in laminate, and under the condition of their deformation compatibility law. (see Fig 4)

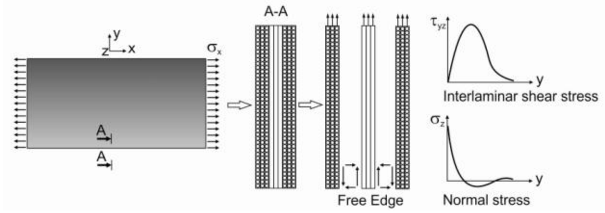

Fig. 4 - Free edge effect formation pattern

Three dimensional finite-element models (FEM) were used to study the edge effects. The example of such a model is presented in Fig. 5.

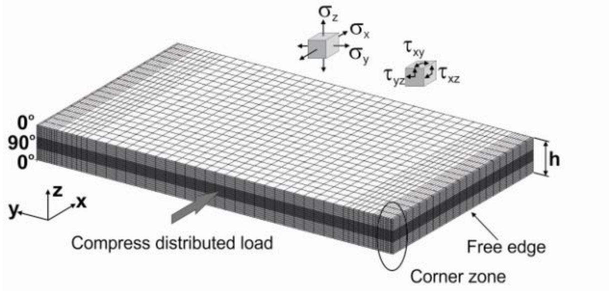

Fig. 5 – FEM to study the edge effects

The stress-strain state of the model shows that there is a triaxial stress state with transverse shear stress components [τ_{yz}] and delaminating normal stresses [σ_z], which decrease as moving from the free edge (see Fig. 6). These stresses values are relatively low, but it must be taken into account that the laminate strength in this direction is also low.

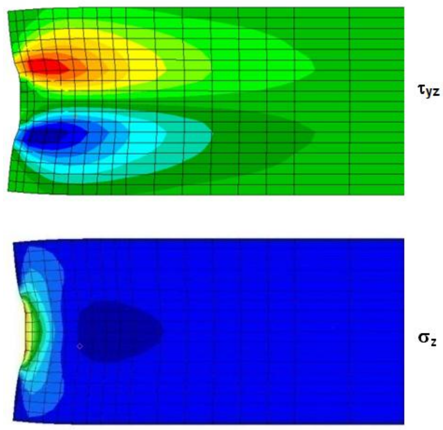

Fig. 6 - τ_{yz} σ_z distribution in laminate

The delaminating stress level and the distribution in free edge area highly depends on the laminate stacking sequence. On this matter, the comparison was made of designed data and the experimental ones [6] and of the specimens, composed of the same layers and materials with various laminate stacking sequences, As a result, it was shown that the laminate stacking sequence affects both the failure load level, and the fracture pattern (see Fig. 7).

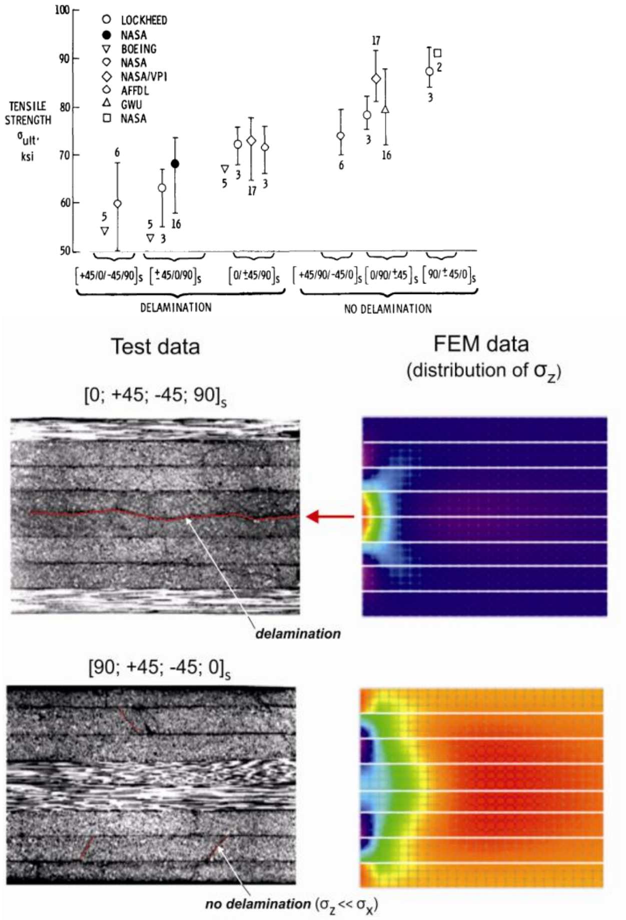

Fig . 7 - Calculated and experimental data comparison

This relationship can't be described in terms of laminate plate theory.

