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EXPERIENCE OF USING NONLINEAR NUMERICAL ANALYSIS TO ESTIMATE AIRCRAFT PRIMARY STRUCTURE STRENGTH

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Abstract

Structure nonlinear behavior is a common problem in strength analysis. In this paper the simplified methods' drawbacks are shown. To highlight typical methods' limitations, nonlinear finite element method and experimental data were used. Typical reasons for incorrect analysis are shown and analyzed.

1 Introduction

Aircraft structure as a rule behaves nonlinearly when the load approaches ultimate level. Geometrical nonlinearity is related to large displacements and structure elements' buckling. Physical nonlinearity is caused by structural material nonlinear strain-stress dependence. Structural nonlinearity appears from changes in a structure topological connectivity (contact interaction, structure elements failure). The most common nonlinearity caused by structure elements' local bucking. Obviously, nonlinear structure behavior must be taken into account in static strength analysis methods.

There are a lot of methods for analysis. Analytical methods as well as numerical ones are used for structure nonlinear analysis.

Analytical models in which local buckling is taken into account by using reduction coefficients, that reduce the stiffness of buckled structure elements [1, 2, 3] are widely used and gives good results. Application of such analytical models is limited to consideration of relatively simple geometric shapes and associated with certain assumptions, which in some cases can lead to incorrect results.

The nonlinear finite-element (FE) method of analysis [4, 5] is widely used to estimate structural strength. This method allows studying postbuckling stress strain state in a more precise formulation without using the simplifications typical for the analytical models and obtaining highly confident structure strength estimation (that is experimentally proved). However, correct application of the method requires special approach to FE-modeling, solution methods and obtained results' analysis. [6, 7]

The nonlinear analysis method can be used to verify analytical models to estimate the feasible areas of their application. Several examples are presented in this paper.

2 Skin-stringer deformation interaction

Skin-stringer deformation interaction can occur in stringer stiffened panels and affect on its strength. The essence of this fact is that the buckled skin cause local stringer bending, as a result, stringer is no longer in a flat state. Most of analytical methods do not take it into account, as a result in certain cases it may lead to a significant panel failure load level overestimation. It is shown on the following model example.

2.1 Model example

Panel parameters are shown in Fig. 1. In this example stringer's relative height is sufficiently large in order to represent deformation interaction to a maximum extent. Nonlinear analysis shows that local skin buckling occurs under compression load 5300N (Fig. 2a), global panel buckling load level is 22750N (Fig. 2b).

Stress-strain state analysis shows that the highest stress level is in the stringer-skin interface zone and it is close to the material failure stress level (Fig. 2c).

