Interlaminar fracture toughness approach of delaminated composites under variable loading

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Abstract

The purpose of the present paper approaches the modelling of progressive damage and failure in composite laminates which is designed by interlaminar delamination in the aeronautical structures. The analytical model is based on a fracture mechanics approach; it's used to estimate the total mixed mode energy release rate for composite laminates. A finite element simulation has been achieved in combination with the virtual crack closure technique (VCCT) to analyse the effect of temperature on the mixed-mode interlaminar fracture toughness and fatigue delamination growth rate of a carbon/epoxy material, namely IM7/8552 subjected to mechanical loading at variable temperatures. The developed model may serve as the basis for treating different types of thermal and mechanical loading, different stacking sequences and thickness of lamina in order to build safe working conditions for composite laminates.

Keywords: *Interlaminar; Failure; Composite laminate; fracture toughness; Temperature; Delamination.*

1. Introduction

The laminated composites are increasingly used in many aerospace applications such as advanced aircraft fuselage, rocket motor cases, pressure vessels, containment structures, and other components with various shapes and sizes due to their number of advantages over conventional materials. They have exceptional characteristics such as: high specific strength and stiffness, low density, good fatigue performance, resistance to corrosion and high temperatures, ability to create complex shapes. For implementation of composite materials in aviation, the most important feature is their behaviour on dynamic loads and resistance to fatigue[1].

However, Fiber reinforced composites often exhibit complex failure mechanisms as an interaction among various damage modes on both microscopic and macroscopic scales such as matrix cracking and interlaminar damage modes (Interfacial cracking between layers) or delamination.

This paper addresses delamination, the most frequent failure mode in laminated composite materials and it may cause catastrophic failure in critical aeronautic structures.

In recent years, several studies have been carried out into the fracture of composites in their different stress modes under static loading, of which modes I and II have attracted more attention. A series of numerical investigations presented in literatures lead to excellent results. These methods are more suitable because of their low cost and time consuming. R. Krueger developed a finite element models using 3D shell elements which demonstrated good accordance with experimental results [2]. The calculation of delamination can be performed using cohesive elements [3, 4], which combine aspects of strength based analysis to predict the onset of damage at the interface and fracture mechanics to predict the propagation of a delamination. Initiation and propagation of delamination studied numerically with using cohesive elements and different constitutive laws lead to excellent results [5]. Over the past two decades, The criteria used to characterize the onset and growth of composite reinforced delamination under mixed-mode loading conditions are those usually established in terms of the components of the energy release rate and fracture toughness. It is assumed that the growth of delamination in composite structures starts when strain energy release rate G under service loads exceeds the fracture energy G_C. Wang et al. evaluated strain energy release rates for the damage-tolerance analysis of skin-stiffener interfaces using Finite element analysis in conjunction with the virtual-crack-closure technique (VCCT) [6,7]. They used a wall offset to move the nodes from the reference surfaces to a coincident location on the interface between the skin and the flange.

In the present work, an attempt has been made to predict the initiation and evolution of delamination mechanism of Carbone /Epoxy composite material by adopting one of the numerical intelligence concepts that have proved to be useful for various engineering applications. For this purpose, a numerical model has been developed by using a special shell finite element model that guarantees interlaminar shear stress continuity between different oriented layers, at a temperature range of operating conditions for composites in aeronautics. And then,generate mode I and mode II components of mixed-mode fracture toughness.

Further, the prediction results have been compared with the available references experimental data. It was found that the finite element model proposed gives better prediction with less computational time than other intelligent models.

2. Approach and Methodology

2.1 Delamination approach and theory

The well-known Paris law is the most commonly used method to model fatigue delamination growth. In its simplest form, the Paris law can be written as:

$$\frac{da}{dN} = A(\Delta G)^m \quad (1)$$

Where a is the crack length, N is the number of cycles, is the total energy release rate range, and A and m are material specific parameters which must be determined experimentally.

Following the Griffith fracture theory [8], crack extension occurs when the amount of energy required to produce unit area of fracture surface is supplied by the system. The fracture surface energy which is a so called energy release rate is equal to the derivative of potential energy with respect to crack size. In the classical fracture mechanics, energy release rate is determined experimentally from the compliance method as follows:

$$G = \frac{P^2}{2B}\frac{dC}{da} \quad (2)$$

Where a represents the crack length, B the specimen thickness, P the applied load, and C represents the compliance.

