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Acoustic detection of invisible damage in aircraft composite panels

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Abstract

The wing and body panels of modern commercial and military aircraft often consist of a three-layer structure in which two thin skins of fibre-reinforced composite or of aluminium are held apart by a much thicker core consisting of a honeycomb structure made from either folded paper-like material impregnated with aramid resin or from thin, folded aluminium sheet. A major maintenance inspection problem arises from the fact that impact by a heavy soft object has the potential to deflect the skin and damage the core, after which the skin can return to its original shape so that the defect is nearly invisible. This paper gives details of an acoustic inspection system that can reveal such damage and provide information on its nature and size using a hand-held "pitch-catch" device that can be scanned over the suspected area to produce a visual display on a computer screen. The whole system operates in the frequency range 10–30 kHz and embedded programs provide optimal examination procedures.

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1. Introduction

Modern commercial and military aircraft often employ a three-layer sandwich construction for both wing and fuselage panels in order to maximise functionality and strength while minimising weight. A typical sandwich panel consists of two stiff skin layers 1–5 mm in thickness separated by a light core 10–50 mm thick. These panels usually have skins formed from laminated multi-ply composites of either carbon-fibre or fibre-glass in a matrix of polymer resin, although other skin materials such as aluminium are sometimes used. Sandwich panel cores are stiff, lightweight, and are usually either a honeycomb structure or a closed-cell foam. A very common core used in modern aircraft, commercially known as Nomex, is made from a paper-like material folded into a honeycomb shape and impregnated with an aramid resin to add strength, though some panels use aluminium for cores as well as for skins.

Composite sandwich panels are very effective in terms of strength-for-weight, but can suffer from a potentially serious problem. If the panel is subjected to an impact by a low velocity object such as a worker's dropped tool, or even something soft such as a tyre fragment thrown up from the runway or collision with a bird in flight, then the skin at the impact site can be deflected inwards without noticeable cracking and cause crumpling or fracture within the core, as shown by the example in Fig. 1. After the impact the skin is free to restore itself back to its original shape so that the damage can be almost invisible. This form of damage is often referred to as Barely Visible Impact Damage, or BVID. Even a relatively small amount of crumpling of a honeycomb core can significantly affect the local stiffness of the panel, so that it is desirable to identify any instances of impacts and verify that the structure is still adequately load-bearing.

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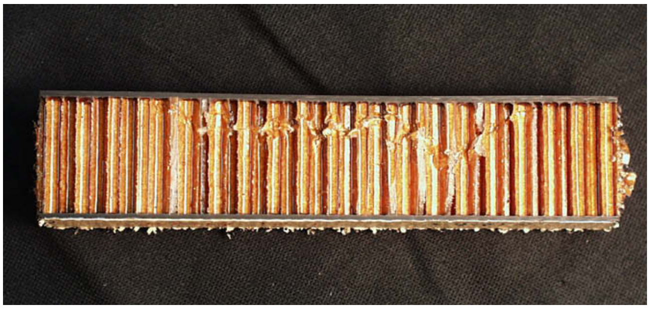

Fig. 1. Cross-section showing damage in a panel with 1.6 mm fibre-reinforced skins and 24 mm Nomex core after impact by a rubber-tipped metal "tup" of mass 2 kg and diameter 25 mm dropped from a height of about 1 m. Damage from such an impact typically begins for a drop height of about 50 cm. Note the irregularity of the damage depth.

Traditionally the most common method for inspection of aircraft used to be visual inspection by ground staff, with light tapping of any suspected areas with a hard, blunt instrument to give an indication of the underlying structure from the sound of the tap. This is called a "coin tap" test and, although originally subjective as the name implies, was later developed into an automated analysis system. This automated system involves a controlled light impact on the panel, with monitoring of both impact force and panel deflection as functions of time to give a measure of its local mechanical stiffness [1]. Other methods have been also been devised to inspect aircraft, such as the use of X-rays, thermography, dye-penetrants, eddy current measurements, shearography, and magnetic particles [2] but by far the most commonly implemented methods for testing composite panels are those using acoustic techniques.

It might have been expected that the best way to examine composite panels would be to use ultrasound, and much has been published on ultrasonic techniques [3], but the great difference in acoustic impedance between the skin and core creates considerable difficulties and, in addition, it is often not possible to access both sides of an assembled panel to utilise through-transmission techniques. This has led to the development of techniques for lateral wave propagation measurements at lower frequencies. One such family of commercially available acoustic sensors, developed in the 1980s, is called "pitch-catch" probes [4]. Unlike ultrasonic techniques, the basic operation of acoustic pitch-catch probes, originally developed by Lange [5,6] in Russia, has received relatively little recent published attention, although there are several commercial embodiments of the technique available. What is presented has usually been empirical in nature, rather than addressing basic issues. Recently CSIRO in Australia has developed an extended version of an acoustic pitch-catch probe housed within a hand-held device named the "Bandicoot" after a small Australian marsupial animal with a long sensitive nose that it uses to detect small insects for food. The name, when written "baNDIcoot" also incorporates the acronym "NDI" for "Non-Destructive Inspection". This instrument and its associated software are attracting considerable interest from aircraft manufacturers and operators around the world. The present paper aims to provide some insight into the manner in which such probes operate and to show how different types of damage can be located and identified.

2. Waves and vibrations in composite panels

The transducers used in the baNDIcoot are sensitive only to vibrations normal to the panel surface, but there are several types of vibrational waves that can contribute to this motion. Let us consider these wave types in turn. All are complicated by the fact that the panel is not a simple plate of uniform composition but contains a light core sandwiched between its two skins. Even for a homogeneous plate, the detailed equations of motion are extremely complicated if the wavelength is not long compared with the plate thickness [7], and analysis of a composite panel is more complex even if this simplifying assumption can be made [8], but an adequate approximation has been given by Thwaites and Clark [9].

Under transverse stress the panel will deform in whichever way requires least elastic energy. If the wavelength is greater than about five times the panel-thickness, then the panel deforms by bending. It is this sort of deformation that the panels are designed to resist, the two skins providing a large stiffness against the stretching and compression necessarily involved in the wave, with the core serving to hold the skins apart and so magnify the effect. If the skin thickness is h and the core thickness H and we make the assumption that the Young's modulus of the core material is essentially zero in the plane of the core but large in the normal direction, then bending waves simply stretch or compress the skins, with their separation remaining constant. To a first approximation the bending stiffness is then $E_s hH^2/2(1-\sigma_s^2)$, where E_s is the Young's modulus for stretching of the skin in the surface plane and σ_s is its Poisson's ratio, while the moving mass per unit area is $\rho_s h + \rho_c H$, where ρ_s is the skin density and ρ_c is the density of the core. Solution of the standard bending wave equation then gives the speed of a bending wave of angular frequency ω as

$$c_b \approx \omega^{1/2}\left[\frac{E_s hH^2}{2(1-\sigma_s^2)(\rho_s h+\rho_c H)}\right]^{1/4}, \tag{1}$$

Note that the speed of these bending waves increases proportionally to the square root of the frequency.

If the wavelength is less than about five times the panel-thickness, then the deformation becomes a shear wave in the panel. The shear stiffness is simply the combined stiffness of core and skins, and the mass is also the combined mass, and the speed of shear waves is thus

$$c_s \approx \left(\frac{G_c H + 2G_s h}{\rho_s h + \rho_c H}\right)^{1/2}, \tag{2}$$