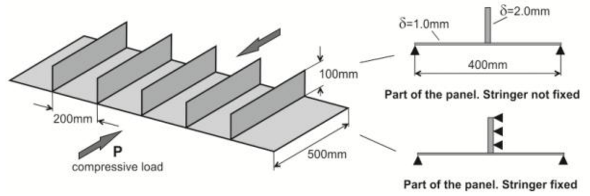


Fig. 1 - Panel parametrs

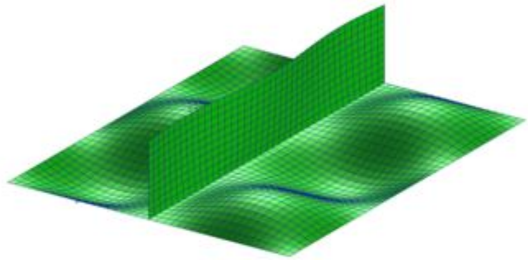


Fig. 2a - Local skin buckling P=5300N

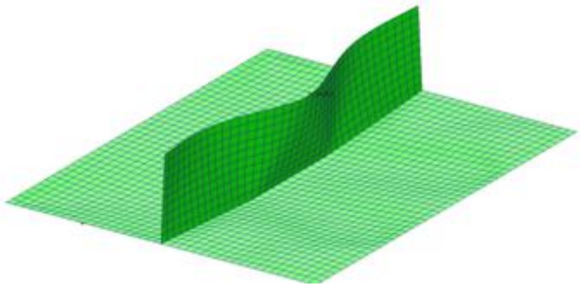


Fig. 2b - Global buckling P=22750N

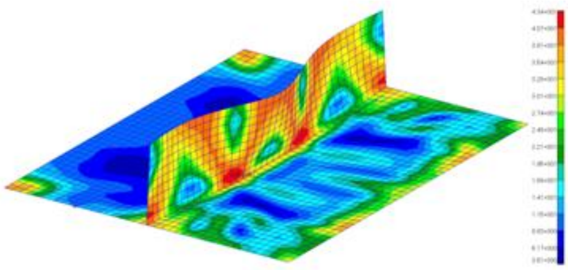


Fig. 2c - Von Mises stresses P=22750N

Dependence of plane skin and stringer deflections on compression load is shown in Fig. 3.

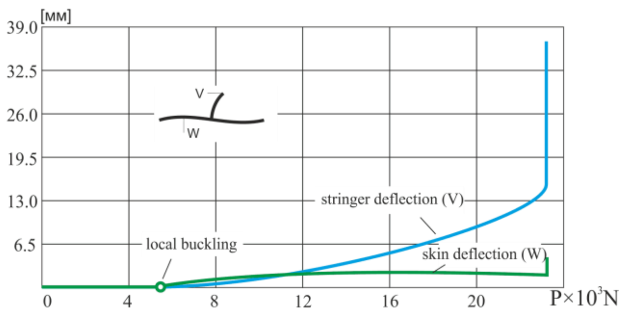


Fig. 3 - Skin and stringer deflection

There is an intensive growth of stringer's out of plane deflection after skin buckling. Skin deflection varies slightly and even tends to decrease due to flexural center shifting. The panel bends preventing from skin deflection increment.

Let us assume that there is no skin-stringer deformation interaction and the stringer remains flat after skin buckling, as in most analytical models. In the model under consideration the stringer was fixed to prevent its displacements out of the plane. Results are presented in Fig. 4 a,b,c.

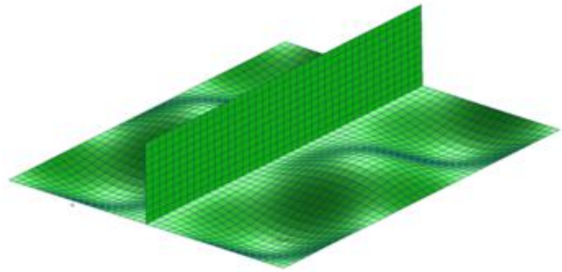


Fig. 4a - Local skin buckling P=5300N

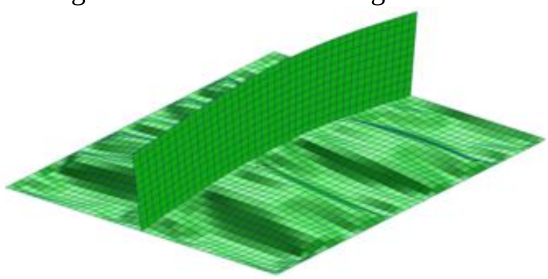


Fig. 4b - Global buckling P=86300N

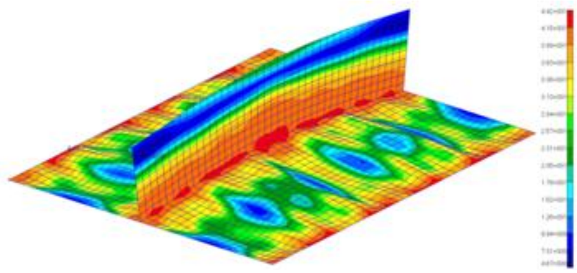


Fig. 4c - Von Mises stresses P=79930N

Material stress limit is reached at load level 79930N in the stringer-skin interface zone. Maximum stress-load graph is shown in Fig. 5. In this case only skin buckles under the same compression load (5300N). Global panel buckling takes place when load reaches 86300N.

