FEM analysis of joinwing aircraft configuration

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ABSTRACT

Novel aircraft configuration – joined wing, or in other nomenclature Prandtl plane can have many aerodynamic and structural advantages, but introduces also new problems to solve. The article focuses on the structural problems, analyzed and optimized with use of Finite Element Method (FEM).

1 INTRODUCTION

The aerodynamic configuration promises great induced drag reduction for high lift coefficients. To check the feasibility of the idea project MOSUPS was started [1, 2, 3]. The project includes multidisciplinary analysis, design and building airworthy aircraft Fig. 1. After preliminary analysis it was proved that front upper wing and lower rear wing, which is less commonly used wings configuration, improves aerodynamic characteristics resulting from positive interference of the wings. Although, aerodynamic benefits where the original motivation for the concept of the Prandtl plane, this configuration could also bring structural benefits. Wings structure of the airplane creates closed frame, which is different from classical cantilever wings configuration. Loads acting on the wings could be optimally distributed between the wings. Higher stiffness of the closed frame may significantly improve aeroelastic features of the lightweight structure.

Figure 1: MOSUPS aircraft in flight.

FEM analysis were done to see how the initially proposed structure works. Three dimensional beam wings model was created. Beams consisting of two covers, and two spars with rods on the lower and upper ends, modeled wingbox structure. It showed some problems with considered simplifications in the regions of connection of wings to the side plate Fig. 2. In the problematic regions the structure had very low thicknesses after optimization which resulted in high stress values. Additionally, any increasing of the thickness in affected regions significantly changed internal loads distribution over whole wings. That makes optimization of the structure more difficult. Not all optimization algorithms assure good convergence. As a result to get the optimal solution it was decided to set section dimensions constant in the joining locations. To investigate the phenomena in the work more accurately shell models were prepared and analyzed. Buckling criterions, which often determine critical conditions, were analyzed for many cases from load envelope. Finally, basic skin thickness optimization was done and results discussed.

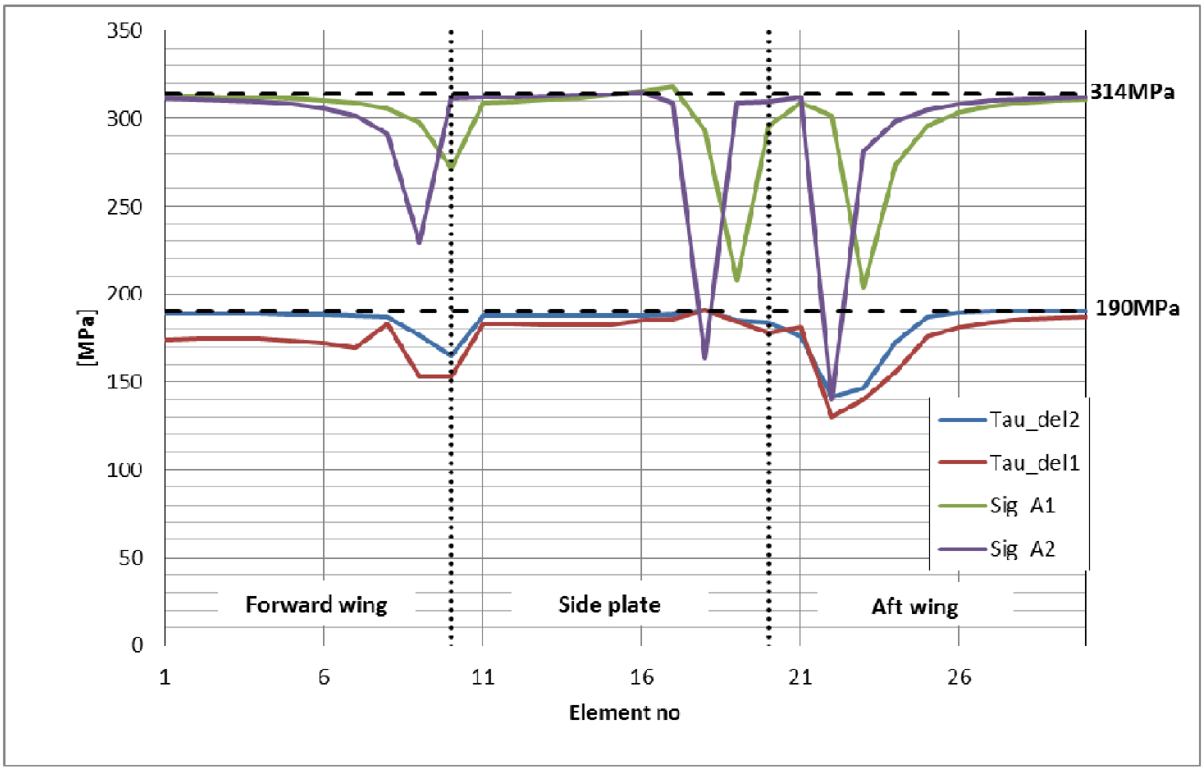

Figure 2: Optimized FEM beam model stress distribution.
Tau are shear stresses in covers, whereas Sig are normal stresses in belts.

