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Aeroelastic assessment of cracked composite plate by means of fully coupled finite element and Doublet Lattice Method

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ABSTRACT

This paper presents an investigation on flutter speed of cracked composite plates. This work is divided into two sections: (a) variation of crack length at a fixed location on the plate, and (b) variation of crack location on the plate with a fixed crack length, modelled as a unidirectional composite for 0^0,90^0 and 135^0 orientations. Mori-Tanaka homogenization model is applied to obtain the effective composite constitutive properties as the function of fiber and matrix volume fraction. Doublet Lattice Method (DLM) is used to calculate the unsteady aerodynamic forces, i.e., lift distributions. It is found that the existence of small crack ratio on the composite plate (less than 0.4) has triggered an increment of the flutter speed. To support this statement, flutter response modes for each crack ratio are plotted, where the structure appears to be more stiffened than the undamaged plate. However, the crack results in the reduction of flutter speed when the crack ratio reaches 0.5. For the crack location assessment, the flutter speed increases as the crack location moves from the root to the tip due to the reduction of flutter frequency. The results show a good agreement with the validation using Strip Theory considering unsteady aerodynamics.

1. Introduction

In this paper, computational investigations of the flutter effect to several cracked composite plates are performed. It is believed that the existence of crack will affect the stiffness of the structure [1]. There is a work that investigates the stiffness effect on symmetric laminates with arbitrary sequence [2]. The reduction in transverse and shear stiffness of the laminate as a function of the crack density in one ply was estimated by deriving an analytical solution. Thus, the accuracy in predicting the stress redistribution, from a cracked ply to the rest of the laminate has been achieved. Hence, it is a logical reason to investigate the flutter speed of cracked composite structures since one of the parameters that could affect the flutter speed estimation is the stiffness system.

Flutter is an instability problem due to structural vibration exerted by the aerodynamic load. Flutter often categorised as a self-excitation phenomenon, as the aerodynamic load is a function of the structural dynamic responses. A critical speed in which the structural vibration could lead to a catastrophic failure is called 'critical flutter speed'. One of the most well-known examples of flutter vibration leading to a catastrophic failure is well presented in the incident of the Tacoma bridge collapse on the 7th of November 1940 [3].

It was reported that 42 mph speed of wind had excited several vibration modes on that day [4]. The dominant mode was moving vertically with a node at midspan and thus changed to torsional motion with a node at midspan abruptly. Within 4 s, the vibration amplitude has twisted the bridge about 45^0 before it collapse.

The existence of crack will affect the stiffness distributions as discussed in the literature. It is also a requirement to determine the flutter boundary by considering the structural stiffness. Castravete and Ibrahim demonstrated that the stiffness significantly affects the flutter boundary [5]. This evidence has attracted attention to investigating the flutter boundary when there is an existence of crack on the structure.

In studying the circumstance, one of the aircraft crash incidents of North American P-51D Mustang that related to the event is referred as an example. The racing aircraft which also known as "The Galloping Ghost" crashed at the National Championship Air Races in Reno/Stead Airport, Nevada, USA. The technical investigation report by National Transportation Safety Board (NTSB) revealed that the existing fatigue crack in one screw caused the reduction of elevator trim tab stiffness [6]. This situation had triggered aerodynamics flutter to occur at racing speed.

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There are some works reported regarding supersonic flutter on damaged composite such that shear deformable laminated composite flat panels by Birman and Librescu [7], microstructural continuum damage by Pidaparti [8] and, Pidaparti and Chang [9]. The coupling between two-dimensional static aerodynamic technique and a higher order transverse shear deformation theory for the structural plate model were performed in [7]. Aerodynamic models of Piston theory were applied, and the structures were modelled based on the damage mechanics theory with an internal state variable to mark damage characteristic in the material [8,9].

There is a work that model a crack on a composite panel using XFEM at the supersonic region presented in [10]. A rectangular plate made of a Functionally Graded Material (FGM) is considered in this work as an advanced composite structure. The recent investigation of interaction between cracks on flutter was presented by Viola et al. [11]. The numerical flutter analysis was performed on a multi-cracked Euler-Bernoulli beams under subtangential force as the non-conservative dynamic load.

Some researchers applied the probabilistic approach to assess the flutter failure of a composite structure with crack in subsonic flow. The application of Monte Carlo simulation in [12] and Polynomial Chaos Expansion method in [13] show the statistical studies of flutter with the presence of multiple damage uncertainties.

Based on the overview, it can be seen that there is a lack of publication on the flutter of cracked composite. Moreover, at subsonic regime, to the author's knowledge, only Wang et al. [14] studied it by means of analytical/semi-computational model. As most transport and light aircrafts are operating in subsonic regime, thus it is considered a great benefit to investigate the flutter effect on cracked composite within this airspeed regime.

In the present work, a novel implementation of fully computational approach to investigate the flutter on cracked composite within subsonic regime is elaborated. Laminated finite element is used to model the composite structure. The load is modelled as unsteady aerodynamic load in frequency domain by means of Doublet Lattice Method (DLM). The *pk*-method is applied to obtain the flutter solution. In the following sections, the general overview of the computational methods used are presented.

