Computational Modelling of Failure of Stiffened Composite Panels

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Abstract

This dissertation presents an in-depth computational study on the buckling, postbuckling and strength of stiffened composite panels. It follows up the finished COCOMAT project, supported by the European Commission, with the aim of exploiting large strength reserves in stiffened carbon fiber reinforced polymer (CFRP) fuselage structures. The main goals are to improve the structural efficiency and decrease the structural weight and development and operation costs.

Several Finite Element (FE) models were developed throughout this work and extensive simulations were carried out. The first numerical simulations comprised the postbuckling analysis of a thin-walled stiffened CFRP panel subjected to axial compression with T-shaped stringers, similar to that studied in the COCOMAT project. Alternative damage models considering strength-based criteria and fracture mechanics (Hashin, cohesive elements and eXtended FE Method (XFEM)) were implemented to capture intra-laminar damage in the composite and adhesive failure, respectively. Fiber failure and the detachment between the skin and stringers, caused by damage of the adhesive, were identified as the most severe damage mechanisms leading to structural collapse. Validation of the model of the first panel design arose from the good agreement obtained between the numerical and the experimental and numerical results obtained in the COCOMAT project.

Additional models of several panel designs with different stringer cross-section shapes were created to evaluate their structural behavior under axial compression and bending. The load/moment-carrying capacity and collapse of those panels were analyzed and compared. The one with Ω-shaped stringers revealed to be the most efficient, presenting the highest exploitation of postbuckling reserve strength and lowest weight, thus being recommended to be studied for possible future applications.

Keywords: Stiffened panel, structural efficiency, composite materials, computational analyses, buckling and postbuckling, damage mechanisms

1. Introduction

In recent years, advanced carbon fiber-reinforced polymers (CFRP) are increasingly being introduced into primary fuselage aircraft and spacecraft structures, as engineers are always striving for improving performance and structural efficiency, whilst reducing emissions and weight. The design of fuselage structures taking into account their postbuckling strength has emerged throughout the years. These structures can carry high loads even after their initial buckling loads have been exceeded. Postbuckling-based design has successfully been applied to metallic aircraft structures, but its application with composites has been limited to date. In combination with the high performance of composite materials, the concept of postbuckling-based design has the potential to improve
significantly the structural efficiency, since the ultimate loads can be increased by allowing the structures to be operated past the buckling points. Additionally, composite fuselage structures are lighter, which goes along with the continuous demand for cost reduction.

This new generation of composite fuselage structures requires a reliable and accurate simulation of postbuckling and collapse. Under compression, these structures experience buckling, adopt specific mode shapes and develop a wide range of damage mechanisms, which under further compression into the deep postbuckling region can lead to the collapse of the structure.

The work presented in this thesis was mainly focused on three objectives. The first was to integrate different modeling approaches in the FEA to represent the critical damage mechanisms in a thin-walled stiffened CFRP panel under axial compression, comprising T-shaped stringers similar to those of the panel studied in the COCOMAT project. Three damage models were implemented and evaluated and then considered as an alternative to the user subroutines previously developed in the that project, which are very time-consuming when used with ABAQUS. With the adopted approaches, the prediction of more realistic deformation patterns and closer approximations between numerical and experimental load-axial shortening curves were attempted, but the evaluation of the different failure criteria on the structural behavior of the panel was the first principal purpose. The second main focus of this work was to create several panels designs with different stringer geometries to evaluate their postbuckling structural behavior under axial compression. The load-carrying capacity and collapse of those panels were analyzed and compared. Finally, the last goal of this work was to incorporate an additional bending analysis to all panel configurations, as the study of this load case can contribute to the design of more efficient composite structures.

2. Literature review

Though experimental tests and numerical simulations have been performed on buckling and postbuckling of flat stiffened composite panels, on the other hand, studies on stiffened composite shells and stiffened composite curved panels were scarce at the starting time of the POSICOSS project (“Improved POStbuckling Simulation for Design of Fibre COMposite Stiffened Fuselage Structures”) [1] and its successor COCOMAT (“Improved MATerial Exploitation at Safe Design of COMposite Airframe Structures by Accurate Simulation of COllapse”) [2].

The COCOMAT project, which was comprised of 15 European partners and co-ordinated by the DLR (German Aerospace Center), benefited from the fast and reliable procedures developed by the POSICOSS team, which equivalently investigated the behaviour of stiffened composite panels under compression, but did not take material damage into account. Furthermore, the COCOMAT project went beyond the POSICOSS project by a simulation of structural collapse. The numerical simulations of COCOMAT were performed employing geometrical nonlinear analysis with explicit and implicit solution procedures. The numerical model developed in the COCOMAT project was not able to capture the deformations patterns of the panel observed experimentally. Additionally, the numerical results attained almost completely misrepresented the degree of damage of the adhesive layer. Nevertheless, this project has shown that the incorporation of damage into the FE models is essential.

