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STRUCTURAL ANALYSES OF ORTHOGRID FUSELAGE PANEL FOR INTEGRATED KU-BAND SATCOM ANTENNA – EMUS 2019

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Abstract. The aim of this work is to describe the structural analysis of a multifunctional aircraft fuselage panel. The structure of the panel has an embedded antenna tiles. The panel consists of UniDirectional (UD) carbon fibre reinforced composite skin stiffened with ortho-grid ribs, and a transparent skin window made using UD glass fibre reinforced composite. The ortho-grid structure is a structural reinforcement but also the antenna tiles support. The presented work proposes a numerical multiscale strategy. The laminate is simulated with solid elements, in order to capture the real kinematics of the material, but several laminas are condensed in a single finite element. The performance of each lamina is obtained using the Serial-Parallel (SP) mixing theory. The specific formulations developed have been very useful to identify and study the mechanical performance of these new structures and the localization of unknown and un-predicted hot-spots in the structure.

1 INTRODUCTION

The Fokker 100 is the target aircraft to integrate the ortho-grid stiffness fuselage panel with the integrated antenna. The forward crown panel at the top of the fuselage is selected for the ortho-grid panel location, as in this section the antenna will have the best performance with the lower loads. It should be noted that in this work no aerodynamic loads are included, and that a variation of temperature conditions is not foreseen.

The panel is manufactured using thermoset prepreg materials, which are suitable for aerospace structural applications. The panel skin and the ortho-grid ribs consist of carbon fibre reinforced material: Hexcel 8552 resin and AS4 fibre. On the other hand, the transparent panel window skin consists of glass fibre reinforced material: Cytec FM906-27 resin and S2 glass 187-460 fibre.The ortho-grid ribs are laid-up using automated fibre placement over the panel skin. The panel skin has a material transition in the glass window edges through interleave plies of carbon and glass. Finally, the whole fuselage panel is cured in an autoclave.

Even with nowadays computational capacity it is almost impossible to analyse a structural component taking into account a detailed definition of the laminate layup. For this reason, in this work a numerical multiscale strategy is proposed. The laminate is simulated with solid elements, in order to capture the real kinematics of the material, but several laminas are condensed in a single finite element. The performance of each lamina is obtained using the serial-parallel mixing theory.

2 SERIAL-PARALLEL MIXING THEORY

The serial-parallel mixing theory could be defined as a phenomenological homogenization, where the behavior of the composite is obtained from the constitutive response of their materials components. This theory has been developed by Rastellini et al. [1], and is a natural evolution of the parallel mixing theory developed by Car et al. [2]. The theory is based on the compatibility conditions, but introduces a modification in the iso-strain hypothesis. The iso-strain condition is imposed in the reinforcement direction and a new iso-stress condition is imposed in the transversal directions. Commonly, for UD composites, the formulation applies an iso-strain condition in fibre direction and a iso-stress condition in the other ones.

Taking only two composite components, the equations that define the stress (σ) equilibrium and setting up the strain (ε) compatibility between the individual components follow the hypothesis previously described are:

Parallel behavior:

$$\begin{aligned} {}^{c}\varepsilon_p &= {}^{m}\varepsilon_p = {}^{f}\varepsilon_p \\ {}^{c}\sigma_p &= {}^{m}k\,{}^{m}\sigma_p + {}^{f}k\,{}^{f}\sigma_p \end{aligned} \tag{1}$$

Serial behavior:

$$\begin{aligned} {}^{c}\varepsilon_s &= {}^{m}k\,{}^{m}\varepsilon_s + {}^{f}k\,{}^{f}\varepsilon_s \\ {}^{c}\sigma_s &= {}^{m}\sigma_s = {}^{f}\sigma_s \end{aligned} \tag{2}$$

where the superscripts c, m and f stand for composite, matrix and fibre, respectively and ${}^{i}k$ is the volume-fraction coefficient of each constituent in the composite.

This theory can predict the linear and non-linear behavior of structural elements made of composite materials. Composite materials that can be modelled are those formed of long fibres embedded in a matrix. The theory predicts the different behavior of the composite, depending on the load direction. The potential of this theory is to predict accurately the response of composites in the linear and non linear range as has been proved in several papers [3, 4]. Among the different failure modes that can be captured, in [5] it is shown that the serial-parallel mixing theory is able to simulate the delamination problem naturally, without having to define specific elements or predefine the path of fracture.

