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Diagnosis of Modal Characteristics of Aircraft Wing Model subjected to Irregularities

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ABSTRACT: *Aircraft wings are subjected to various forces and structural stresses during it's flying as well as static condition. The force of gravity is influential in the static condition. Any acceleration or deceleration during flight enhances the forces and structural stresses on the wings, which subsequently are being transmitted to the fuselage structure. The wings of the aircraft adhere to particular modal characteristics i.e. natural frequency, mode shape, damping etc. when subjected to different loading conditions. As the behavior of the modal parameters are dependent on the elastic property of the material of the wing and the elastic properties are dependent on the structural integrity of the wing, any irregularities like crack or damage developed in the course of action lead to the alteration of the modal characteristics. The deviations in the modal characteristics facilitate the researcher to diagnose the crack position and severity utilizing the existing reverse engineering techniques. In this paper, a simple model of the small aircraft wing is subjected to modal analysis by numerical method i.e. Finite Element Method (FEM) as well as experimental method. The natural frequencies of vibration for first three modes extracted from both the methods show close proximity to each other. Moreover, the natural frequencies of the damaged wing are lowered in contrast to the undamaged wing. The amplified distorted shapes of the wing corresponding to different mode exhibit significant distinction between the damaged and undamaged wing structures.*

Keywords*: Modal characteristics, natural frequency, crack, finite element method, experimental method*

I. INTRODUCTION

Vibration-based diagnosis methods are very effective non-destructive testing methods, which are based on the fact that any changes introduced in a structure (including damage or crack) change its physical properties like stiffness and this in turn changes the structural vibration response. For most aerodynamic applications, modal characteristics of the structure and especially its lower resonant frequencies, serve as an important tool to investigate the irregularities developed the aircraft and assures the aircraft's safe service life. Chinchalkar [1] has developed a generalized numerical method for fault finding using finite element approach. His approach is based on the measurement of first three natural frequencies of the cracked beam. The developed method of fault detection accommodates different boundary conditions and having wide variations in crack depth. Douka et al. [2] have studied the non-linear dynamic behavior of a cantilever beam both theoretically and experimentally. They have analyzed both the simulated and experimental response data by applying empirical mode decomposition and Hilbert transform method. They have concluded that the developed methodology can accurately analyze the nonlinearities caused by the presence of a breathing crack. Prasad et al. [3] have investigated the effect of location of crack from free end to fixed end in a vibrating cantilever beam. They compared and analyzed crack growth rate at different frequencies using the experimental setup. Rezaee et al. [4] have used perturbation method for analysis of vibration of a simply supported beam with breathing crack. From the analysis it is observed that for a given crack location on the beam structure with the increase in the relative crack depth the stiffness of the beam decreases with time. Wang et al. [5] have studied a composite cantilever having a surface crack and found that the variation in the modal response depends on two parameters i.e. crack location and material properties. They have concluded that the change in frequency can be effectively used to locate the crack position and measure its severities. Owolabi et al. [6] have carried out experimental investigations of crack location and crack intensity for fixed beams and simply supported beams made of Aluminum. They have measured the changes in the first three natural frequencies and the corresponding amplitudes to forecast the crack in a structure. Gounaris et al. [7] have presented a crack identification method in beam structures assuming the crack to be open and using eigenmodes of the structure. During the investigation, they have found out the relationship between the crack parameters and modal response. Finally, they have checked the authenticity of their method by comparing the eigenmodes for the damaged and undamaged beam in pre-plotted graphs. Shen et al. [8] have proposed a crack diagnostic procedure by measuring the natural frequencies and mode shapes. They have checked the robustness of their proposed method from the simulation results of a simply supported Bernoulli-Euler beam with one-side or symmetric crack. Trendafilova et al. [9] have adopted principal component analysis and simple pattern recognition to diagnose the fault in vibrating structure using frequency response function of the healthy and damaged structures as input parameters. The vibration response of the structure is used as a tool to detect the damage in a scaled aircraft wing modeled by finite element method. Ciesielski et al. [10] have compared the fatigue crack growth rate obtained from macroscopic and microscopic analysis of the fatigue fracture surfaces by rendering the PZL I-22 Iryda aircraft to real operational conditions. The results are referred as to the development of cracks in wing skin under stable crack growth conditions. Fujimoto & Sekine [11] have identified the locations and shapes of crack repaired with bonded FRP composite patches in aircraft structural panels by converging the measured in plane strain range and calculated in plane strain range. The characteristics of crack are affected by strain measurement points, position of strain measurement plane and measurement of errors in plane strain ranges. Koski et al. [12] have used the flight measurement to determine the life span of aircraft's wing subjected to loading cycles. The flight measurement results obtained by FE method and engineering judgment together help to observe the multi-axial fatigue effects. Liao et al. [13] have investigated the maximum depth of exfoliation damage by ultrasonic non-destructive inspection by performing the fatigue tests under fully reversed constant amplitude loading and fractographic analysis. Molent [14] has compared the failures of F-111 aircraft wing with the failures of combat type aircraft operated by RAAF due to the initiation of discontinuities. These failures are found to be dependent upon several design and production factors along with the original discontinuity size. Barter et al. [15] have investigated the metallic air frame structures with the fatigue cracks initiated due to the presence of discontinuities observed from the coupon, component, full scale tests, service aircraft, commercial transport aircraft and military aircraft. In this paper, simple aluminium wings shaped like structure without crack and with crack are subjected to vibration to arbitrate the first three natural frequencies by experimental and numerical methods. The deformed shapes of the wing structure are also illustrated.

