Stiffened plates and panels are very often used in aeronautical structures. Ships and offshore structures are also built up of stiffened shells and plates. Stiffened panels efficiently provide increased buckling stability than non-stiffened plates. Due to their high strength-to-weight and stiffness-to-weight ratios, composite symmetric laminates are used in stiffened composite panels for both skin and stiffeners, eventually combined with metallic layers (like in the FML concept). The design of composite stiffened panels presents a number of calculus challenges, due to the additional failure modes and anisotropy effects introduced by composite materials. This paper presents some results obtained from a finite element investigation upon the stability of stiffened composite panels in comparison with available experimental results. Finite element analysis provides the highest modelling fidelity, as it allows incorporating many local details, imperfections and various boundary conditions. Numerical simulations can lead to important time economy and resources consumed by experimental tests.

Keywords: FEM, Composites, Stiffened Panels, Buckling

1. INTRODUCTION

The use of composite materials in aircraft structures is dating back in military aviation in early ’60s, civil aviation being generally more conservative, due to passenger security reasons. Nevertheless, the advantages of this class of advanced materials lead in recent years to their use for different parts of civil aircrafts, mainly secondary structures. For aeronautical structures, the weight criterion is essential. Accordingly, these materials successfully replace metallic materials due to their high strength/weight and stiffness/weight ratios. Special applications of composite materials are encountered also in the spatial industry, due to the high velocities and high temperature conditions. Composite structures have also better corrosion resistance and better performances concerning fatigue behaviour and crack propagation with respect to metallic ones. Furthermore, composite structures are simpler than metallic ones, as parts number and riveted connections can be eliminated. Advanced control methods and in service structural integrity assessment are also possible with adequate techniques.

Recently, the Boeing Company, together with other partners, work to obtain a composite material fuselage for the future aircraft Boeing 7E7 Dreamliner, [1]. The European competitor, Airbus, is designing the A 750, with an important percentage of primary structure made of composites. The problems related to such structures concern the material architecture, technological aspects, computer modelling, strength analysis and then continuous in service assessment of the structure by appropriate maintenance procedures. Basically, it is intended to obtain light components, strength and damage tolerant under impact and other type of loads. In such conditions, the composite materials are qualifying for important (primary) parts of aircrafts like fuselage, wings and empennages. By achieving airframe weight reduction, air transport companies will lower fuel consumption, decrease the environmental impact and overall maintenance cost. New materials, such as new Aluminium alloys (Al Lithium), Glare, CFRP, allow more flexibility and optimisation in design. The fibre metal laminates (FML) are hybrid materials consisting of alternate metal and fibre prepreg layers. The Glare family is a recent candidate of FML consisting of alternating layers of metal sheets (aluminium 2024-T3 alloy) and glass fibre reinforced prepreg layers. This material combines the properties of both metal and the fibre reinforced plastics and has also good fatigue properties. The metal
gives the possibility to plastically deform the FML, and the fibres give the strength. The design of composite stiffened panels present additional challenges due to the specific failure modes and anisotropy effects introduced by composite materials. The huge grow of air traffic in recent years, which is expected to continue during the next decades, leads to the need of very large aircraft, to allow the reduction of the total absolute number of flights. This new generation of aircraft referred to as ultra-high capacity aircraft (UHCA) require very large sub-structures. In this case the average skin thickness and panel dimensions will be very large compared to conventional aircraft. For this reason, nowadays, the stiffened panels represent an actual subject for theoretical and experimental research. The buckling strength of such a panel is vital for the overall behaviour of the structure.

However, Aluminium remains today the most common material used for aircraft construction. Many parts are made of sheets having a thickness of about one millimetre. Metallic stiffened plates and panels are also used in ships and offshore structures [2]. The modelling stage used to obtain different types of numerical simulations of static, dynamic and fatigue behaviour is also an important component of the design process for isotropic or composite stiffened panels. Numerical simulations can lead to important time economy and resources consumed during experimental tests. Several analysis methods like simple physical models, smeared models, finite strip models and finite element models can be used. Among all, finite element analysis provides the highest fidelity, as it allows incorporating many local details, such as stiffener configurations, cut-outs, as well as various boundary conditions.

2. STIFFENED PANEL DESCRIPTION

A panel is generally a rectangular flat or curved region between lateral frames (ribs or transverse girders) and longitudinal members (see fig.1). There are a multitude of shapes for stiffeners, some of these being shown in fig. 2. The stiffeners increase the stability of the skin and panel buckling loads, but also the weight of the panel. They are usually fabricated from extruded aluminium profiles. Linear buckling loads and mode shapes for a given panel configuration will be studied. A configuration is defined by width and thickness of the stiffeners and the stiffener spacing. The panel is considered usually with simply supported edges. The linear buckling load provide a measure of the compressive load carrying capacity of the stiffened panels but prior to failure the panels present substantial nonlinear transverse deformations.

