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Large Notch Damage Evolution in Omega Stiffened Composite Panels

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Abstract

The aim of this work is the study of the influence of a large notch damage in a stiffened aeronautical panel. In particular, the damage onset and evolution due to a cut-out located in the bay of an omega stiffened composite panel subjected to a compressive load is investigated. Three different cut-outs are considered: parallel, normal, and 45° oriented respect to the load. The effects of such configurations are compared in terms of fibre and matrix failures, in order to better understand which configuration is the most sensitive to these type of damages.

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1. Introduction

Composite materials are usually characterised by high stiffness/weight and strength/weight ratios, hence they are suitable for application in many industrial sectors where the weight reduction is of primary concern. However, composites have been demonstrated very sensitive to damage (usually characterised by complex and hardy predictable failure mechanisms) leading to over-conservative designs, far from a full realization of the promising economic benefits.

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A relevant number of numerical approached can be found in literature for the prediction of onset and the evolution of intra-laminar and inter-laminar damages [1,2]. Among these, the Ultimate Laminate Failure models adopt a Progressive Failure Approach (PFA) to evaluate the ultimate load of composite laminates [3-8].

From literature, it appears clear that the material degradation phase and the related definition of damaged element residual stiffness are among the most critical aspects of a Progressive Failure Analysis. These aspects are still under investigation especially for complex composite components, such as stiffened panels, with a pre-existent damage susceptible to propagation under service loading conditions.

In this paper, the large notch damage evolution in an omega stiffened composite panel subjected to a compressive load by means of a Progressive Failure Approach has been investigated. Three different configurations of the large notch damage have been considered with notch differently oriented with respect to the load direction. The effects of notch orientation on the mechanical behaviour of the panel have been assessed. The numerical results for all the three analysed configurations have been compared in terms of intra-laminar damage onset and evolution, in order to better understand which configuration is the most notch sensitive. In Section 2, the theoretical background for the intra-laminar damage model adopted in this paper is presented. In Section 3, the investigated panels have been introduced with geometrical and FE models detailed description. Finally, in Section 4 the numerical results obtained for all the investigated configuration are presented and compared.

2. Intra-laminar damage model

An intra-laminar damage model has been adopted in this paper in order to study the effects of the large notch orientation on stiffened composite panels compressive behaviour. In particular, the Hashin's failure criteria [9-11] implemented in the standard version of the FE code ABAQUS and shown in Table 1, allowing the evaluation fibre and matrix compressive and tensile failure modes, have been used. The progressive intra-laminar damage has been simulated by assuming as internal state variables the damage index d_i, representative of the material stiffness degradation for each failure mode i.

Table 1: Test Matrix

Fiber tension	$\hat{\sigma}_{11} > 0$	$F_{ft} = \left(\frac{\hat{\sigma}_{11}}{X_T}\right)^2 + \left(\frac{\hat{\sigma}_{12}}{S_L}\right)^2 = 1$
Fiber Compression	$\hat{\sigma}_{11} < 0$	$F_{fc} = \left(\frac{\hat{\sigma}_{11}}{X_C}\right)^2 = 1$
Matrix tension	$\hat{\sigma}_{22} \geq 0$	$F_{mt} = \left(\frac{\hat{\sigma}_{22}}{Y_T}\right)^2 + \left(\frac{\hat{\sigma}_{12}}{S_L}\right)^2 = 1$
Matrix compression	$\hat{\sigma}_{22} \leq 0$	$F_{mc} = \left(\frac{\hat{\sigma}_{22}}{2S_T}\right)^2 + \left[\left(\frac{Y_C}{2S_T}\right)^2 - 1\right] \cdot \frac{\hat{\sigma}_{22}}{Y_C} + \left(\frac{\hat{\sigma}_{12}}{S_L}\right)^2 = 1$

According to Table 1, $\hat{\sigma}_{11}$ and $\hat{\sigma}_{22}$ are respectively the stress in the fibre direction and normally to the fibre direction; X_T, X_C, Y_T, Y_C, S_L, and S_T are respectively the Fibre tensile, Fibre compressive, Matrix Tensile, Matrix compressive, Longitudinal, and Transversal strengths. F_{ft}, F_{fc}, F_{mt}, and F_{mc} are respectively the values representative of the Fibre tensile, Fibre compressive, Matrix Tensile, and Matrix compressive criteria: when one of such criteria is met, the corresponding damage is initiated.

The graphical representation of the constitutive relations, describing the damage onset and evolution, adopted for each failure mode is shown in Figure 1.