The fact of composite specimens ends premature delamination was detected in tests. In this regard, a typical specimen laminate with stacking sequence $(45/0/-45/0/0/90/0/0/-45/0/45)_3$ from carbon fiber reinforced plastic was investigated for edge effects. Corner region of the plate end was found out to be the most critical area; in this region the normal delaminating stress reaches the value of 14% of mean compressive stress. On the basis of this value and the monolayer transverse strength, the delaminating load level was determined (63 kgf/mm^2).

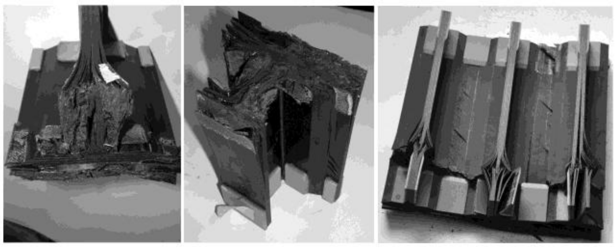
Fig. 8 – Specimens failure patterns

The experimental data analysis showed that the delamination takes place at all specimens ends and that the highest delamination rate is in the corners. The experimental fatal load level was in the range of 55-60 kgm/mm^2, and this value agrees quite well with the numerical results. Thus, it is possible to conclude that the edge effects at panels' ends are reason of the premature laminate specimen delamination.

The edge effects can also take place on holes free edges. Spar web or ribs with holes, working mainly loaded with shear load are constructive examples. Some free edge effect at hole edge under shear load research results are presented further. The research was carried out on three-dimensional FEM of plate with a central hole (see Fig 9).

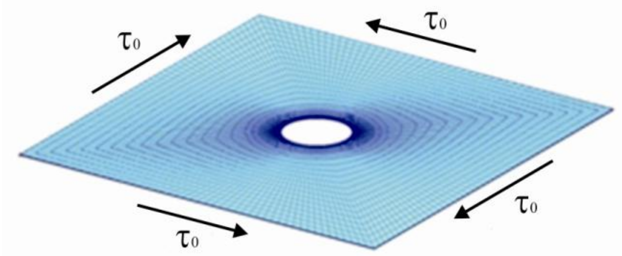

Fig. 9 - Finite-element model

There a significant edge effect is also observed. The edge effect formation mechanism in this case is more complicated in comparison with the uniaxial loading, as laminate in a hole area works in a complex stress state (see Fig. 10).

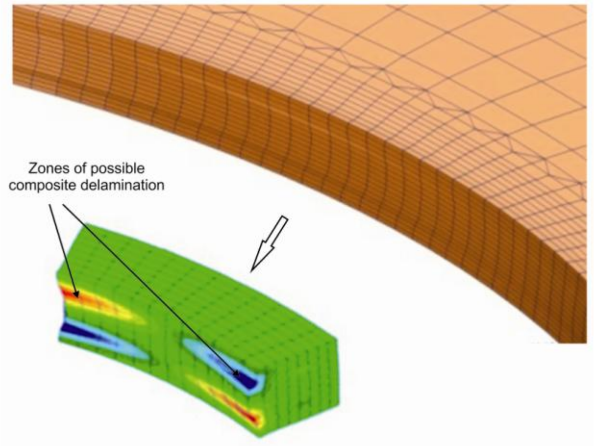

Fig. 10 – The hole area stress strain state

The analyses showed that in the studied cases the most critical situation in terms of possible delamination at the hole edge, take place in the region of layers with 45°orientation and layers with 0° or 90°orientations adjunction. Here the determinative stresses are transverse shear stresses.