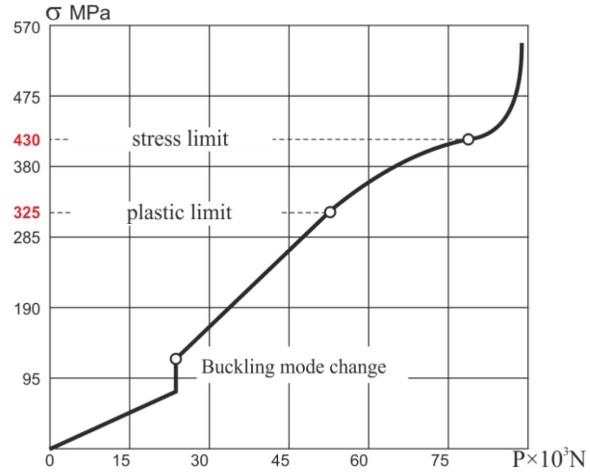


Fig. 5 - Maximum stress-load dependece

These results show that using analytical models in which stringer-skin deformation interaction is not taken into account, can lead to significant panel strength overestimation. In the present case, ignoring the stringer-skin deformation interaction led to a 3-fold failure load overestimation. As a result, more detailed technique that takes into account stringer stiffness decrease should be used.

2.2 Real structure example

The above example is not typical in terms of real load-bearing panels design. It only shows the significance of stringer-skin deformation interaction effect.

Let us consider a real fuselage panel with a Z-type stringer. Panel geometry is shown in Fig. 6.

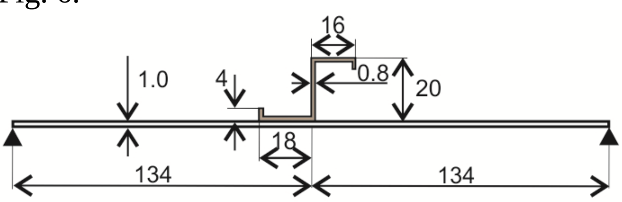


Fig. 6 - Fuselage panel geometry

In this case the stringer-skin deformation interaction causes an 8% decrease in panel load capacity. In Fig. 7 load bearing capacity for compress-shear combination for both analytical and nonlinear approaches are shown. These results are experimentally approved with static airplane test results.

Experimental failure load level is less than that obtained using engineering approaches and in a good agreement with the FE method. The deformed state of the model and real fuselage panels' global buckling are shown in Fig. 8.

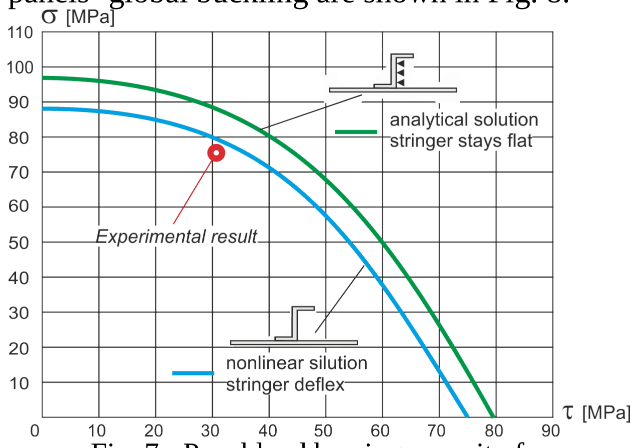


Fig. 7 - Panel load bearing capacity for compress-shear combination

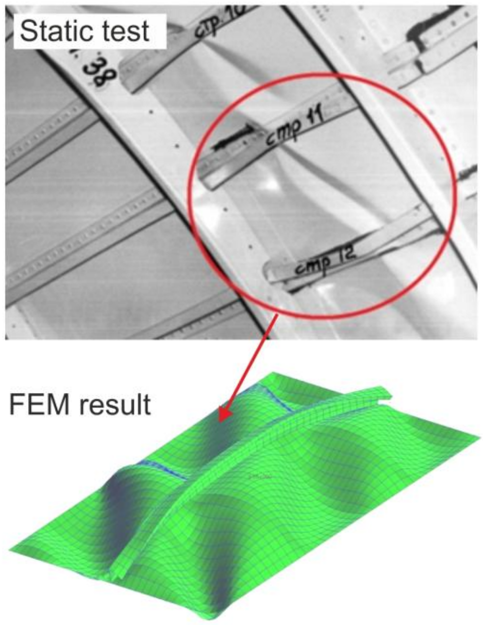


Fig. 8 – Static test and FE model results

In some cases when failure occurs before global buckling it is difficult to determine the initial reason for failure process start. In such cases nonlinear FE analysis can be a useful tool. In the following example, stringer tears off skin in its bottom area under load less than critical global buckling load level (Fig. 9). This failure mode is typical for heavy panels, such as wing stringer panels.