2 INITIAL FEM ANALYSIS

Aerodynamic pressure loads, were derived from panel method [4] and mapped on the airplane's structure in FEM software [5] Fig. 3. Numerical model for the FEM analysis was simplified, by elimination of the fuselage and the vertical stabilizer. Wings have added ribs and all panels have constant thickness. Nodes of the root profiles of the front and rear wings were fixed. Simple aluminum alloy material was set for the first analyze.

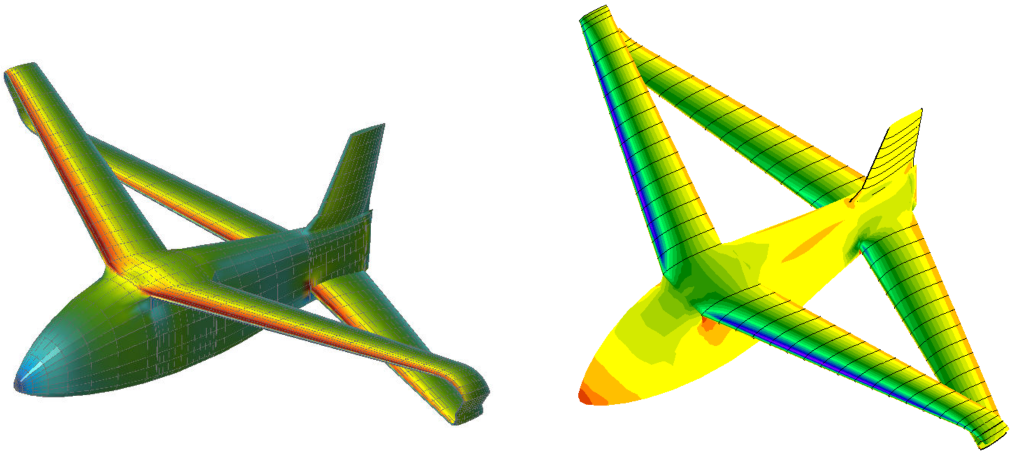

Figure 3: Geometry and pressure loads a) in panel method b) after export to FEM solver.

After the analysis there were found the same stress concentrations at the arc joints between the wings and the side plates connecting the wings Fig. 4. Amplitude of reduced von Mises stress at the arcs is almost half of the maximum stress, which is present near the wings root sections. This result can be explained by high deformation of the wing at the ends, which is similar to the cantilever wing Fig. 5. Structure at the arcs has to rotate and deform significantly, what results in the increased stress.

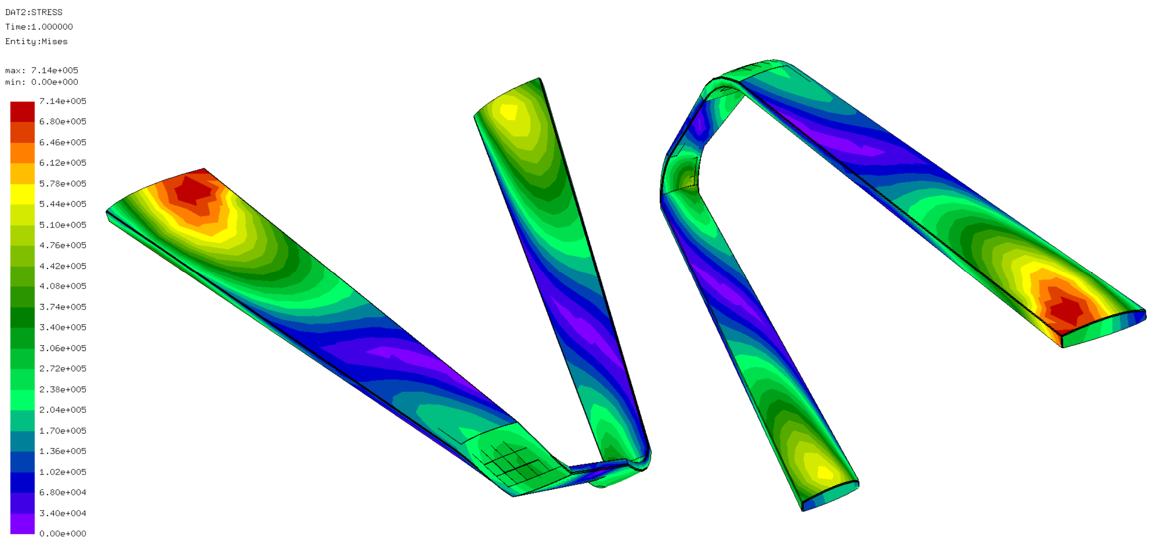

Figure 4: Von Mises stress, with concentration at the arcs.

