*The 2015 World Congress on*
***Advances in Structural Engineering and Mechanics (ASEM15)***
*Incheon, Korea, August 25-29, 2015*

# Stiffened composite panels with cut-outs: Strategies for modelling compressive response

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## ABSTRACT

In the case of compressive loading, near-surface instability or fibre microbuckling in the $0^0$ plies may occur where high stress gradients exist. The critical load can be estimated by using the maximum stress failure criterion, giving a conservative estimate for the failure load. Alternatively, the Soutis-Fleck fracture model can be employed, where the microbuckling growing from the edge of the hole is mathematically represented by a through-the-thickness crack and fracture mechanics concepts are applied. In both models, the unnotched compressive strength is required, which can be estimated by using a fibre instability model based on a 3-D stability theory of deformable bodies.

## 1. INTRODUCTION

The investigation of compressive behaviour of stiffened thin-skinned composite panels with stress concentrators is dictated by the demands of aircraft industry. A typical aircraft structure such as a fuselage shell or a wing surface usually consists of a thin skin reinforced with stiffeners. The need for an open cut-out in a structural component is required by practical concerns. For example, cut-outs in wing spars and cover panels of commercial and military transport wings are needed to provide access for hydraulic and electrical lines and for damage inspection. Also, cut-outs in a fuselage can serve as access panels and lightening holes. However, such cut-outs introduce high local stresses that can initiate damage and lead to premature failures. Low velocity impact damage caused by dropped tools, runway debris and hailstones can be another source of stress concentration and therefore weakening of the structure (Guz et al 2012).

During a component's service life, it will experience compressive loads and its strength becomes an important design parameter, since compressive strength of currently used carbon fibre-epoxy composites is only 60-70% of their tensile strength (Budiansky and Fleck 1994; Schultheisz and Waas 1996; Niu and Talreja 2000). A

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better understanding of the compressive strength and failure mechanisms is therefore fundamental to the development of improved materials.

In some applications, these structural members are required primarily to resist buckling, and in other cases they must carry load well into the post-buckling range in order to yield weight savings. Thus, understanding their buckling and post-buckling behaviour is needed for the appropriate design.

## 2. EXPERIMENTAL OBSERVATIONS AND FAILURE MECHANISMS

Usually research on open holes and impact damage in carbon-fibre composites are based on testing of small laminates rather than structural elements or full-scale structures. Most of the relevant research has focused on the buckling and post-buckling response of stiffened panels where failure occurs due to large out-of-plane deflections (more than twice the skin thickness) at compressive loads far below the ultimate static strength of the composite material.

A comprehensive review of the early experimental studies of buckling and post-buckling behaviour of laminated composite plates with cut-outs was given in Nemeth (1996). Later the influence of stiffeners on behaviour of damaged and undamaged plates was experimentally studied in Greenhalgh et al (1996), Greenhalgh et al (1999), Jegley (1992, 1998), Kong et al (1998), Stevens et al (1995). The post-buckling compressive strength of undamaged panels with I-stiffeners was reported (Kong et al 1998) to be 5-7 times higher than the buckling load for the same plate. The effect of stiffeners on the stress distribution around a hole, as well as on buckling and failure characteristics of lightweight composite panels is the subject of the survey (Leissa 1987). More recently, this subject was addressed in Degenhardt et al (2007), Falzon et al (2001), Found et al (2002), Greenhalgh and Hiley (2003), Greenhalgh and Garcia (2004), Ishikawa et al (2005) and numerous other publications mentioned there.

The important point that surfaced in the work by Nemeth (1996) is that understanding the response of stiffened plates with an open hole is very fragmented. There is a definite need for studies that attempt to isolate and articulate each fundamental aspect of the compressive behaviour in a consistent manner. Based on available experimental data it was concluded by Nemeth (1996) that careful studying of the behaviour of subcomponents is a necessary fundamental step in any research of composite structures with discontinuities.

Knowledge of the basic response of the subcomponent provides a valuable insight into modelling complex structures with general finite element codes. Furthermore, knowledge of the subcomponent response is very useful for identifying erroneous results that may be obtained due to improper finite element modelling. The above observations justify special attention, which is paid to the “step-by-step” analysis of the general problem starting from the simplest fracture mechanisms for subcomponents or heterogeneous materials of subcomponents (i.e. starting from sub-problems). In doing so both special purpose analytical methods, e.g., (Guz 1998, 2005; Guz and Herrmann 2003), and more general finite element methods, e.g., (Winiarski and Guz 2008) with wider domain of applicability have distinct advantages of their own. The special purpose analyses are typically more limited in scope than the finite element methods, but they are being used to conduct extensive parametric studies of buckling behaviour. These analyses are valuable because they can easily establish trends in behaviour that are in good qualitative agreement with experimental data (Nemeth 1996).

Soutis and Fleck (1990) and Soutis and Curtis (1996) examined the influence of single and multiple holes on the compressive behaviour of several T800/924C carbon fibre-epoxy laminates without stiffeners. In this and later publications it was found that open holes reduce the in-plane compressive strength by more than 40% depending on lay-up and hole size. Damage is initiated by $0^0$ fibre microbuckling at the edge of the hole at approximately 80% of the failure load, and is accompanied by matrix cracking of the off-axis plies and delamination between the plies. This damage zone continues to grow, first in short discrete increments and then rapidly across the laminate width at a failure load that is higher than that predicted by the maximum stress criterion. Fibre buckling leads to local delamination when the local strain necessary to accommodate the localised fibre displacement and rotation exceeds the resin ductility. These local delaminations do not propagate to become macroscopic delaminations until final compressive failure occurs. For $0^0$ dominated laminates, the damage zone is more crack-like in nature and its length immediately prior to failure is in the region of 2-3mm long.