Fig. 9 – Wing stringer panel static test result

The nonlinear method made it possible to find the reason for such behavior. Numerical analysis showed that stringer local buckling takes place before the failure. As a result, in stringer bottom area extra stresses appear and in combination with axial stresses it causes tearing. Strain gauges data, obtained in tests, prove this assumption. On the basis of the obtained results panel cross section was modified to eliminate local buckling without overweighting.

3. Linear buckling method limitations

Linear buckling solution method is used quite often, but its application range is limited with the initial assumptions. The initial assumptions are well known: the deflections must be small, stresses must be elastic and force distribution due to the applied loads must remain constant. This hardly meets the requirements of thin-shelled aircraft structures load bearing capacity analysis.

3.1 Example

There is an example when linear buckling solution can give wrong results. In this case a problem of determination of minimum required rib stiffness for efficient wing panels operation is considered. As an elementary model let us consider a beam with three fulcrums with the middle one being elastic (see Fig. 10). The beam represents a panel and the elastic fulcrum represents a rib. For efficient operation, the rib must be stiffened enough to eliminate the first buckling mode.

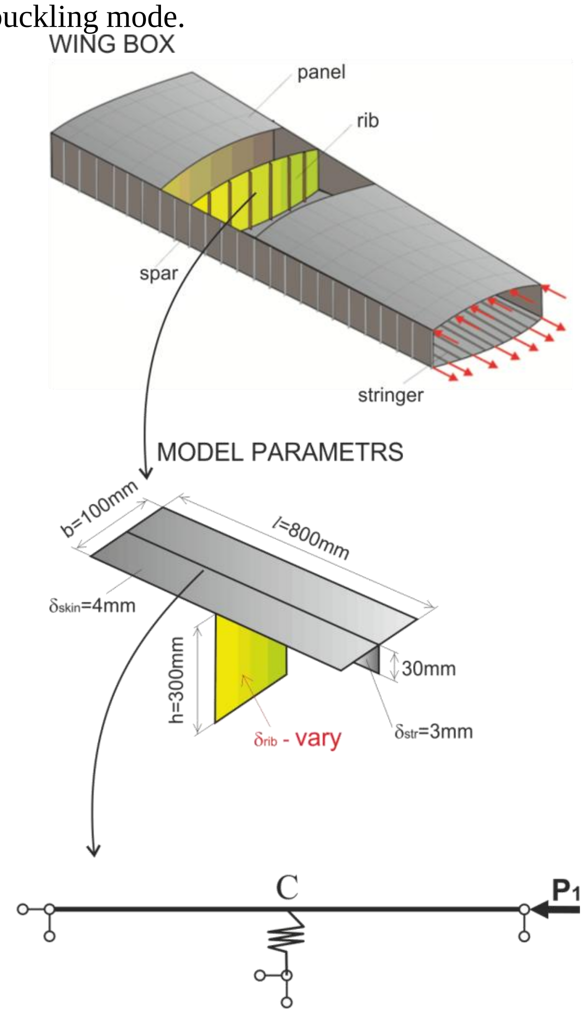


Fig. 10 – Elementary model

There is an analytical solution for this problem based on linear buckling method. The dependence of critical buckling load level on fulcrum stiffness C (rib thickness) is shown in Fig. 11a.

Linear numerical (FE) and analytical results [8] are in a good agreement. Graph breakpoint determinates the minimum rib thickness, where buckling mode change occurs. It is worth noting that the minimum required rib thickness is unnaturally small (0.017 mm).

The result of nonlinear numerical (FE) analysis (Fig. 10b) is considerably different. For the same beam, the minimum rib thickness is much larger (1.8mm).