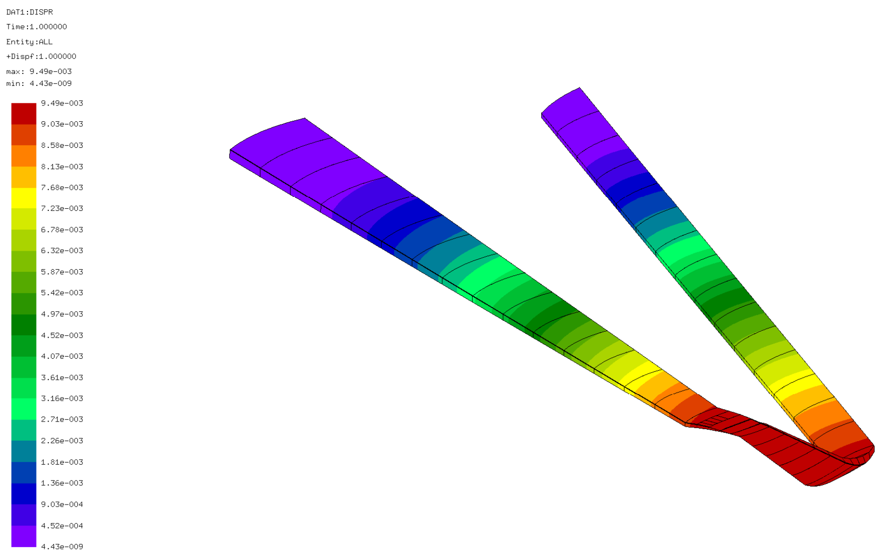

Figure 5: Wing deformation.

3 BASIC STRENGTH OPTIMIZATION

Initial FEM analysis showed the problem of concentrated stress in the area of wing's arcs. Authors tried to improve the situation by utilization of numerical optimization procedures coupled with FEM analysis solver [6]. Optimization task included 183 variables, which defined panel sets thickness. Sets of the panels included panels between the ribs on the upper and lower skin, front wall of the spar, rear wall of the spar and for the particular ribs. The objective was to minimize mass of the wing (1), together with satisfying the constrains on the maximum node displacement (2) and maximum allowed stress (3). The safety factor was set to 1.8. The optimization was done on a rather rare grid, because of the time savings and didn't include the buckling criteria. Although, the optimization task is simple and doesn't include all the real world phenomena it gives initial clues how to design wing's structure of the join wing aircraft.

$$m_{\min} \qquad (1)$$

$$\varepsilon < \varepsilon_{\max} \qquad (2)$$

$$\sigma < \nu \cdot \sigma_{dop} \qquad (3)$$

Results of the optimization are shown on Fig. 6. Stress concentrations almost disappear and the distribution is more uniform. Stress distribution improved, but it is still far away from perfect. The optimization is similar in many points to the topological and topographical optimization [7]. In these kind of optimization algorithms results can be improved by filtering adjacent panel thickness [8] and making panel thickness distribution more smooth. Blending function (4) was introduced to include these feature. To determine the particular panel set thickness, thicknesses of the adjacent panel sets are taken into account by computing weighted average (5). Weights are set on the basis of the (4) equation. Near panel sets have more influence than the sets placed far away from the considered panel.

