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Failure analysis in postbuckled composite T-sections

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ABSTRACT

Blade-stiffened skin designs made of composite materials have the potential to produce highly efficient structures, when the large strength reserves in the postbuckling range are utilised. This paper investigates the failure under postbuckling deformations of T-section specimens cut from a blade-stiffened panel, by comparing experimental results to finite element models. In the experimental work, T-section specimens with a particular lay-up and geometry were tested to failure in antisymmetric and symmetric loading rigs. These loading rigs simulate deformations on skin-stiffener interfaces during panel postbuckling. For the numerical analysis, two-dimensional models of the interface cross-section were used with a strength-based criterion that monitored failure within each ply. The use of a zero-thickness layer of cohesive elements has also been investigated in order to simulate the delamination behaviour. The numerical predictions are compared to the experimental results in terms of the failure load, specimen stiffness and specimen behaviour. The analysis approach is shown to be capable of predicting the critical damage locations and initiation loads for both antisymmetric and symmetric loadings. The successful prediction of failure in skin-stiffener interfaces can be linked to a global-local approach for efficient analysis of large, fuselage-representative composite structures.

1. Introduction

Postbuckling is the phenomenon in which structures continue to carry loads higher than the critical buckling load. Stiffened skin panels made of composite materials have the potential to produce highly efficient structures, when the large strength reserves in the postbuckling range are utilised. Such composite structures are ideal for fuselage and wing skin panels on aircraft. Postbuckling design has been used with metallic aircraft for decades to reduce weight. However, to date the application of postbuckling design with composite structures has been limited, owing to concerns relating to the durability of the composite structures. Unlike metals, composites do not yield locally under the high local stress field set up during the postbuckling phenomenon. Furthermore, there are concerns regarding the high through-thickness stresses that are set up and the development of defects within the laminates is not well understood and cannot be predicted accurately using current design tools.

When blade-stiffened composite panels are loaded in compression, generally two types of buckling deformation patterns are observed, as shown in Fig. 1. One is "local" buckling of the skin between the stiffeners shown in Fig. 1a, and the other is "global" buckling along the panel length as shown in Fig. 1b. Failure under these types of deformations usually occurs at nodal or anti-nodal lines in the structure, where either the bending or twisting moments are at their maximum [1].

In experimental testing of stiffened composite structures in compression, failure typically involves delaminations at the stiffener base or at the edge of the stiffener flange [2]. The onset of delamination typically leads to catastrophic collapse, resulting in a rapid decline in the load carrying capacity of the structure. It is, therefore, important to develop design tools to predict such failures and thus lead to improved panel design.

In finite element (FE) analyses of postbuckling stiffened structures, it has been shown that good definition of the global structural response can be obtained by relatively coarse modelling with shell elements [3]. However, to accurately determine the correct failure mechanisms requires fine models of the skin-stiffener interface with a large number of elements, which is infeasible in the analysis of large structures. An alternative approach is to use a two-step approach, in which a coarse global model is combined with the analysis of local models at regions of potential failure

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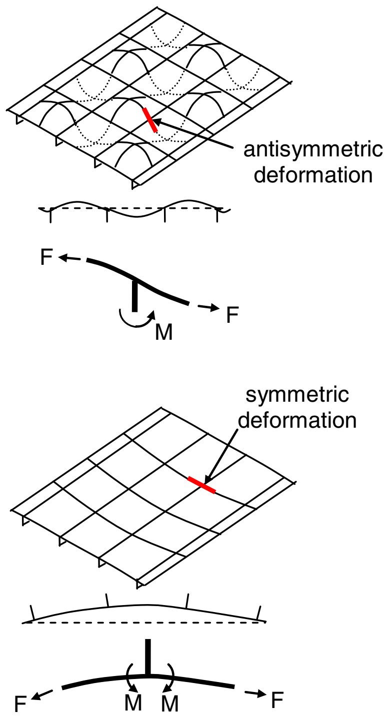

Fig. 1. Postbuckling deformations: Top: antisymmetric; Bottom: symmetric.

based on the global panel buckling patterns. This approach can be applied for the rapid analysis of large structures, and has been developed and demonstrated in experimental and numerical investigations of fuselage-representative composite structures [4].