A high degree of nonlinearity is present in composite stiffened structures, where certain types of nonlinearities act simultaneously. The postbuckling analysis of this structures under compression
involves large strains/rotations and thus nonlinear strain measures and kinematics must be taken under consideration (geometric nonlinearity). Additionally, Orificii [3] claimed that the compression of composite stiffened structures results in several damage mechanisms that represent nonlinearities, such as the reduction in material properties resulting from ply damage mechanisms (material nonlinearity) or the loss of contact caused by the separation/debonding between the skin and stiffeners, as well as the potential delamination between the composite plies (contact nonlinearity) [4] [5].

The Classical laminated plate theory (CLPT) is an extension of the Classical Plate Theory to laminated plates and is the most commonly used in composite postbuckling analysis. In this theory, the in-plane displacements are assumed to vary linearly through the thickness and the transverse displacement is assumed to be constant through the thickness, which means that there is no strain in the thickness direction. This underlying two-dimensional assumption (2D) is accurate as long as the thickness of the laminate is small (at least two orders of magnitude less than the in-plane dimensions) [6]. The main components of a composite fuselage panel (skin and stringers) consist of multiple single unidirectional layers that are relatively thin, so the simplified condition of plane stress is accurate, and loading can be considered to be in the plane of the layer.

For an efficient design of composite structures, damage mechanisms must be considered, so modeling material damage and failure is required. In composite materials, the extreme anisotropy in both stiffness and strength properties and the presence of two different constituents (fibers and matrix) result in various failure/damage mechanisms at distinct levels [7]. Those mechanisms that are relevant to stiffened composite structures can be divided in intra-laminar damage (ply failure), inter-laminar damage (delamination) and a typical failure in stiffened structures known as skin-stringer debonding.

There are two different approaches to characterize the onset and growth of damage in composite structures:

1) **Continuum damage mechanics:**

Within the framework of Continuum Damage Mechanics (CDM), maximum allowable strength-based criteria are commonly used to predict the onset of failure and the progression of damage is achieved by introducing damage variables into the material constitutive law.

2) **Fracture mechanics:**

Classical fracture mechanics is a theory based on the growth of existing defects/cracks in the structure. In this theory, crack propagation is predicted by comparing the computed values of the stress intensity factors, or the components of the strain energy release rate, with the corresponding critical values, taken as material properties.

3. FE model description

The FE models developed for the purpose of this thesis consisted of seven different curved stringer-stiffened panels, which were assumed to be representative of a fuselage section. Each panel comprised a skin with cylindrical shape and longitudinal stiffeners (stringers), created as separate parts, and adhesively bonded. The seven panels were made of carbon/epoxy IM7/8552 prepreg tape and differed from each other either by the number of stringers or by the stringer section geometry considered. The COCOMAT panel (D1) was taken as start design for the purpose of validation of the
experimental data (available within the COCOMAT project), as well as for comparison with the results obtained numerically by Degenhardt et al. [2].

The reference panel is labelled as T5 panel because it has 5 T-shaped stringers. Then, a variation of the number of stringers was performed to check their influence: panels T4 and T6 containing 4 and 6 T-shaped stringers were also modelled. Then, in order to study the influence of stringer geometry, other four different shapes (I, C, J and Ω) of stringer were considered (each panel always with 5 stringers). The geometry of the T5 panel was based on the COCOMAT panel D1, manufactured by Aernnova Engineering Solutions and tested by the Institute of Composite Structures and Adaptive Systems of DLR. It consists of a thin CFRP skin stiffened with five T-shaped CFRP stringers. The geometric data of the panel is given in Table 1.