$$w = 0.5 * (1 - \cos(\theta)) \quad \theta \in \langle 0; 2\pi \rangle \qquad (4)$$

$$th_i = \sum_n \frac{w_j \cdot th_j}{n} \qquad (5)$$

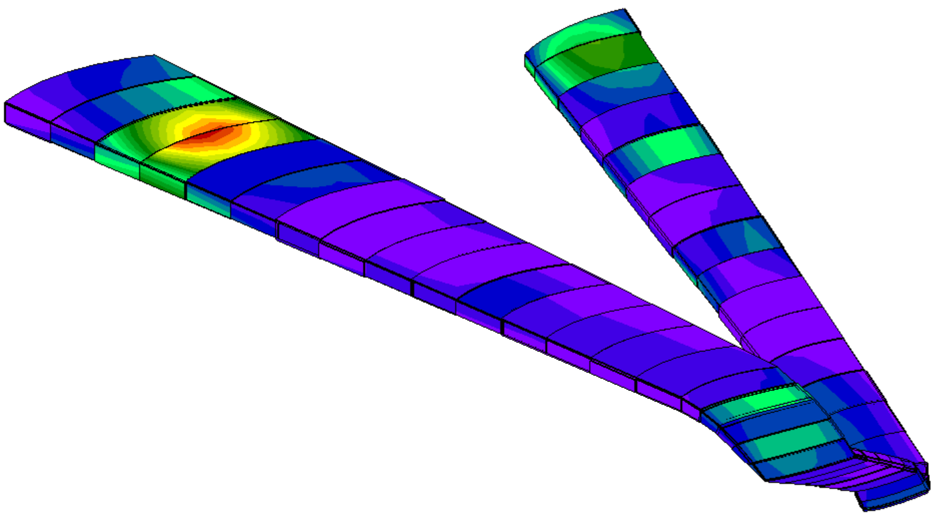

Figure 6: Initial results after optimization.

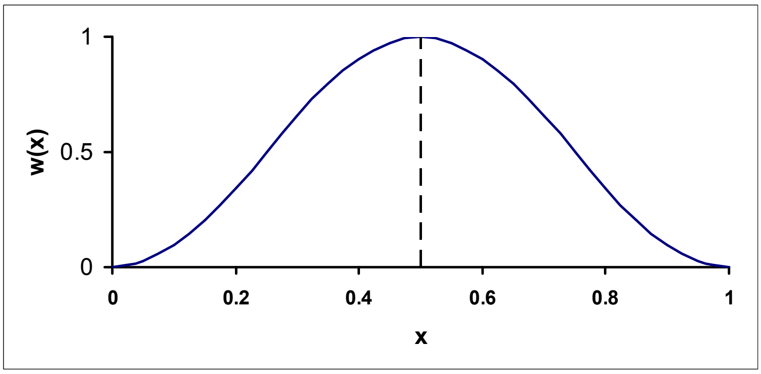

Figure 7: Blending function for panel thickness filtering.

Results obtained from the optimization, after utilizing blending function and filtering are show on Fig. 8. Stress distribution is much more uniform and all the stress concentrations disappear. Uniform stress distribution indicates that the material in the wing is optimally used. All constrains were satisfied, nodes displacement is below the specified and critical stress conditions were not achieved. Panel sets thickness distribution on the wing skin along the wing span is shown on Fig. 9. It is clear that to reduce stress in the wing arcs thickness in that place increased.

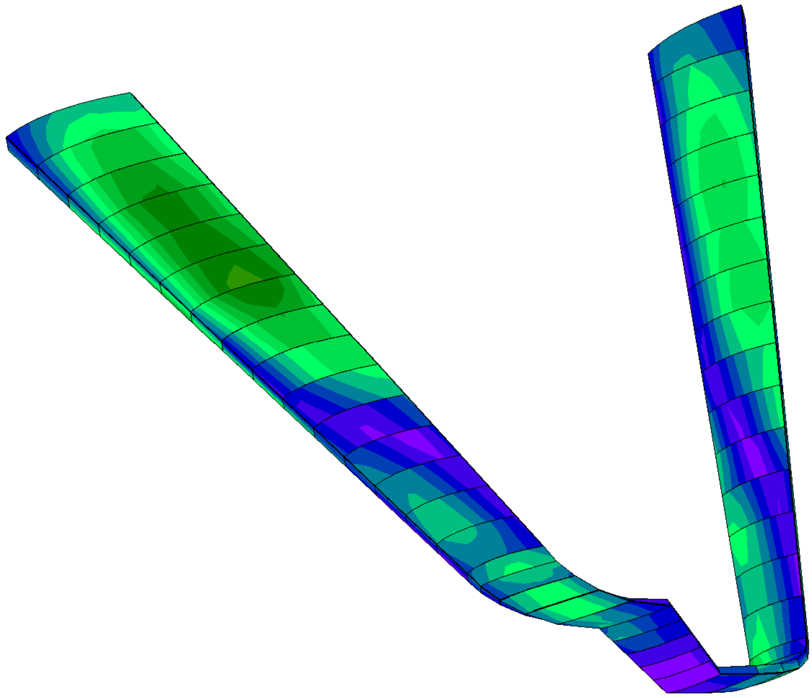

Figure 8: Results of optimization with panel thickness filtering.