II. MATHEMATICAL MODELING

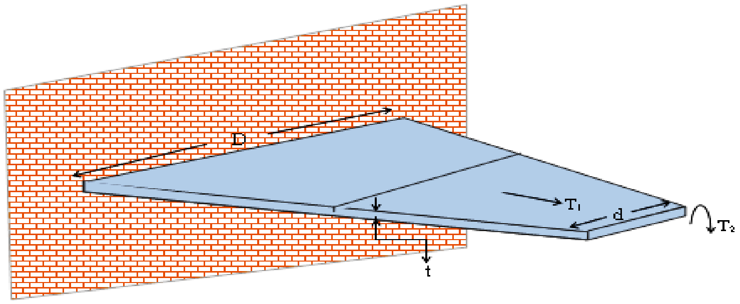

Fig.1 Schematic diagram of cracked aluminium wing shaped structure

A cracked aluminium wing shaped structure with a crack depth 'b_1' on span length of 'L', end diameters 'D' & 'd' and thickness 't' is utilized for the ongoing investigations. The beams are encountered with axial force 'T_1'and bending moment 'T_2' as shown in fig. 1. The presence of crack in the wing has originated a local stiffness, which can be formulated in a square matrix of second order due to the assumption of two degree of freedom system.

The strain energy release rate 'U' at the cracked position assuming plain strain condition will be,

$$U=\frac{1-v^2}{E}(J_{11}+J_{12})^2 \qquad (1)$$

$J_{11,}$ J_{12} are the stress intensity factors of first mode of vibration for T_1 and T_2 respectively.

The strain energy can also be formulated as,

$$U=\int_0^{b_1}\frac{\partial T}{\partial b}db \qquad (2)$$

The flexibility influence co-efficient 'B_{ij}' fundamentally can be expressed as,

$$B_{ij}=\frac{\partial^2}{\partial T_i \partial T_j}\int_0^{b_1}T(b)db \qquad (3)$$

The localized stiffness matrix can be determined by inverting the compliance matrix i.e.

$$K=\begin{bmatrix}K_{11} & K_{12}\\ K_{21} & K_{22}\end{bmatrix}=\begin{bmatrix}B_{11} & B_{12}\\ B_{21} & B_{22}\end{bmatrix}^{-1} \qquad (4)$$

In Eq. (4), K_{ij} & B_{ij} are elements of i^{th} row and j^{th} column of stiffness and compliance matrix respectively.