<table>
<thead>
<tr>
<th>Panel length</th>
<th>L = 780 mm</th>
<th>Stringer blade height</th>
<th>h = 14 mm</th>
</tr>
</thead>
<tbody>
<tr>
<td>Free length</td>
<td>L_f = 660 mm</td>
<td>Stringer width</td>
<td>b = 32 mm</td>
</tr>
<tr>
<td>Radius</td>
<td>r = 1000 mm</td>
<td>Ply thickness of all CFRP layers</td>
<td>t = 0.125 mm</td>
</tr>
<tr>
<td>Arc length</td>
<td>a = 560 mm</td>
<td>Laminate set up of the skin</td>
<td>[90, +45, -45, 0]_S</td>
</tr>
<tr>
<td>Number of stringers</td>
<td>5</td>
<td>Laminate set up of the stinger flange</td>
<td>[(45, -45)_3, 0]_S</td>
</tr>
<tr>
<td>Distance between stringers</td>
<td>d = 130 mm</td>
<td>Laminate set up of the stinger blade</td>
<td>[(45, -45)_3, 0]_S</td>
</tr>
</tbody>
</table>

The additional six panels considered in this work were designed as follows. Panels T4 and T6 present identical T-shaped stringers but differ in their number, as the first contains four and the latter contains six stringers. The remaining panel designs comprised the same number of stringers as the original one (panel T5), but the stringer geometry was varied. Table 2 shows the geometry of those panels.

<table>
<thead>
<tr>
<th>Panel design</th>
<th>Geometry</th>
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<tbody>
<tr>
<td>I</td>
<td>![I]</td>
</tr>
<tr>
<td>C</td>
<td>![C]</td>
</tr>
<tr>
<td>J</td>
<td>![J]</td>
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<tr>
<td>Ω</td>
<td>![Ω]</td>
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Three different damage models (DM) were implemented in the FEA, including damage of the composite material and damage of the adhesive. DM-H includes Hashin’s damage initiation criteria and a damage evolution law for the composite structure to model intra-laminar failure of the CFRP parts (skin and stiffeners), and no damage in the adhesive layer. The other two models incorporate the same damage model for the composite but also damage initiation and evolution of the adhesive by means of two distinct approaches: (i) DM-HC comprises the cohesive element technology applied at the interface of the skin with the stringers to simulate adhesive failure and (ii) DM-HX incorporates the XFEM with VCCT for modelling adhesive failure based on fracture mechanics.

All numerical models were created using conventional shell elements, in which a laminate material definition according to CLPT was applied. The skin and stringers were discretized with 4 node shell elements with reduced integration, denoted by S4R in Abaqus designation. The Hashin’s failure criteria for unidirectional fiber-reinforced composites [8] were applied to evaluate the initiation of fiber rupture and kinking, matrix crushing and cracking and fiber-matrix shear failure. The failure modes are represented and modelled by the degradation (reduction) of the material stiffness to implement the loss in load-carrying capacity [9]. To allow for damage evolution, the critical energy release rate, $G_c$, also known as fracture energy, was specified for each failure mode.

An 8 node 3D cohesive element, COH3D8 in Abaqus nomenclature, was applied in the adhesive layer of the model with DM-HC. Modelling with cohesive elements has many important advantages over other approaches, especially for delamination and debonding, since they have the capacity to predict both initiation and growth of damage in the same analysis, as well as to incorporate both strength and fracture mechanics damage theories [3]. The response of the cohesive elements in the model was specified as a traction-separation law (TSL) through the cohesive section definition, which assumes linear elastic behavior followed by the initiation and evolution of damage [4]. The maximum nominal stress criterion (MAXS) was selected for damage initiation in the cohesive elements. The bond material is assumed to behave with zero ductility until it fails, which means that the initial response of the cohesive element is assumed to be linear. Once damage initiation criterion is met, i.e., after the element passes the strength limit of the bond material, the stiffness is gradually reduced. The loss of stiffness of the interface continues until it reaches a value of zero, at which point the substructures are completely delaminated. The work done in reducing the material stiffness to zero is equal to the fracture toughness, also known as the critical energy release rate ($G_c$).

The XFEM analysis is only available for 3D geometric parts. In the models with DM-HX, the 8 node linear hexahedral solid element, C3D8 in Abaqus designation, with full integration, was used to model the adhesive layer. As no initial crack is present in the structure, damage initiation was specified in the material property definition using the MAXS criterion with the strength values used in the cohesive zone model approach. The strain energy release rates at the crack tip were calculated based on the VCCT, which is based on the assumptions that (i) the energy released in crack growth is equal to work required to close the crack to its original length, and (ii) the crack growth does not significantly modify the state at the crack tip [10]. VCCT becomes active when damage initiation criteria are met, and a crack appears and propagates according to XFEM. The simulation using XFEM with VCCT caused some complications as only one crack initiated and propagated in the selected 3D region, and thus modeling multiple cracks could only be achieved by creating partitions in the whole model.
The postbuckling behavior of the panels, either in compression or bending, is highly sensitive to the applied boundary conditions (BC). The fixed (clamped) side of the panels has all 6 DOF restrained. The right and left sides of the panels were set free, according to the experimental tests performed in the COCOMAT project. These BC were common to all load cases. At both ends, the first 60 mm in length were encased in resin to restrict out-of-plane movement and all rotations. The end loadings were applied, depending on the analysis or load case, using a prescribed concentrated force of $P = -1 \, N$, an axial displacement of $u_z = 4 \, mm$ or a rotation about the out-of-plane axis of $|\theta_\gamma| = 0.015 \, rad$.