2. Flutter speed determination

Flutter is defined as a state or phenomenon of flight instability which can cause structural failure due to the unfavourable interaction of aerodynamics, elastic, and inertia forces [15]. Flutter can deform an aircraft due to dynamics instability. In practice, structural damping, g versus velocity, V for each mode shape is plotted to determine the flutter speed graphically. Based on the Federal Aviation Administration Regulations in [16], the required structural damping, (g) value for plotting Fig. 1 must exceed more than 3%, $g > +0.03$ in the unstable region so that the plot can be stated as in flutter region.

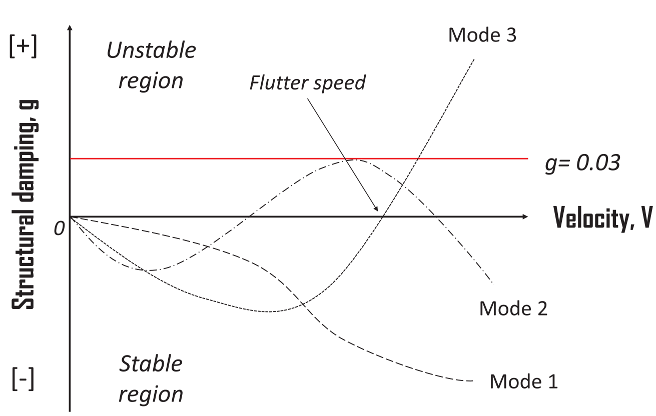

Fig. 1. Structural damping graph guided by FAA (2004).

The procedure has been performed by Nissim and Gilyard [17] to estimate the flutter speed experimentally by using the parameter identification technique. It is pointed out that there is an issue of difficulty when the 'exact' analytical scheme to solve the flutter equations. Since the damping merged with aerodynamic terms only, the system is assumed to be an undamped structural system. This is the reason why the system excitation at zero damping that led to the zero dynamic pressure could not be performed and hence will trigger the responses at resonance become infinite values. To solve this, the 3% of structural damping is assumed and at the same time, the responses of the 'exact system' is calculated. When this procedure objective is achieved, the flutter speed can be determined at zero structural damping.

Fig. 1 is referred as an explanation for the flutter phenomenon in graphical presentation. Mode 1 moves towards the instability region in the first place, but the plot free from the unstable region as the speed is increasing. Mode 2 crosses the velocity axis where the structural damping is zero. Since the plot of Mode 2 still has not exceeded $g = 0.03$, the structure is in a safe zone. Mode 3 crosses the velocity axis where the structural damping is zero and has surpassed the limitation of $g = 0.03$. It is concluded that Mode 3 is the most dangerous state where the flutter is expected to happen.

In this study, the flutter speed for each composite structure is determined by using this technique. Several parameters are concerned to be investigated; the unidirectional composite angle, θ, crack ratio, η and the dimensionless crack location, ξ_c.

3. Mean field homogenization

In this part, a process called homogenization which is considered to represent the composite material properties is performed. Representative volume element (RVE) is used to represent the microscale of the structure. Solving the mesoscale iteration at every guess, the RVE is computed, and then, the information is passed to macroscale. The homogenization procedures are explained more in [18–20].

The objective of applying this process is to estimate the stresses and strains as the matrix and the fibers are mixed. In this study, the homogenization of composite structures is carried out by applying the Eshelby method. Fig. 2 shows the schematic diagram of homogenization based on the Eshelby method presented in [21,22].

Fig. 2(a) shows an initial unstressed elastic homogeneous material. A visualization of a cutting section called as inclusion is assumed to this structure, presented as the circle. The inclusion is presumed encounters a shape change free behaviour; causing the transformation strain $\boldsymbol{\varepsilon}_{ij}^{T}$ in Fig. 2(b) from the constraining matrix.

Assuming the strain is uniform within the inclusion, the stress in the inclusion, $\boldsymbol{\sigma}_{ij}^{I}$ is estimated using Eq. (1).

$$\boldsymbol{\sigma}_{ij}^{I} = \boldsymbol{C}_{ijkl}^{M}(\boldsymbol{\varepsilon}_{kl}^{C} - \boldsymbol{\varepsilon}_{kl}^{T}) \tag{1}$$

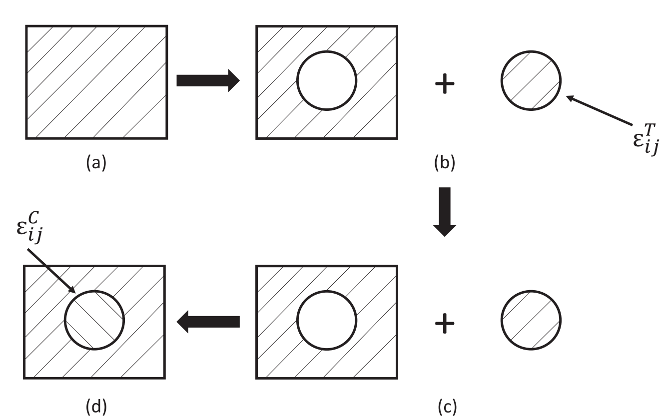

Fig. 2. Schematic diagram of homogenization based on the Eshelby method.