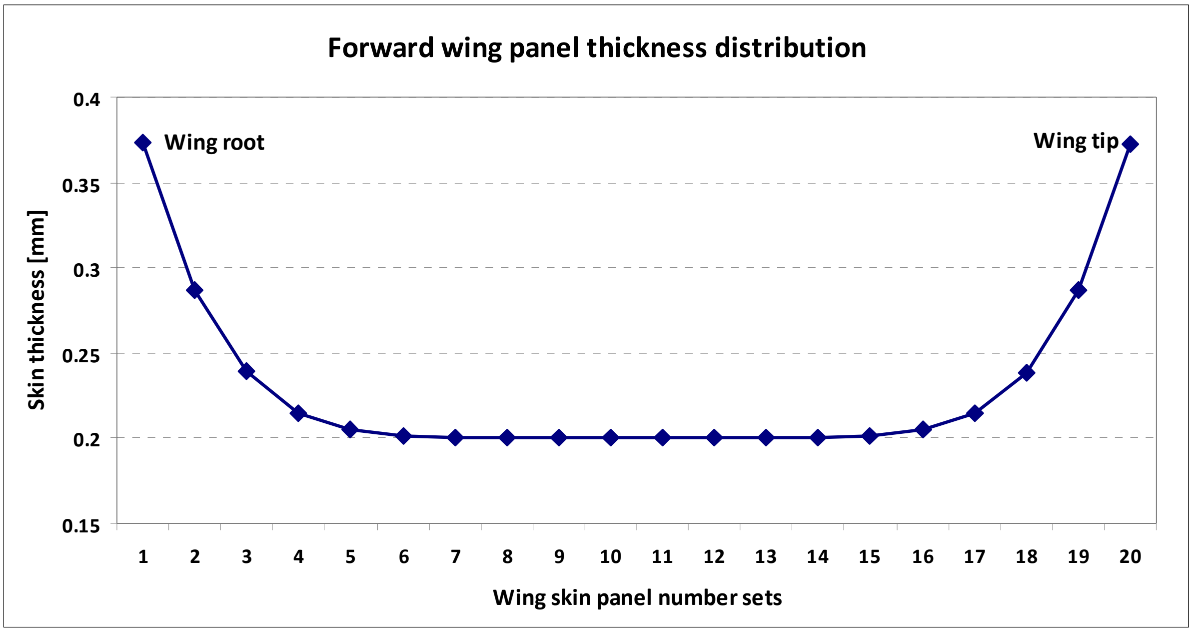

Figure 9: Panel thickness distribution along the wing span.

4 MULTIPLE CASES ANALYSIS

Typically one can simply determine which flight case is the most critical for particular aircraft component. There is mostly one or two sizing load cases per component that are always considered.

In a case of joined wing configuration there is an obstacle, static indeterminacy which can produce unexpected results and causes difficulties in determining the most critical loads. To ensure that structure is sized for entire flight envelope authors decided to perform calculations for the characteristic points, that create the maneuvers load envelope Fig. 10. Preparation of the envelope is performed according to certification specifications document [9].

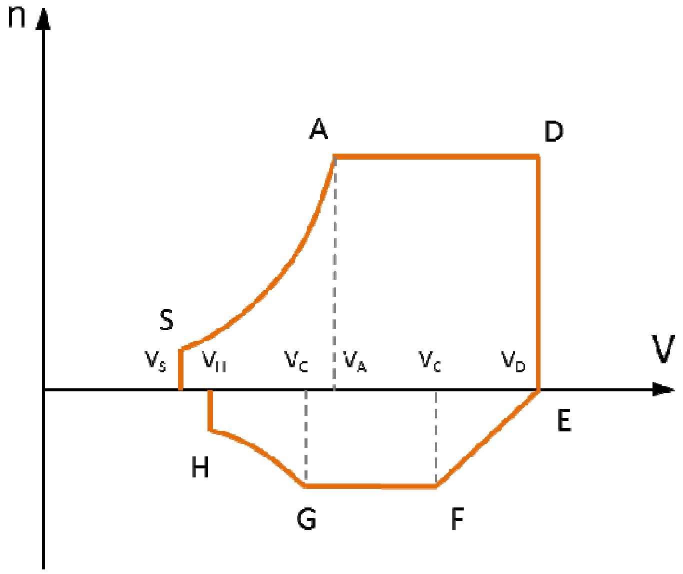

Figure 10: Definition of characteristic points of maneuverable envelope.