To ensure that all nodes located at the panel ends had the same displacement/rotation, the loaded edge had to possess a rigid body motion. That was achieved by applying a reference point (RP) located at the centre of curvature was assigned to the rigid edge.

In all panels, a linear buckling/eigenvalue analysis was firstly conducted to extract eigenvalues (buckling loads) and buckling modes. The latter were subsequently used in the nonlinear analysis as imperfections. The geometric imperfections were introduced to all nonlinear models and were based on the first 3 buckling modes extracted from the linear buckling analysis, as the lowest buckling modes are considered to provide the most critical imperfections. The nonlinear analysis (compression and bending) for all panel variations was carried out with the implicit solver provided in Abaqus/Standard based on Newton’s Raphson method. In order to assist with convergence issues, numerical damping was incorporated into the analysis. The automatic stabilization scheme was activated using a damping factor of $2 \times 10^{-6}$.

4. Numerical simulations and evaluation

This section presents the main numerical results of the analyses of the FE models developed throughout this work.

4.1. Postbuckling of reference panel T5 under compression

Figure 1 exhibits the load-shortening curve of panel design T5 without damage of the materials included.

![Panel design T5 - no damage included](image)

*Figure 1: Load-shortening curve for panel design T5 without damage*
The results without damage exposed the typical behavior of compressed stringer-dominant panel designs, where the 3 remarkable load levels can be easily distinguished. The lowest one, a local buckling region, where buckling waves develop in the skin between the stiffeners, occurs in this structures as the first buckling mode. Afterwards, a slight stiffness reduction occurs. The second level is the onset of buckling of the stiffeners (global buckling) and is represented by a higher reduction of the axial stiffness. Collapse is the highest level and is specified by the point of the curve where a sharp decrease in the axial stiffness occurs.

All models studied by means of non-linear analyses were ran with imperfections displaying maximum value of 10% of the thickness, which had a minimal effect on the deformation progression and load-carrying capacity, as well as on other panel characteristics.

**Numerical results of models with damage**

The assessment of the achieved numerical results comprised the validation by comparison with the experimental results, which included the comparison of the load-shortening curves, given in Figure 3 and deformation patterns or mode shapes (not shown in this paper).

![Panel design T5 - comparison of results](image)

*Figure 2: Load-shortening curves of the numerical models developed and experiment*

From Figure 3, one can see that all numerical load-axial shortening curves show a very good agreement regarding initial axial stiffness, up to the point of global buckling. However, all models predicted a higher global buckling load than that measured on the experimental test, with a relative difference of about 20%. From that point, the first model developed, i.e., the one without the inclusion of damage, was the one that most overestimated the panel load carrying capacity in the postbuckling region, which underlined the necessity of implementing damage models. It can be concluded that the model that combined the XFEM with Hashin’s damage criteria (DM-HX) was the one that resulted in the closest prediction of the load-carrying capacity determined experimentally. The progression of this numerical load-axial shortening curve also looked very similar to its experimental counterpart, although the former always lies above the latter. The collapse occurred for $u_z = 2.65$ mm, which is a very good prediction because the tested panel collapsed for $u_z = 2.71$ mm.
4.2. Postbuckling analysis of different panels in compression and bending

The postbuckling analysis of panels subjected to compression was extended to the additional six designs (T4, T6, I, J and Ω). The load-shortening curves of all panel designs are shown in Figure 3. The influence of the number of T-shaped stringers (comparison between the reference panel T5 with T4 and T6) and the influence of the stringer shape (comparison between the reference panel T5 with I, C, J and Ω) on the postbuckling and strength of panels was studied in detail. Herein, all panel designs were analyzed and compared merely including the composite damage model (DM-H).

Panel design T4 turned out to be the one presenting the lower stiffness of the three T-shaped panel versions, as expected. On the other hand, panel design T6 was naturally predicted to be the stiffest of all variations of the T-shaped panels. Panel design I was the stiffest of all panel designs, since it is the one with largest total stiffener cross-section area. This panel was capable of withstanding a maximum axial load of $P_u = 217.8$ kN, which is almost two times that of panel T5. However, this panel collapsed suddenly for $u_z = 2.18$ mm and was considered to the most brittle of all panels. Panel design C presented a similar behavior as that of panel T5, with practically the same initial stiffness, since both panels possess the same total stiffener cross-section area. Panel design J presented a similar initial stiffness as that of T6 panel up to the first global buckling point and the maximum load of $P_u = 156.3$ kN was achieved right before global buckling.