Interlayer shear $\tau_{\theta Z}$ and normal stress σ_Z distributions along the hole contour are presented in Fig 11. The stresses are presented in relative values (σ/τ_0) in adjunction area of layers with orientations 45° and 0° for laminate with stacking sequence $(+45/-45/0)_s$.

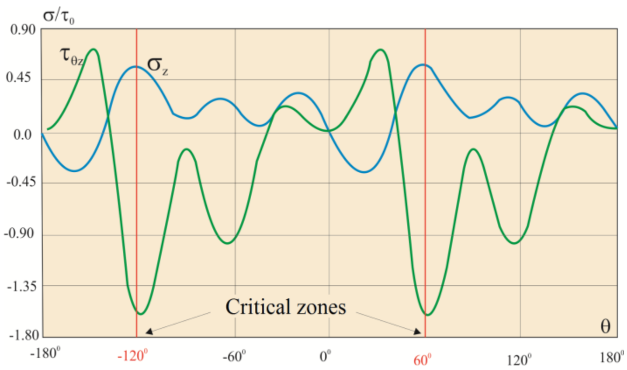

Fig 11 - Delaminating stress distributions at the hole contour

Laminate strength analysis based on Hashin-Rotem failure criteria showed that initial delamination occurs at relatively low load τ_0 level, in comparison with the fatal load predicted by composite plate theory. However the appeared delamination does not mean the structure failure, since the edge effect is realized in a very local area.

Obviously, some steps must be made in order to reduce the edge effect influence on the structure strength. The following methods can be used:

- Specimen ends reinforcing
- Special gear application to avoid the ends delaminations.
- Specimen edges covering with additional isotropic material layer, for example, with epoxy resin, in order to reduce the normal delaminating stress. The computation shows that covering with such layer reduces the normal stresses quite well (Fig 12).

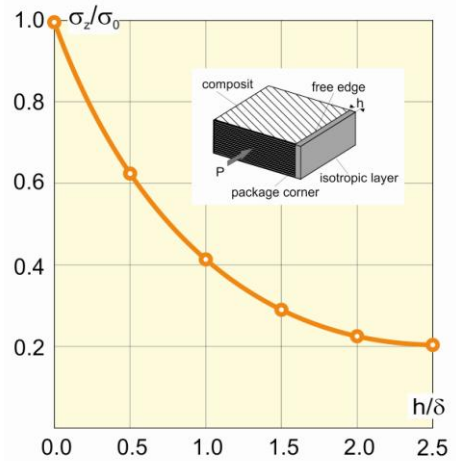

Fig 12. -The relation between delaminating stress and isotropic layer thickness

-In order to reduce the edge effect negative influence on the hole edges the removal of the 0° and 90° layers in hole area may be an effective way.

Another laminate feature is its low transverse shear stiffness, because it is basically determined by the matrix characteristics. It may have a negative influence on the composite structure buckling properties. Thus, expression for simply supported beam buckling stress, taking into account the transverse shear deformations, is:

$$\sigma_{cr} = \frac{\sigma_E G_{xz}}{\sigma_E + G_{xz}},$$

where: σ_E – critical Euler stress, G_{xz} – beam transverse shear modulus.

The expression implies that as the shear modulus tends to zero, the critical stresses tend to its value and does not depend on beam size. The critical buckling stresses dependence of beams σ_{cr} with various transverse anisotropy

G_{xz}/E_x levels from its flexibility μ is shown in Fig. 13.

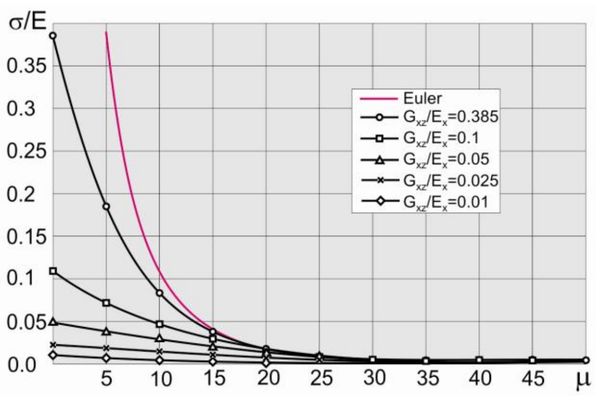

Fig. 13 - Simply supported beam critical buckling stresses

The graph shows that the buckling stresses of short beams sharply decrease as the transverse anisotropy degree increases. For laminates this parameter may be lower than 0.01. Therefore, for extremely short beams, which are actually a laminate model, the critical buckling stress can be lower than their compressive strength; it means that the problem of laminate local buckling emerges.