3 NUMERICAL MODEL DESCRIPTION

Finite element numerical models are used to study the structural response of the panel for different loading cases. To reduce the computation time required for the simulations, and to be able to include all details in the models developed, only a reduced section of the fuselage

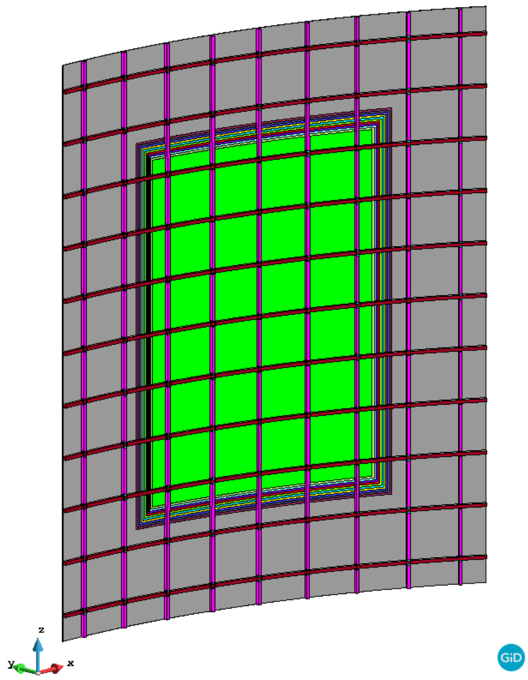

Figure 1: Reduce numerical model for the ortho-grid fuselage panel.

is analysed. In order to obtain the same results that would be reached with the whole panel consistent boundary conditions are imposed.

In the Figure 1, the central green window represents the UD glass fibre reinforced composite transparent window skin. While the grey part of the skin uses UD carbon fibre prepreg material. Figure 2 shows in detail the different geometries in which every part of the model is discretized to model the connection between skin/ribs, rib/rib and the glass/carbon skin transition.

To represent the transition regions, it is necessary to define models with a high level of detail in order to account for changes in layer orientation, composite intersections (glass to carbon), and all different specificities shown in Figure 3. Consequently the quantity of elements required by the model is high, which increases the computational cost of the numerical analysis, in terms of time and memory. In order to improve the machine time and simplify the numerical model a numerical multiscale approach is proposed.

In the numerical model, the laminate is simulated with solid finite elements, to capture the real kinematics of the material, but several layers are condensed in a single finite element. The performance of each lamina is obtained using the SP theory. On the right side of the Figure 3 the homogenized transition zone is shown. As can be seen, there are three element through the thickness, and each one of these elements contains internally several layers. The upper part of the figure shows the numerical model developed while in the lower part are represented the different layers assigned to each region of the model.

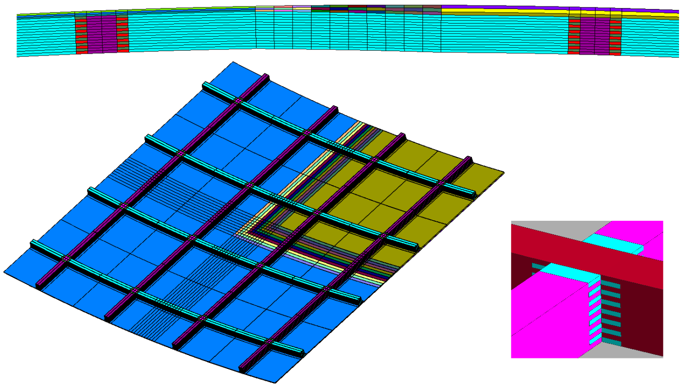

Figure 2: Detailed view of the glass/carbon skin transition, and rib connections.