The present study deals with a composite stiffened panel configuration analysed in [3] with the finite element code STAGS and by experiment. The skin of the stiffened panel consists in a FML Glare type material, with both stringers and girders of Aluminium alloy and Z-shaped cross-section. This configuration was designed at Delft University using a modification of Kuhn’s method developed for stiffened panels from isotropic materials. In our work the ANSYS finite element method is used to study the linear buckling and the nonlinear one.
3. THE FINITE ELEMENT MODEL

The test panel is of square form of dimensions of 1 m × 1 m. It has six equally spaced stiffeners and two girders (frames) as in figure 3. The skin has the total thickness of 3 mm. It is from Glare 3-5/4-0.4 and has 5 layers of Aluminium alloy and 8 layers of glass fibres. Each Aluminium layer has a thickness of 0.4 mm and each glass fibre reinforced layer has the thickness 0.125 mm and fibre orientation at 0° and 90° resulting 13 layers disposed symmetrically with respect to the mid-plane, like in the figure 4.

![Figure 3. Overall dimensions of the panel](image)

![Figure 4. Stacking sequence of the panel skin](image)

The Z type Aluminium stiffeners are made from 3.6 mm thick shell, having the cross-section area \( A = 321 \text{ mm}^2 \) and the moment of inertia \( I = 180000 \text{ mm}^4 \). The elastic constants for Aluminium are \( E = 0.7e11 \text{ Pa} \) and \( \nu = 0.3 \). In the case of nonlinear behaviour, for the Aluminium, the value of the yield stress \( F_{xy} = F_{yz} = 240 \text{ MPa} \) and the modulus of plasticity \( E_p = 0.25e8 \text{ Pa} \). For the glass fibres the considered values were \( E_x = 0.5e11 \text{ Pa} \), \( E_y = E_z = 0.152e11 \text{ Pa} \), \( \nu_{xy} = 0.077 \), \( \nu_{yz} = 0.428 \).

The model simulates the so called picture frame method to investigate experimentally the buckling behaviour of a stiffened shear panel, subjecting it to an applied force on his diagonal. The panel is mounted using a very rigid four hinged beam structure, placed in a tensile testing machine (figure 5). In the finite elements model the loads are applied by imposing displacements in the point A. The corresponding applied force is obtained by measuring the reactions in B (figure 6).

![Figure 5. The testing load scheme](image)

![Figure 6. The FEM load scheme](image)

The finite element model uses SHELL 91 elements for the panel skin and BEAM 189 elements for the 6 stiffeners, the 2 girders and the 4 rigid beams. These elements are described in details in [4]. The four
hinged rigid beams constitute a substructure coupled with the model of the panel. For the coupling of several substructures see chapter 13 from [5]. The meshed model is presented in the figures 7 and 8. The maximum element size was 20 mm (figure 9).

4. THE RESULTS

In the first step, we performed a linear bifurcation analysis (eigenvalue buckling) in order to obtain the first buckling modes. The load was applied as concentrated forces acting on the joint of rigid beam like in Fig. 8, and the rigid body motion were suppressed by adequate zero imposed displacements. The figure 10 presents the first buckling mode obtained for a load of approximately 710 kN.

For static nonlinear analysis the load (imposed displacement) was applied in 17 steps. This analysis considers nonlinear (elasto-plastic) behaviour only for Aluminium, the glass fibre layers of the Glare panel being assumed to work in the elastic range. In figure 11 are presented the lateral displacements of the points A, B, C located at the centre of the panel. The buckling load is about 465 KN. The reduction of the buckling load value due to the plasticity is very important. Up to this load, the displacements of the skin and of stringers are the same, so the panel do not buckle locally (it was designed in this manner).
The figures 12 and 13 present the displacements of the panel in the buckling state. This type of failure is known as plastic overall buckling. In the local buckling, the skin can buckle first and then the stiffeners or vice-versa. It depends on stiffeners spacing and inertia, skin thickness, plasticity effects etc.

In the table below are presented the obtained results in comparison with those of reference study [3]. The differences lie in a range of 10 % and are given by a lack of data concerning the geometric details of the experimental setup and lack of all exact material characteristics.
Table 1: Buckling force (KN)

<table>
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<tr>
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<th>Present study (ANSYS)</th>
<th>FEM results ([3])</th>
<th>Experiment ([3])</th>
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<tbody>
<tr>
<td>Linear buckling</td>
<td>710</td>
<td>689</td>
<td>-</td>
</tr>
<tr>
<td>Nonlinear buckling</td>
<td>465</td>
<td>520</td>
<td>510</td>
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5. CONCLUSIONS

A finite element model of a stiffened composite FML shear panel was built with ANSYS code in order to study its buckling behaviour. The dimensions of the panel, taken from [3], are representative for the new giants of the aeronautical industry. The skin of the panel is from a Glare type material, a new material used extensively in aeronautics. The model of the panel was coupled with a model of a rigid frame consisting in four hinged beams in order to simulate the experimental loading setup. For convergence reasons it was found that in the case of the nonlinear study, is better to apply on the model a diagonal displacement instead of a diagonal tensile force. As in [3], the nonlinear study considering elasto-plastic behaviour of Aluminium alloy leads to an important reduction of the buckling force. The tested panel fails due to a general instability named overall buckling.

The understanding of the complex phenomena related to the anisotropy and plasticity effects, together with the appropriate modelling of the panel can lead to an increased confidence in the finite element method which can replace successfully experimental tests with big time and resources economy.

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REFERENCES