Determination of the critical load cases were based on the static von Mises stress values. For each point on the load envelope von Mises stress was computed and critical load cases were found Fig. 11.

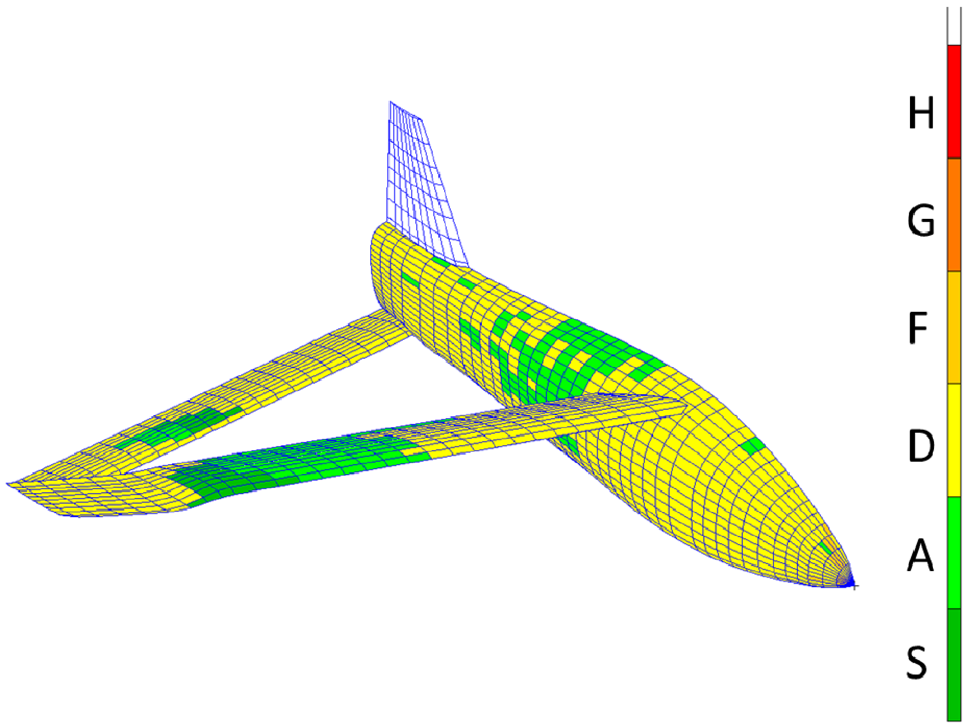

Figure 11: Critical points from the load envelope for different panels.

The study has showed that there are three the most damaging points on the load envelope i.e. points S, A, D. All other flight conditions have negligible influence on the structure strength for the considered joined wing aircraft.

5 BUCKLING ANALYSIS

FEM model was specially prepared for the buckling analysis, for the full aircraft configuration. This was justified by the necessity of checking wing structure behavior with partially fixed supports. The second idea was to check buckling of the fuselage, in the region between wings connection. As global buckling is very unlikely to appear under static loads (it appears usually together with dynamic load in the form of flutter), it was decided to check structure strength for local buckling failures only.
Wing structure was made of complete covers, not only the wingbox. Fuselage was made of bulkheads, stringers and covering panels. Some of the structure components have its analytical idealization, thus solution of the stability equations under simple boundary conditions is well known [10, 11]. Authors decided to consider local buckling analysis of wing covers, fuselage covers and spar webs. These components have almost rectangular shapes and small curvature, so equations for critical loads are available. In the assumed approach components were considered as a simply supported plates (over all edges) with given curvature, loaded by normal and shear stress. Connections with webs were chosen as a buckling area separators Fig. 12. These defines size of the plate where buckling can appear.