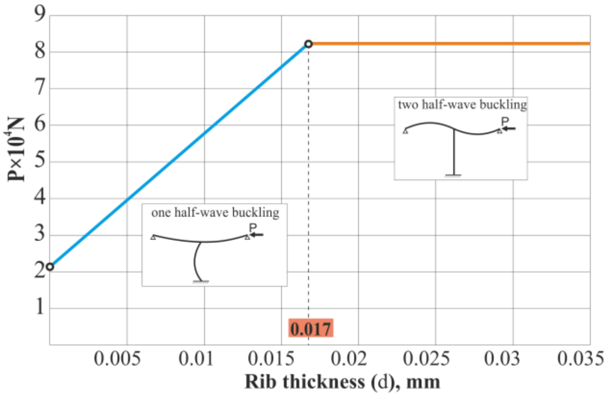


Fig. 11a - Linear solution result

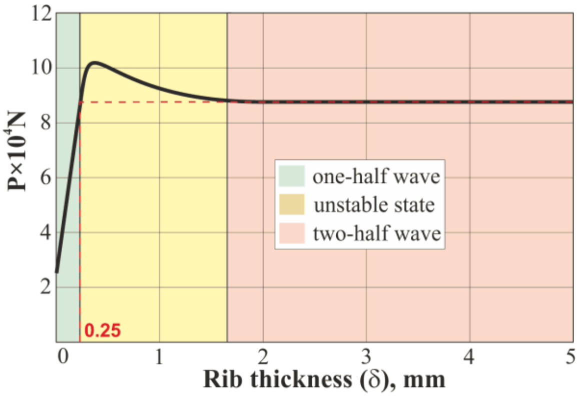


Fig. 11b - Nonlinear solution result

There is a region where critical buckling load exceeds the critical second mode load. This is due to bending moment appearing in the rib as a result of joint point's displacement under loading. When critical level is reached, both the panel and the rib bend as shown in Fig. 12.

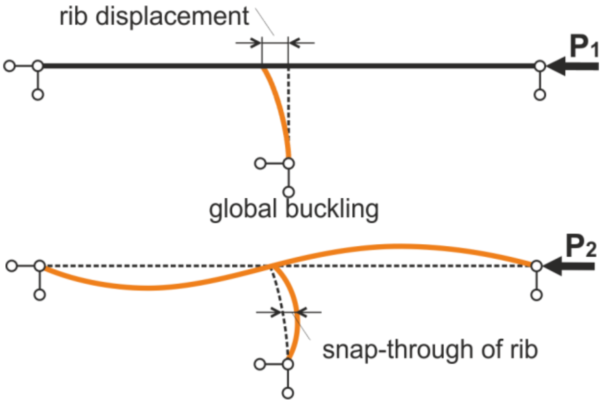


Fig. 12 – Snap-through of rib

Similar results can be obtained with a more complex model. Comparison of linear and nonlinear solution results for a three-dimensional shell torque box (Fig. 13) is shown in Fig. 14 a, b. The intersection point of critical buckling load levels for the panel and the rib determines the minimum rib thickness. Linear buckling analysis gives similar unrealistic rib thickness estimation as in the previous model. Dependence of critical buckling load on rib thickness in nonlinear solution is similar to that obtained with the elementary model.