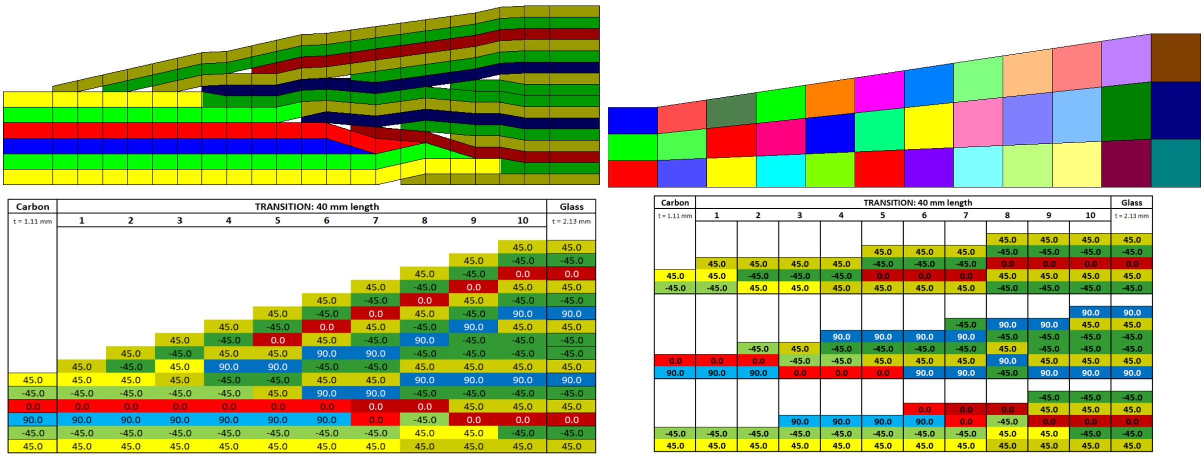

Figure 3: Detailed view of the glass/carbon and homogenized model skin transition.

4 RESULTS

In the following it is presented the numerical results obtained for a tension load case. A tension flow (261.5 N/mm) in the top side and a fixed support in the bottom side of the panel is imposed. It is also applied an the internal pressure (0.08 N/mm^2) of the fuselage. Therefore, specific boundary conditions have to be imposed on both lateral side of the panel.

The total displacement field obtained is shown in Figure 4. The left side shows a inside view of the displacements obtained and the right side a external view of the panel. The maximum values of displacements are presented in the top of the panel because of the tension load, and they are extended to the middle of the panel for the effect of the internal pressure. In Table 1 the most relevant displacement results are presented.

The longitudinal ribs of the panel take most of the load applied. A mean stress value of around 320 MPa is presented in the longitudinal ribs. The maximum value of the principal stress obtained is located on the longitudinal rib close to the Glass/Carbon skin transition in the rib connection. The rib connections have half of the UD carbon plies cut to allow the continuity

Table 1: Relevant displacement results obtained

Max. Long. Displ. (Z dir.) [mm]	2.35
Max. Radial Displ. [mm]	1.85
Mean Radial Displ. [mm]	1.1
Max. Relative Radial Displ. [mm]	0.75

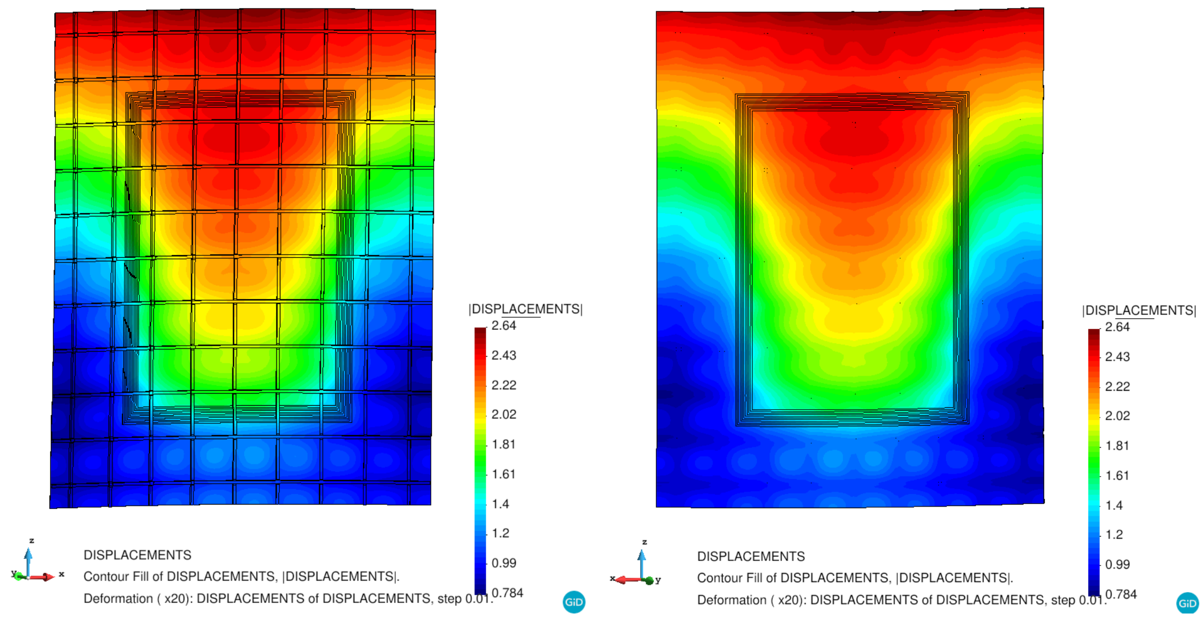

Figure 4: Total displacement field obtained for the tension load case.

of the plies in the perpendicular direction. The maximum stress obtained is around 700 MPa and it is found in the continue layers of the rib connection. The left side of Figure 5 shows the critical zone regarding the stresses level. Another critical area is the region in the connection where the cut layers end, as this region contains only matrix and the stresses are close to 60 MPa. This stress level is close to the maximum ultimate strength of the matrix component.