III. METHODOLOGIES

3.1. Experimental Method

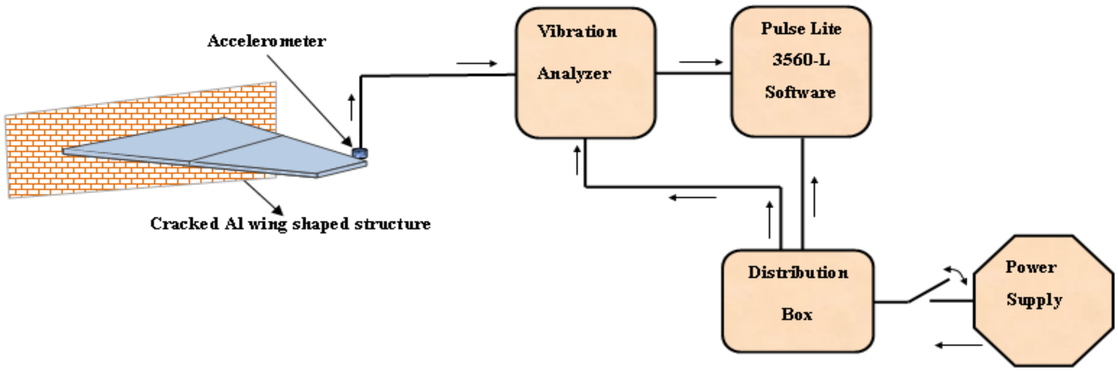

Fig.2 Schematic diagram of experimental setup

The experimental set up as shown in fig.2 consists of constituents like accelerometer, vibration analyzer, pulse lab software for 3560-L device integrated with a computer etc. are employed for the experimentations to be performed. The wing shaped structure with span length 1.5m, end diameters as 0.6m & 0.2m and thickness as 0.01m made up of aluminium is used to determine three natural frequencies corresponding to first three mode of vibration. The readings are noted for wing structure without crack and with crack at the middle of the structure. The crack is introduced manually in to the wing shaped structure.

3.2. Numerical Method

The numerical method implemented for investigation is finite element analysis (FEA), for which commercially available software 'ANSYS' of version 10 is used. To begin with the analysis, wing shaped structure with the dimensions mentioned in experimental method is modelled without crack & with crack of crack depths 3mm, 4mm, 5mm at middle position using the 'Modelling' module of ANSYS. The material properties like Young's modulus, Poisson's ratio and density of aluminium are introduced as 70 Gpa, 0.35 and 2700 kg/m^3 respectively.

The first three natural frequencies of vibration of the beams with crack and without crack are extracted using the 'Modal Analysis' module of ANSYS. The deformed shapes corresponding to each mode of vibration are also depicted.

IV. RESULTS & DISCUSSION

The values of natural frequencies obtained from experimental method and numerical method are categorized in tables 1-4. The deformed shapes are illustrated in figures 3-4.
'f_n' and 'f_e' are numerical and experimental natural frequencies respectively.

Table.1 Wing without

Sl No.	f_e	f_n	% of error
1st	4.904	5.0875	3.6
2nd	24.805	25.839	4
3rd	65.536	68.125	3.8

Table.2 Wing with crack at middle of crack depth 3mm

Sl No.	f_e	f_n	% of error
1st	4.098	5.0789	3.9
2nd	25.840	25.821	3.8
3rd	65.525	68.113	3.8

Table.3 Wing with crack at middle of crack depth 4mm

Sl No.	f_e	f_n	% of error
1st	4.904	5.0979	3.8
2nd	24.832	25.867	4
3rd	65.352	68.146	4.1

Table.4 Wing with crack at middle of crack depth 5mm

Sl No.	f_e	f_n	% of error
1st	4.897	5.0952	3.9
2nd	24.794	25.854	4.1
3rd	65.611	68.132	3.7