Based on some of the conclusions achieved by the previous comparisons, an attempt was made for improving the structural behavior of panel design Ω, specifically with the objective of enhancing its axial stiffness after global buckling, as it was perceived that this panel had the most noteworthy results. Thus, it was found that if the laminate layups of the three flanges of each stiffener were inverted (from $[(45, -45), 0]$ to $[0, (45, -45)]$), the structural response of the panel would become even better, with the stiffness after global buckling being substantially increased.
The assessment and comparison of the load-shortening curves is believed to offer a direct contribution to the future designs of composite stiffened structures, where the onset of damage is allowed in the safety region, the limit load is much larger than the first local buckling load and the ultimate load is shifted towards the structural collapse as close as possible. The best panel design was selected as the one that could withstand the axial load and, at the same time, be the one with the highest structural reserve capacity between the first buckling load and collapse. Following this, panel design Ω-modified was chosen as the best design among all configurations. This panel exhibits a progressive change from local to global buckling, a large and “stable” postbuckling shortening from 0.57 mm to 2.90 mm of axial shortening and a maximum axial load of $P_u = 178.5$ kN. Additionally, this panel design is lighter than panel T5 studied in the COCOMAT project, as it required approximately $524\,690$ mm$^3$ of CFRP prepeg IM7/8552 UD, whereas T5 panel needed about $787\,800$ mm$^3$ of the same carbon/epoxy composite material, which goes along with the continuous demands for decreases in the structural weight.

The analysis of the panels subjected to bending was also incorporated in this work because it is known that the axial stresses developed along the circumferential direction of a fuselage are not uniform and may vary linearly, and thus the stress gradient arising from these 2nd order forces is well defined through a linear stress distribution equivalent to the application of a bending moment. Therefore, accurate predictions of damage and collapse regarding this load case are worthy of being studied and can contribute to the design of more efficient composite structures. The bending moment-rotation curves of all panel designs are shown in Figure 4.

The assessment of the bending moment-rotation curves presented a close correlation with those of the load-axial shortening curves of the same panels subjected to axial compression. Panel Ω-modified was considered to have produced the most noteworthy results regarding the bending response to a prescribed rotation around y-axis, as it exhibited the second highest value of the maximum bending moment with respect to the y-axis that the panel can withstand combined with a significant post bending rotation, as well as the largest rotation at collapse. The panel design I also presented distinguished results with this loading case, though it started to collapse right after the first loss of
flexural stiffness, and thus was the most brittle of all panels, which does not go along with the future aims for the design of stiffened composite panels, where the stiffeners can withstand significant deformations in the safety region before the ultimate loads and collapse.

5. Conclusion

Regarding the postbuckling analysis of panel T5 (panel with five T-shaped stringers) under axial compression, which was created based on the COCOMAT design (D1), it was clear that an analysis approach combining more than one damage model would be highly attractive, as it would allow the potential interactions between strength-based failure criteria and fracture mechanics to be investigated. The numerical results concerning this panel design were then compared with the experimental data, and it was shown that the approach with DM-HX (Hashin + XFEM) led to the closest prediction of the panel behavior in terms of the load-shortening curves and of the shortening and load for which collapse occurred. On the other hand, the model with DM-HC (Hashin + cohesive elements) was able to represent an asymmetric deformation pattern close to that observed experimentally as well as a good prediction of the areas where skin-stringer debonding took place.

The postbuckling analysis of the different panels in compression and bending permitted to identify their load-shortening and moment-rotation response to an applied axial displacement and rotation, respectively, the onset of damage mechanisms and their capability to resist further increase in load after the first global buckling load. Panel design Ω-modified was chosen as the best design in terms of structural efficiency as it evidenced the highest exploitation of postbuckling reserve strength. This panel is also lightest of the ones studied, in particular, lighter than the panel studied in the COCOMAT project, and thus it is recommended to be studied for possible future applications.

Hence, the achieved numerical results are believed to offer a direct contribution to the future designs of composite stiffened structures, as well as to be a contribution to the aim of structural weight reduction, and consequently to allow the European aircraft industry to reduce development and operation costs in the short and long term. Although this project was mainly focused on fuselage panels under axial compression and bending, the analysis approach and the damage models applied can be easily transferable to other composite structures and loading cases.

6. References