G_I and G_{II} were calculated from the crack closure method [8]. That is, G_I and G_{II} were calculated as follows:

$$\begin{cases} G_I = \dfrac{F_y(v_c - v_d)}{2\delta a} \\ G_{II} = \dfrac{F_x(u_c - u_d)}{2\delta a} \end{cases}; \quad (3)$$

$$G_T = G_I + G_{II} \quad (4)$$

Where, δa is a crack extension size, Fx and Fy are forces in x- and y-direction. The displacements, uc (ud) and vc (vd) are the sliding and opening displacements at node “c” (node “d”) on the crack faces, respectively.

2.2 Data preparation and finite element study

A finite element modeling with a series of vcct method has been achieved.

A three-dimensional finite element model of the cracked-lap-shear specimen (CLS) Fig.1. was constructed using Abaqus finite element code in order to determine the total interlaminar fracture toughness mode 1 and II of Carbon-Epoxy composite material.

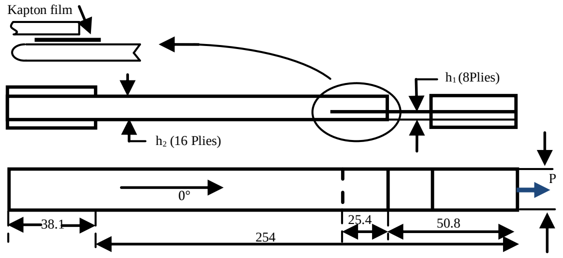

Fig. 1: Shape and size of CLS specimen (All dimensions in mm)

Cracked lap shear (CLS) test specimen made of prepregs high-performance unidirectional carbon fiber reinforced epoxy Hexcel (IM7 / 8552) used for fracture tests. The volume fraction of carbon fiber in the prepreg is 60%. The material properties used are shown in Table. 1, measured in a previous investigation [9], With a nominal ply thickness of 0.0626 mm, and the reference stacking sequence considered in the study is [90-0]8s.

Table 1: IM7-8552 Ply elastic properties

Property	**Mean value**
E1	171.42 (GPa)
E2 =E3	9.08 (GPa)
G12 =G13	5.29 (GPa)
G23	3.98 (GPa)
ν12 = ν13	0.32
ν23	0.5
X^T	2326.2 MPa
X^C	1200.1 MPa
Y^T	62.3 MPa
Y^c	200.8 MPa
S^L	92.3 MPa

As a results, The mode I and mode II energy release rates, GI and GII, were calculated under a plane stress condition. The variation of GI and GII for the CLS specimen is shown in Figure.2.

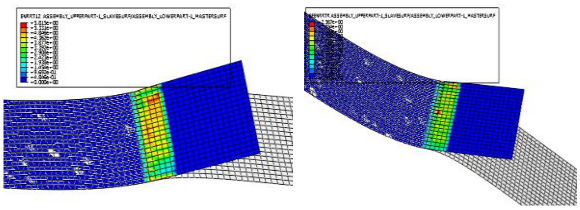

Fig. 2: Mode I & Mode II Energy release rate distributions

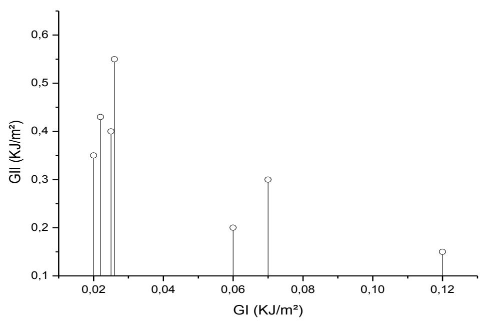

Fig. 3:Mixed-Mode fracture Energies envelope

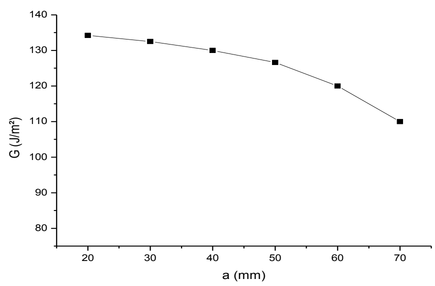

Fig. 4: Total Energy release rate a function of delamination length

Conclusion

Several numerical analysis were conducted for unidirectional CLS specimens in order to determine the critical energy release rate of Carbone/Epoxy composite laminates. The GI GII, mode I and mode II energy release rates were calculated from finite element analysis. The nominal strain energy release rate G is evaluated, and it's concluded that the coupling of the temperature causes an effect of accelerating or retarding the growth of delamination, depending on the loading regime. It was found that a linear fracture envelope may be suitable for a CLS specimen.

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