This paper describes the failure analysis of T-sections from a stiffened composite panel loaded with typical postbuckling deformations. In previous work, experimental and numerical investigations of T-sections specimens with ply drop-offs were conducted [5]. In this work, the analysis approach is extended to study specimens without ply drop-offs, and in an additional loading case. Antisymmetric and symmetric postbuckling deformations have been investigated, where the symmetric loading consisted of both pull and push tests. A strength-based approach is used to predict the initiation of interlaminar damage, in which stresses in the critical directions are compared to material strengths and combined into a single failure index. FE analysis was conducted using the nonlinear solver in MSC.Marc (Marc) (2005r3) and a user subroutine written to determine the failure index for each element. The FE models are compared to the experimental results in terms of the specimen behaviour, specimen stiffness and failure load. Recommendations are then given for the application of this approach for the analysis of fuselage-representative structures.

This work is part of the European Commission Project COCOMAT (Improved MATerial Exploitation at Safe Design of COmposite Airframe Structures by Accurate Simulation of COllapse), an ongoing four-year project that aims to exploit the large strength reserves of composite aerospace structures through more rapid and accurate prediction of collapse [6,7].

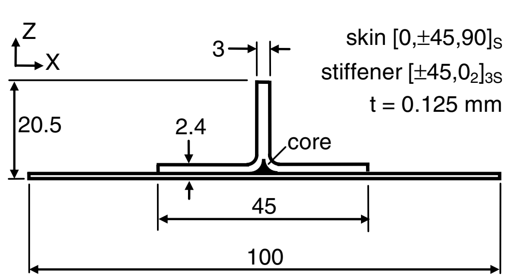

Fig. 2. Specimen geometry and ply lay-up.

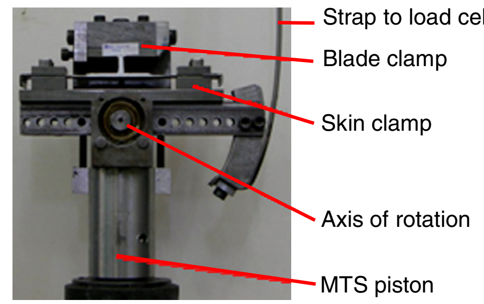

Fig. 3. Antisymmetric test rig.

2. Experimental investigation

A fuselage-representative blade-stiffened composite panel was manufactured at Israel Aircraft Industries and tested at the Aerospace Structures Laboratory at Technion as part of the COCOMAT project. The panel was manufactured from IM7/8552 carbon fibre unidirectional prepreg. Thin strips were cut from these panels consisting of the skin and a single stiffener, each with a width of 25 mm. The ply lay-up and specimen geometry are shown in Fig. 2.

A total of 25 specimens were tested in two test rigs to simulate three postbuckling deformations as shown in Fig. 1: antisymmetric, symmetric pull and symmetric push. There were ten specimens each for the antisymmetric and pull tests, and five specimens for the push tests.

The antisymmetric deformations were simulated using the test rig in Fig. 3. The skin clamps were on a rotating fixture, one end of which was connected to a load cell. The piston moved vertically downwards and the blade and skin clamps translated accordingly, with the skin clamps also rotating about the axis of rotation due to their connection to the load cell. The moment applied to the specimen was measured by multiplying the reaction force at the load cell with the distance between the circular segment and the axis of rotation. This applied moment was then normalised by dividing by the specimen width. The applied energy could be obtained by multiplying the moment with the angle of rotation. However, in this work the normalised moment is used as the defining parameter for the antisymmetric tests.

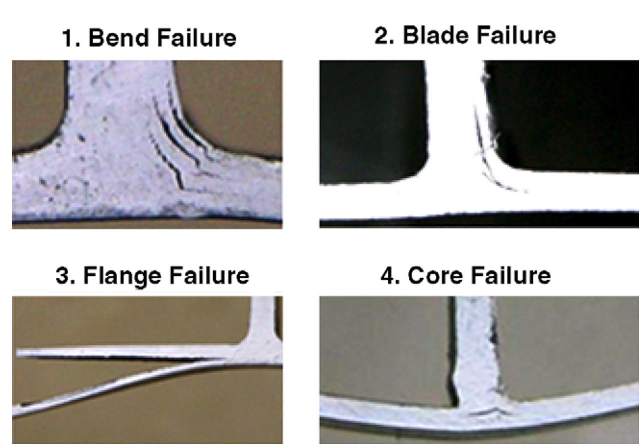

Fig. 4. Failure mode classification.