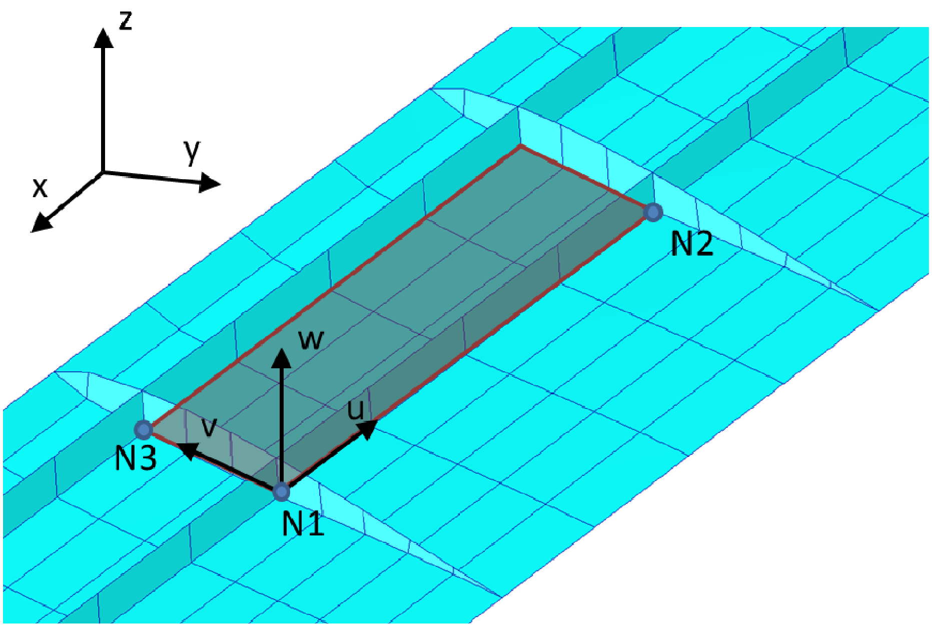

Figure 12: Wing panel idealization.

Stresses obtained from FEM analysis were transformed to local coordinates. Then the reserve factors were calculated in the coordinate frames according to equations (6), (7).

$$RF_s = \frac{(P_{uv})_{critical}}{P_{uv}} \qquad (6)$$

$$RF_n = \frac{(P_u)_{critical}}{P_u} \qquad (7)$$

To consider coupling normal and shear loads following formula was used (8).

$$RF = \frac{1}{\frac{1}{RF_s} + \frac{1}{RF_n}} \qquad (8)$$

Buckling criteria was applied in the optimization algorithm. In each region where applied, the more critical criteria buckling or static stress was the driver of the thickness change Fig. 13, 14. Optimization study was performed for the following design variables: wing and fuselage panels and spar thickness.

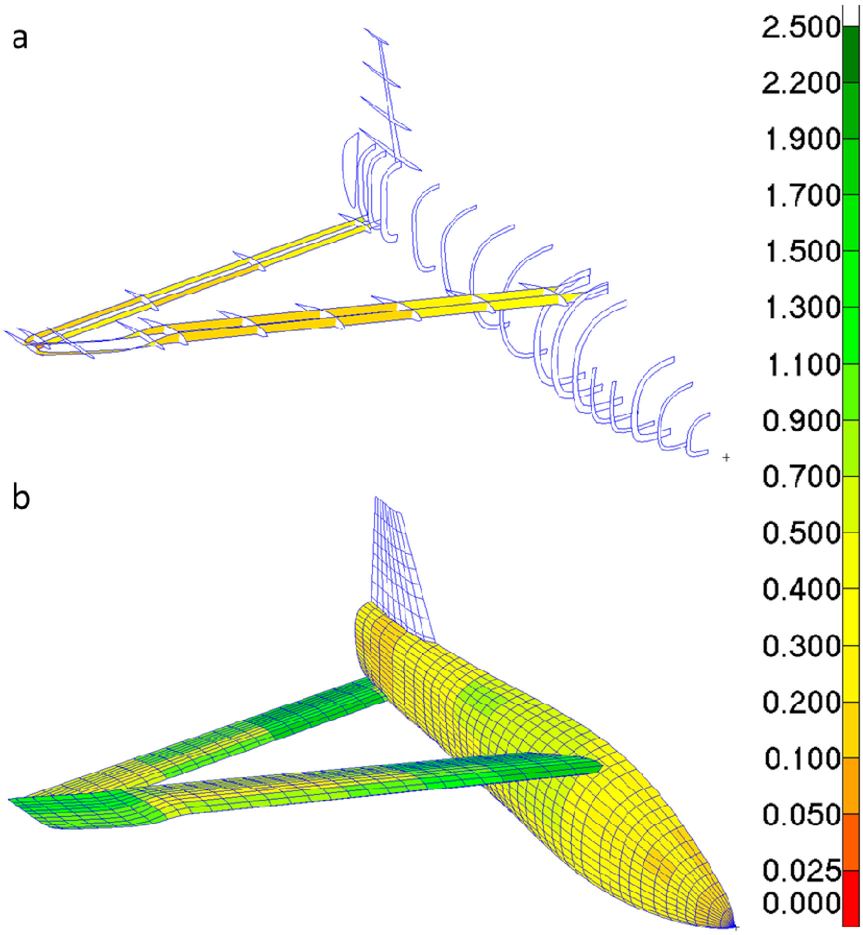

Figure 13: Thickness distribution after optimization with local buckling criteria, a – internal structure, b – external structure.