Deformed Shapes of Wing Structure

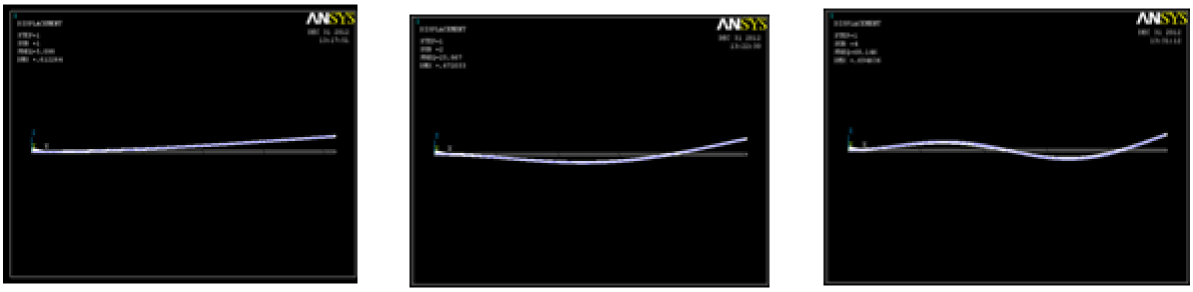

Fig.3 Deformed shapes of first three modes of Wing structure without crack

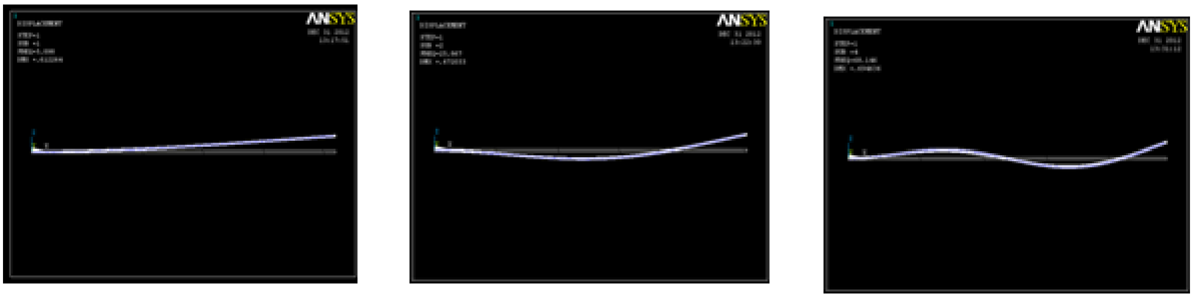

Fig.4 Deformed shapes of first three modes of Wing structure with crack at the middle having crack depth 4mm

Table 1-4 demonstrate the natural frequencies of wing like structure without crack and with crack obtained by numerical method and experimental method. The crack depth is varied from 3mm to 5mm and the position of crack is considered at the middle of the wing shaped structure. The divergence of experimental natural frequencies from that of numerical natural frequencies is estimated in terms of percentage of error for each case of vibration wing structure. Fig. 3 and 4 illustrates the various deformed shapes of wing structure. The natural frequencies and deformed shapes of the vibrating beams with different position of crack are also extracted.

The results from table 1-4 present a very close proximity between the numerical and experimental values of natural frequencies. The deviation lies within 3.5 to 4.5 percent. Though the deformed shapes of first three modes resemble each other for wing shaped structure with and without crack, the magnified views exhibit significant distinction between them. Importantly, natural frequencies get diminished with the introduction of crack and with the increase in crack depth for the cracked wing structure. Moreover, the natural frequencies increase with the crack position move away from the fixed end.

V. CONCLUSION

The reconditioning of various wing structures is possible only when the cracks are thoroughly diagnosed. The deviations of the natural frequencies and deformed shapes of the cracked wing structures with respect to the undamaged wing structures played a decisive role to unearth the crack characteristics employing various reverse engineering techniques.

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