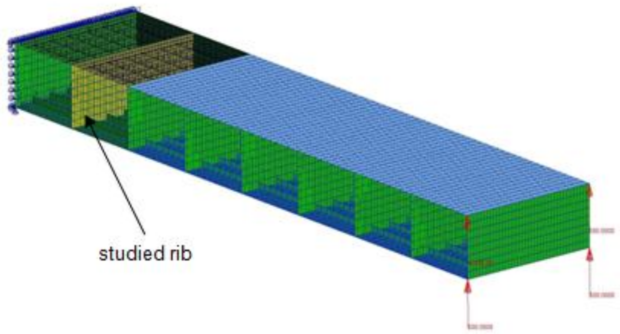


Fig. 13 – Wing box FE model

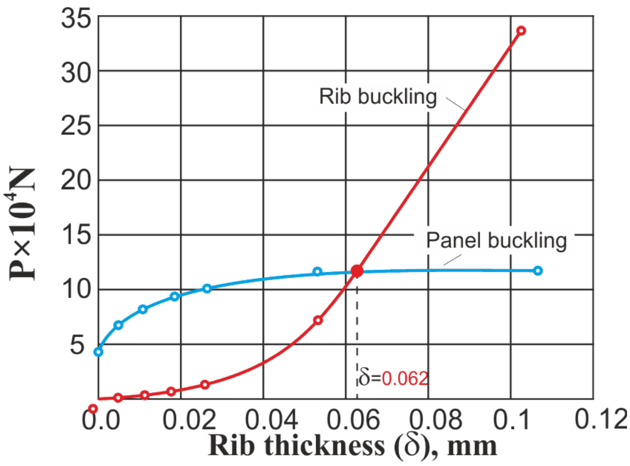


Fig. 14a – Linear solution

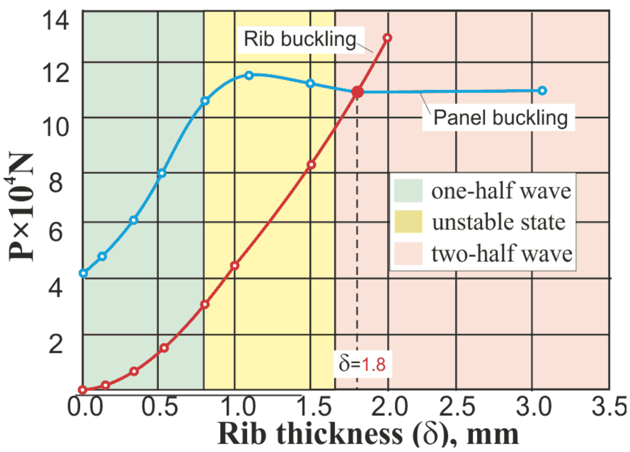
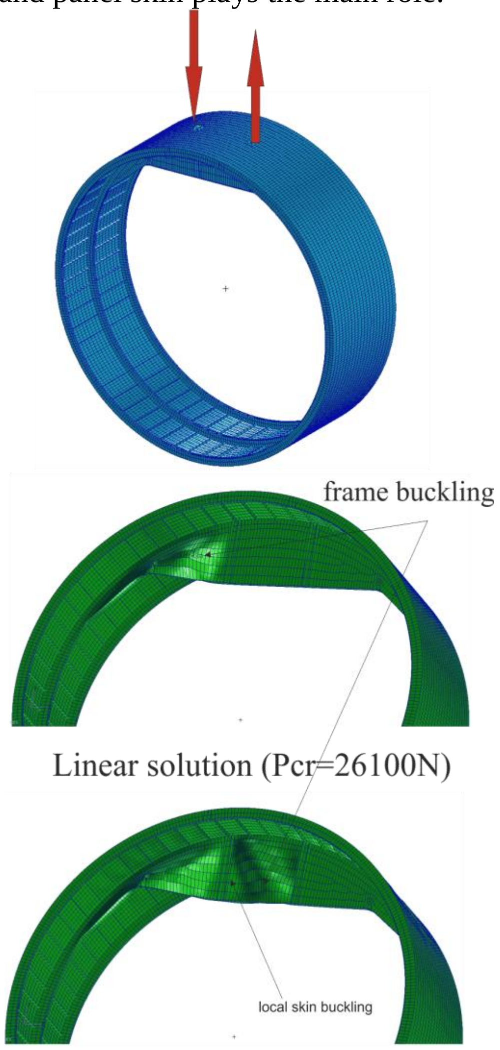


Fig. 14b – Nonlinear solution

Both examples show that linear buckling approach does not give correct minimum rib stiffness estimation. The reason for this is that the structure geometry changes under load and it is not taken into account. As a result, in case when the rib stiffness is low, linear solution can lead to overestimation of panels' critical buckling load.

The same buckling load overestimation can be obtained for fuselage frame as well. In Fig. 15 analysis results for fuselage frame loaded with forces from vertical tail are shown. Linear solution results in P_b=26100N. The nonlinear approach shows that the initial frame buckling takes place under load P=16670N. Obviously, the frame failures under a higher load level when the material stress limit is reached. Nevertheless, the obtained load is significantly lower. It is worth noting that in this case the early appeared buckling of the frame web and panel skin plays the main role.



Nonlinear solution (Pcr=16670N)

Fig. 15 – Fuselage frame analysis results

4 Conclusion

The experience of using nonlinear numerical analysis for airframe elements load-bearing capacity estimation is presented in this paper. The main advantages of the nonlinear numerical analysis method are shown. Examples in this paper prove that taking into account nonlinear processes affects significantly the analysis results.

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