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Predicting the influence of discretely notched layers on fatigue crack growth in fibre metal laminates

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ABSTRACT

This paper presents an analytical model for fatigue crack growth prediction in Fibre Metal Laminates (FMLs) containing discretely notched layers. This model serves as a precursor in the development of a simplified prediction methodology for modelling the effect of load redistribution on a single crack in FMLs containing Multiple-site Damage (MSD) scenario. The model mainly focuses on capturing the influence of load distribution around discretely notched layers on the growth behaviour of an adjacent crack in a FML panel. The utilized approach in the model is the use of linear elastic fracture mechanics (LEFM) in conjunction with the principle of superposition and displacement compatibility. The proposed model is also validated using experimental data.

1. Introduction

The design philosophies used to ensure the integrity of aircraft structures over their lifetime have evolved over time. Currently, the design philosophy known as damage tolerance is recommended by the airworthiness regulations for the design of primary aircraft structures [1,2]. Goranson defines damage tolerance as *the ability of structure to sustain anticipated loads in the presence of fatigue, corrosion or accidental damage until such damage is detected, through inspections or malfunctions, and repaired* [1]. Although this definition is generally agreed upon, various interpretations on the implementation of damage tolerance exist, particularly related to the determination of inspection intervals for metallic and composite aircraft structures. For the case of metallic structures, inspection intervals are set based upon a detection window defined as the service life required for a damage to grow from a detectable size (based on inspection capabilities) to a critical size (based on limit load carrying capability). Central to this is the concept of slow-growth and the ability to predict damage growth behaviour. Conversely, a no-growth approach is typically adopted in composite structures whereby damage growth under service conditions is not permitted and inspection intervals are specified based on the statistical likelihood of damage-causing events. This paper focuses on the slow-growth interpretation most commonly adopted for metallic structures.

A flaw identified in the damage tolerance design philosophy is its compatibility with an indefinite structural life. The philosophy focuses on detection and repair of damages through continued maintenance; however, it does not define a limit to the validity of this approach in terms of structural life. As damage tolerance analyses tend to focus on the evolution of singular or isolated damage states, there is a risk that they will be invalidated over time due to the occurrence of widespread fatigue damage within a structure. The classic example of this occurring is the Aloha Airlines Flight 243 that on April 28,

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Nomenclature

a	half crack or delamination length (mm)
a_0	half saw-cut length (mm)
E_{FML}	Young's modulus of FML panel (MPa)
E_m	Young's modulus of metal layer (MPa)
E_f	Young's modulus of fibre layer (MPa)
E_{notch}	Young's modulus of the remaining material at the discretely notched area (MPa)
$F_{transfer}$	load transfer due to discretely notched layers (N)
G	strain energy release rate (kJ/m^2)
ΔG	strain energy release rate range (kJ/m^2)
j	number of interfaces (–)
K	stress intensity factor (MPa $\sqrt{mm}$)
K_∞	stress intensity factor due to far-field applied load (MPa $\sqrt{mm}$)
K_{br}	stress intensity factor due to bridging load (MPa $\sqrt{mm}$)
K_{total}	total stress intensity factor at the crack tip (MPa $\sqrt{mm}$)
$K_{redistribution}$	stress intensity factor due to far-field load and load redistribution (MPa $\sqrt{mm}$)
ΔK	stress intensity factor range (MPa $\sqrt{mm}$)
n_f	number of fibre layers (–)
n_m	number of metal layers (–)
N	number of fatigue cycles (–)
$P_{applied}$	total applied load (N)
$P_{applied,notch}$	total applied load on a laminate containing notches (N)
$P_{applied,M(T)}$	total applied load on a $M(T)$ laminate (N)
$P_{f,1}$	far-field load in bridging fibres (N)
$P_{f,2}$	far-field load in fibre layers except for bridging fibres (N)
P_m	far-field load in metal layers (N)
s_f	stress in the fibre layers (MPa)
s_{br}	bridging stress (MPa)
t_f	thickness of fibre layer (mm)
t_m	thickness metal layer (mm)
t_{FML}	thickness of FML laminate (mm)
v_∞	crack opening displacement due to far field load (mm)
v_{br}	crack opening displacement due to bridging load (mm)
W	width (mm)
x	position from the centre of a specimen (mm)
x_l	position of left notch edge (mm)
x_r	position of right notch edge (mm)
δ_f	elongation of the fibre layers (mm)
δ_{pp}	shear deformation of the fibre layers (mm)
ε_{yy}	strain distribution ahead of crack tip (mm/mm)
$\varepsilon_{yy,notch}$	strain distribution ahead of crack tip in a laminate containing notches (mm/mm)
$\varepsilon_{yy,M(T)}$	strain distribution ahead of crack tip in a $M(T)$ laminate (mm/mm)
$\sigma_{applied}$	total applied stress in laminate (MPa)
$\sigma_{m,applied}$	far field load in metal layers (MPa)
$\sigma_{f,applied}$	far field load in fibre layers (MPa)
$\sigma_{westergaard}$	Westergaard stress (MPa)
$\sigma_{westergaard,M(T)}$	Westergaard stress for $M(T)$ specimen (MPa)
σ_{yy}	Westergaard stress distribution ahead of crack tip (MPa)
σ_{notch}	stress at the discretely notched area (MPa)

1988 suffered explosive decompression in flight due to the sudden link-up of small fatigue cracks at adjacent rivet holes in a longitudinal lap joint [3].

To combat the possibility of another failure due to widespread fatigue damage, the aircraft regulatory authorities have revised the regulations in 2010 with new rules pertaining to Aging airplane safety *widespread fatigue damage* [4]. This revision included the definition of a Limit of Validity (LOV) of the engineering data (including the damage tolerance analyses) which support the continuing structural maintenance of an aircraft. These new regulations effectively require the aircraft OEMs to establish a firm limit to the operational life of a given aircraft type (within a given type certificate) that is substantiated with test evidence and analysis. As a result, there is a renewed interest in robust and efficient analysis methods for predicting Widespread Fatigue Damage (WFD) and its effects.