Parametric-calculated analyses of the transverse shear stiffness influence on the laminate critical buckling stress were carried out. Two-dimensional FEM was used (see Fig. 14).

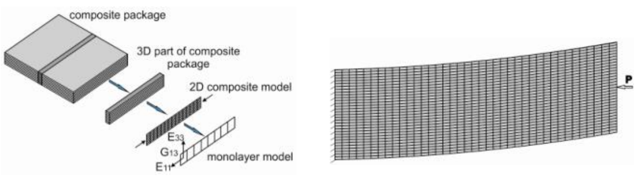

Fig. 14 - The model to analyze the laminate local buckling

It was shown that when the laminate transverse shear stiffness is close to matrix shear stiffness (G_{13}≈135 kgf/mm^2), the critical buckling stress is about 100 kgs/mm^2, that is sufficiently close to the laminate compression strength limit.

The local buckling of laminate with operational or technological defects is a hot problem. Thus, through-thickness matrix cracking the critical stresses sharply decrease as the damaged area is increasing. The pure shear buckling form takes place. (see Fig. 15).

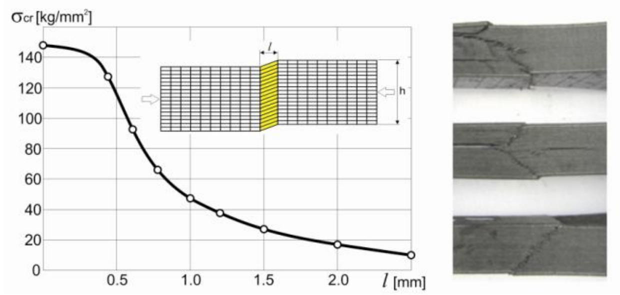

Fig. 15 - The critical buckling stress dependence from the through-thickness matrix cracking area

The critical buckling stresses are reduced more slowly as the delamination area is increasing. In this case, the first eigenvalue corresponded to the buckling mode close to the shear one and the rest eigenvalues corresponded to the modes with delamination opening (see Fig. 16).

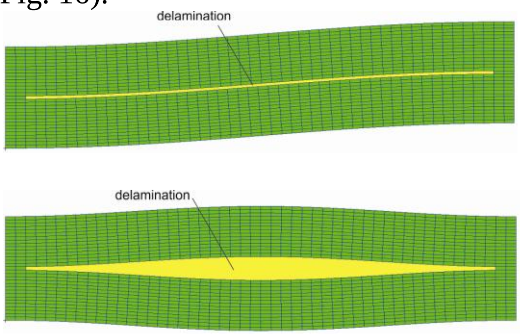

Fig . 16 - The first and the second buckling modes of laminate at presence of delamination

The researches performed enable supposing the following laminate compressive failure mechanism: when the monolayers matrix cracking level in set of stresses is reached, the laminate transverse shear stiffness decrease occurs that results into laminate local buckling.

References

[1] Tsai S.W., Wu E.M. "A General Theory of Strength for Anisotropic Material", Journal of Composite Materials, 5, 58-80, 1971.
[2] Hashin Z. "Failure Criteria for Unidirectional Fibre Composites", Journal of Applied Mechanics, 47, 329-334, 1980.
[3] Puck A., Schurmann H. "Failure analysis of FRP laminates by means of physically based phenomenological models", Composites Scince and Technology, 1988.

[4] Baker A., Dutton S., Kelly D. “Composite Materials for Aircraft Structures”, Second Edition Published by American Institute of Aeronautics and Astronautics Inc.
[5] Dudarkov Yu.I., Limonin M.V., Levchenko E.A. “Free edge effect in layered composites”, Aviation Industry, №4, 2012, p.48-53
[6] Bjeletich J. G., Crossman, F. W., Warren W. J. “The Influence of Stacking Sequence on Failure Modes in Quasi-Isotropic Graphite-Epoxy Laminates”, Failure Modes in Composites IV, AIME, 1977, p. 118.

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