The skin and the material transition zones do not reach the maximum design strain value (i.e. 3600 $\mu\%$). However, in the rib connections and the rib/skin connection the principal strain field presents higher values. An amplified view of the rib connection is shown in the right side of Figure 5. For the additive material in the ribs connection, a higher design value of maximum strain could be accepted. The mentioned figure shows a strain value in this material around 6000 $\mu\%$.

5 CONCLUSIONS

A multiscale strategy that combines the finite element method and the SP formulation is proposed. The numerical approach presented allows the analysis of complex composite struc-

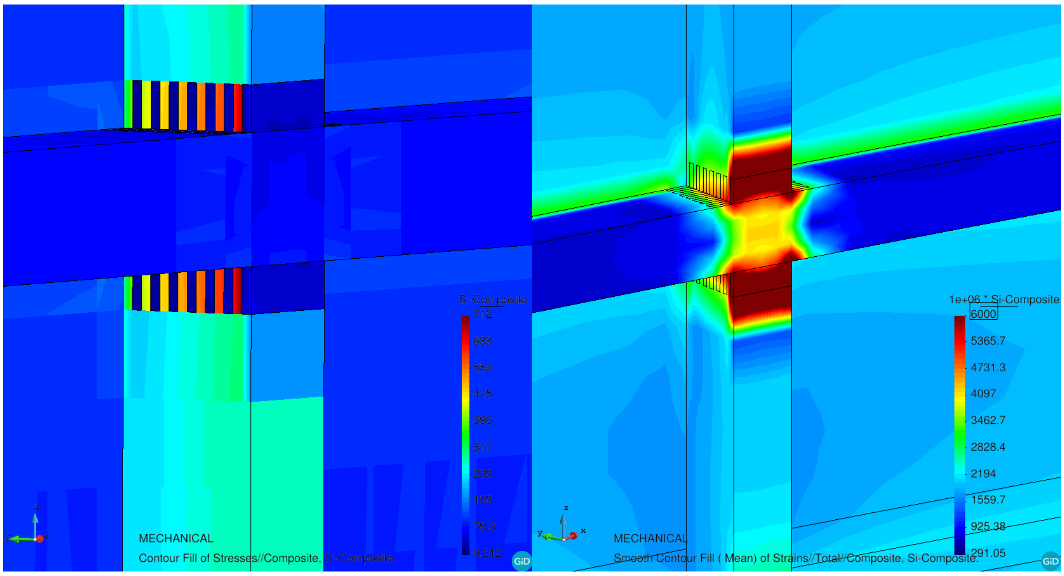

Figure 5: Zone with the maximum value of the principal stresses in the fuselage panel.

tures with an affordable computational cost, which is basic for the development of new multifunctional composite structures. The formulation developed has been very useful to study the mechanical performance of the ortho-gid fuselage panel, and to identify the localization of unknown and un-predicted hot-spots in the structure.

REFERENCES

[1] Rastellini F. and Oller S. and Salomón O. and Oñate, E. Composite materials non-linear modelling for long fibre-reinforced laminates continuum basis, computational aspect and validations. *Computers and Structures.* (2008) **86**:879-896.

[2] Car E. and Oller S. and Oñate, E. An anisotropic elastoplastic constitutive model for large strain analysis of fiber reinforced composite materials. *Computer Methods in Applied Mechanics and Engineering.* (2000) **185**:245-277.

[3] Martinez X. and Oller S. Numerical simulation of matrix reinforced composite materials subjected to compression loads. *Archives of computational methods in engineering.* (2009) **16**:357-397.

[4] Otero F. and Oller S. and Martínez X. and Salomón O. Numerical homogenization for composite materials analysis. Comparison with other micro mechanical formulations. *Composite Structures.* (2015) **122**:405-416.

[5] Martinez X. and Oller S. and Barbero E. Study of Delamination in Composites by Using the Serial/Parallel Mixing Theory and a Damage Formulation. *In: Mechanical Response of Composites. Computational Methods in Applied Sciences*, vol 10. Springer, Dordrecht (2008)