A similar damage pattern was observed in plates with impact damage under uniaxial compression (Soutis and Curtis 1996). The distribution of damage through-the-thickness determined from sectioning studies is roughly cylindrical in shape; ultrasonic C-scan images and X-ray shadow radiographs indicate that the shape of the overall damage is approximately circular (equivalent to an open hole).

## 3. MODELLING STRATEGIES

The presence of an open hole or impact damage introduces high local stresses that may initiate material failures and lead to catastrophic fracture of the structure or component before Euler buckling occurs. In the case of compressive loading, near-surface instability or fibre microbuckling in the $0^0$ plies may occur where high stress gradients exist. The critical load can be estimated by using the maximum stress failure criterion, giving a conservative estimate for the failure load (Zhuk et al 2001, 2002). Alternatively, the Soutis-Fleck fracture model (Soutis and Fleck 1990) can be employed, where the microbuckling growing from the edge of the hole is mathematically represented by a through-the-thickness crack and fracture mechanics concepts are applied – see, for example (Zhuk et al 2000, 2001, 2002). In both models, the unnotched compressive strength is required, which can be estimated by using a fibre instability model based on a 3-D stability theory of deformable bodies (Guz 1999).

In the analysis by Guz et al (2012) the stiffened panel is assumed to be in the pre-buckling state. The strategy is to divide the problem into simpler “sub-problems”. In the case of the stiffened panel, the critical load for the unnotched laminated skin is obtained first, and then the effect of the hole or impact damage is introduced, followed by material anisotropy, finite width effects and finally the presence of stiffeners. In a multidirectional laminate the location of the $0^0$ layer through the laminate thickness and the orientation of the neighbouring (supporting) ply can also have a significant effect on the initiation and final failure. For instance, the failure strain of a laminate with $0^0$ outer layers can be more than 10% lower than a similar lay-up with $\pm 45^0$ outer plies, due to out-of-plane fibre microbuckling (Soutis 1994). The outer off-axis plies provide better lateral support to the $0^0$ layers, permitting them to fail by in-plane microbuckling, which is a higher strain failure event.

Using the three-dimensional stability theory (Guz 1999) and treating the laminate as a homogeneous anisotropic material, the critical failure stress for near-surface microbuckling can be obtained following (Guz et al 2012), where a simple formula was derived which accounts for all geometric and material variables. It provides an easy way of examining the effect of each parameter on the critical failure load of a stiffened composite panel under uniaxial compression, where large out-of-plane deflections do not occur. This critical value of stress, being the result of a 3-D exact solution of the corresponding problem (assuming perfect fibres and interfaces), allows us to identify the effect of material properties on the critical load in a very clear way. However, this does not account for the high-localised stress field developed near the hole or impact damage. It needs to be modified and applied at the ply level (ply-by-ply stress analysis).

Considering the plate with effective constants (i.e. as a homogeneous quasi-isotropic), the model cannot take into account the actual failure initiation, which is always associated with the particular, “critical” layer in the laminated skin. To resolve this issue the “first ply failure” concept is adopted. It means the above critical failure stress should be found for the particular “critical” layer, where failure is more likely to initiate, rather than for the whole skin using effective (average) elastic constants, see (Guz et al 2012).

Fibre waviness within a conventional laminate severely degrades the compressive strength and stiffness of polymeric fibre composites (Berbinau et al 1999). Prepreg tape has an inherent waviness, which is compounded during subsequent layering and compaction. Automated tow placement reduces waviness during placement but does not eliminate it during compaction and cure. Previous works (Berbinau et al 1999; Soutis 2000) have shown that a misalignment angle between the fibres and the loading axis of only 10 to 20 is sufficient to reduce the compressive strength of the T800/924C carbon fibre-epoxy system by more than 40%. Post-failure examination of the fracture surfaces using a scanning electron microscope revealed that failure is by fibre microbuckling. The microbuckling model developed by Soutis (1994) and Berbinau et al (1999) or the 3-D stability theory (Guz 1999) can be used to get a better estimate of the critical buckling stress, but the mathematical expressions involved are more complex to solve.

In the study by Guz et al. (2012), a knock down factor, which is consistent with theoretical findings (Berbinau et al 1999; Zhuk et al 2001, 2002), was introduced to account for the discrepancy caused by the irregular spacing of fibres, matrix plasticity, fibre misalignment and poor fibre-matrix bonding. It helps to maintain the simplicity of the developed approach. In the range of the model applicability, critical loads predicted considering the phenomenon at the microlevel and taking into account the geometry of plate and stiffeners were close to measured data.

## 4. CONCLUSIONS

A comprehensive fracture analysis of stiffened composite panels should consist of two stages. In the first, one should perform the in-plane fracture analysis described in this work and the second stage should be devoted to obtain the load when out-of-plane buckling generates fracture. The minimum of these two loads should be taken as the design fracture load (critical load) of the given panel. In the case of impact damage information is required on the location and size of damage that can be obtained by appropriate structural health monitoring (SHM) methods.

## ACKNOWLEDGEMENTS

Financial support of the part of this research by The Royal Society, The Royal Academy of Engineering, and The Carnegie Trust for the Universities of Scotland is gratefully acknowledged.

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