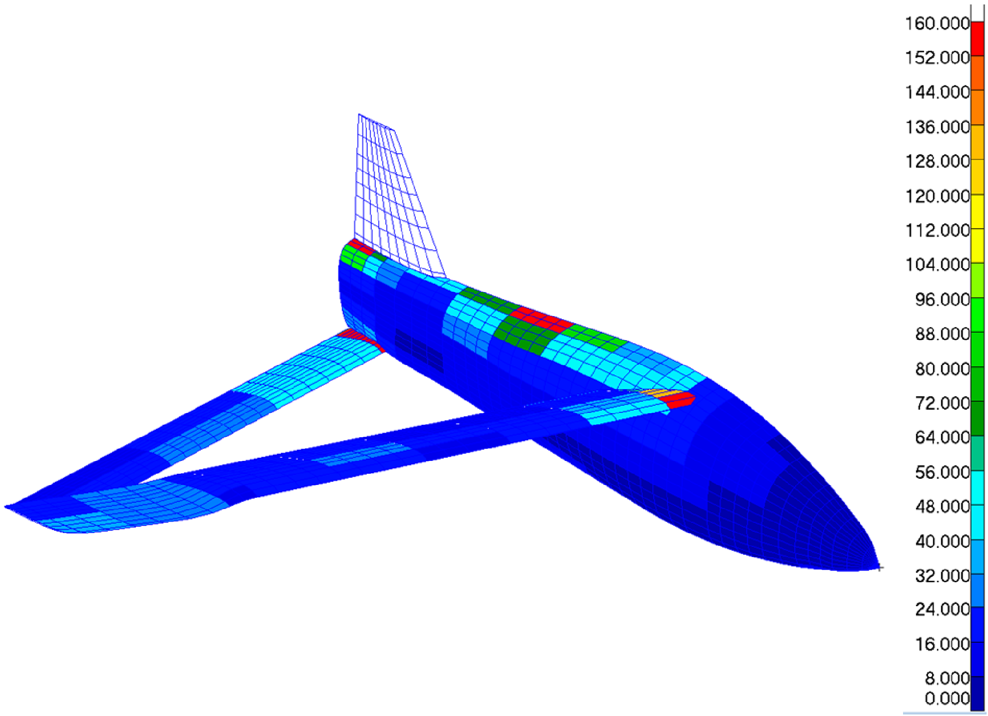

Figure 14: Maximum Stress per panel after optimization with local buckling criteria.

6 DISCUSSION

Initial analysis showed, that the wing has stress concentrations in the area of joining arcs. To solve the problem numerical optimization of panel thickness distribution was done. From the numerical point of view optimization was successful, giving more uniform stress distribution and avoiding stress concentrations, but in a result panel thickness at the arcs also increased. Although globally mass of the wing was decreased in the specific area mass was increased and these should be rather avoided.
There are three possible solutions of optimizing the engineering problem. The first one was "proposed" by the optimization algorithm and it is the current solution, with pros and cons.
The second possibility could be making the structure thinner in the arcs area, which would result in a compliant mechanism, with flexible joints at the arcs. But the optimization procedure didn't choose that way and there could be few reasons. Constrain of maximum displacement of nodes didn't allow to make the structure compliant. The constrain could be relaxed, but then the risk of high wing twist change and flutter will rise. In the middle of the half wing distance minimum panel thickness 0.2mm was reached. This also prevents wing of becoming a compliant mechanism, but even 0.2mm thickness might be not enough for skin buckling (this constrain wasn't included in the optimization).
Finally there is third solution derived from the previous conclusions. Tips of the wings should move freely transferring forces, but not moments, that wouldn't produce additional stress at the arcs. Classical solution would be to put hinges at the place of bending. This possibility wasn't simulated yet and can be considered as a possible future progress of the work.
Separate buckling analyses were done, which showed that local stability criterion is critical for almost whole wing structure and should be considered in structure sizing process. For the preliminary design this can be done with simple idealization of the structure and using analytical models. This significantly increases computational performance and allows optimization of joined wing structure.
The analysis of the structure should be performed for whole flight load envelope. In different structure regions different points from the load envelope will be critical. However the main impact on the structure have loads with positive n factors.

7 CONCLUSIONS

FEM analysis of the joinwing aircraft showed that the structure configuration behaves in a different way than the classical cantilever wing. It introduces new problems to be solved, which were described in detail in the article. Satisfying all structure strength requirements and achieving positive result is a nontrivial task. There are few possible solutions of the problem, but indicating the best one still requires further investigations.

8 LITERATURE

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