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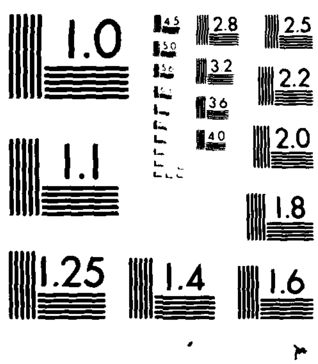

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by

R. JONES

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(10) R. JONES

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SUMMARY

In recent years several advanced finite element methods have been developed for the analysis of laminated composites; these take into account the membrane, bending, and interlaminar stresses. Similarly, finite element methods have also been developed for the analysis of structures repaired with a bonded overlay of fibre composite material. The present paper discusses these methods and indicates how the finite element method developed for the analysis of structural repairs is connected to those methods specifically developed for the analysis of composite laminates.

POSTAL ADDRESS: Chief Superintendent, Aeronautical Research Laboratories, Box 4331, P.O., Melbourne, Victoria, 3001, Australia.

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ABSTRACT

In recent years several advanced finite element methods have been developed for the analysis of laminated composites; these take into account the membrane, bending, and interlaminar stresses. Similarly, finite element methods have also been developed for the analysis of structures repaired with a bonded overlay of fibre composite material. The present paper discusses these methods and indicates how the finite element method developed for the analysis of structural repairs is connected to those methods specifically developed for the analysis of composite laminates.

CONTENTS

NOTATION

G_a	Shear modulus of the adhesive.
G	Shear modulus of a composite ply.
G_0, G_s	Transverse shear moduli of the overlay and sheet respectively.
τ_{zx}, τ_{zy}	Transverse shear stresses.
t_s, t_a, t_0	Thicknesses of the sheet, adhesive and overlay respectively.
u^l, v^l	In plane displacements in the *lth* ply.
t	Distance between mid-surface of adjacent plies.
u_0, v_0; u_s, v_s	In plane displacements in the overlay and the sheet respectively.

1. INTRODUCTION

In recent years a number of different approaches [1–14] have been developed for the analysis of laminated composites. We begin by describing these methods in some detail paying particular attention to the need to accurately determine the interlaminar stresses developed in a laminate. Following this discussion we examine the methods developed for the analysis of adhesively bonded metallic, or fibre composite, repairs to damaged structures [27–52].

We then show that the mathematical models developed in [2, 3] for the analysis of laminated composites subjected to in-plane loading only, and the models developed in [27, 28, 29] for the analysis of bonded repair schemes are very similar and may be made identical.

2. INTERLAMINAR STRESSES

In classical laminate theory no account is taken of the interlaminar shear stresses τ_{zx} and τ_{zy} or the peel stresses σ_z. Rather, only the stresses in the plane of the laminate, i.e. σ_x, σ_y, and τ_{xy}, are considered. Accordingly classical laminate theory is incapable of providing predictions for some of the stresses that may causes failure of a composite laminate. One of the earliest attempts to overcome this shortcoming, inherent in classical laminate theory, was the work of Pipes and Pagano [1] who undertook a detailed three-dimensional stress analysis. Following this work Levy and co-workers [2, 3] developed a finite element method, based to some extent on the earlier work of Puppo and Evensen [4], which took the interlaminar shear stresses developed in the composite into account. This method, which is applicable for in-plane loading only, separates the membrane and interlaminar properties of a laminated composite by using alternating orthotropic membrane elements and isotropic shear elements as shown in Figure 1. The orthotropic segments carry in-plane stresses only while the shear segments only carry the interlaminar (i.e. transverse) shear stresses; for full detai's see [2, 3].

In the shear segments the transverse shear stresses τ_{zx} and τ_{zy} are related to the displacements u^l, v^l, and u^{l-1}, v^{l-1} in the lth and $l-1$th layers respectively by the formulae

$$\tau_{zx} = (u^{l-1} - u^l)/G\,t \tag{1}$$

$$\tau_{zy} = (v^{l-1} - v^l)/G\,t \tag{2}$$

where t is the distance between the midsurface of the lth and $l-1$th layers. Here we have used the notation given in [3] with G being the transverse shear modulus of the composite. Indeed in [2, 3] it was assumed that $G_{zx} = G_{zy} = G$. This approach is very simple to use and, as shown in [3], is easily extended so as to allow for inelastic effects. Furthermore in the case of a $(\pm 45)_s$ laminate subjected to uniaxial strain it was shown to give identical results to those given in [1] which were obtained from a full three-dimensional analysis.

Whilst the above methods were being developed research was also underway into developing sophisticated plate bending elements [6, 7] which allow for the transverse shear deformation occurring in laminated composites. These elements were shown to give excellent agreement with known exact solutions obtained from reference [8]. Indeed these approaches have been widely accepted. They have been built into a variety of commercially available packages and implemented on the main frame computers at a number of aerospace companies, e.g. [9].

Although these approaches have been very successful they nevertheless retain the disadvantage, inherent in the analysis presented in [2, 3, 6, 7], that they neglect the "peel" stress σ_z. This stress may in certain circumstances be a prime design variable. Hence several fully three-dimensional composite elements have been developed. Some of these elements are described in references [5, 10]. Most work very well and yield results in agreement with known analytical solutions. One disadvantage of these three-dimensional elements is the size and the band width of the stiffness matrix (and hence the cost of an analysis) which results when attempting to model

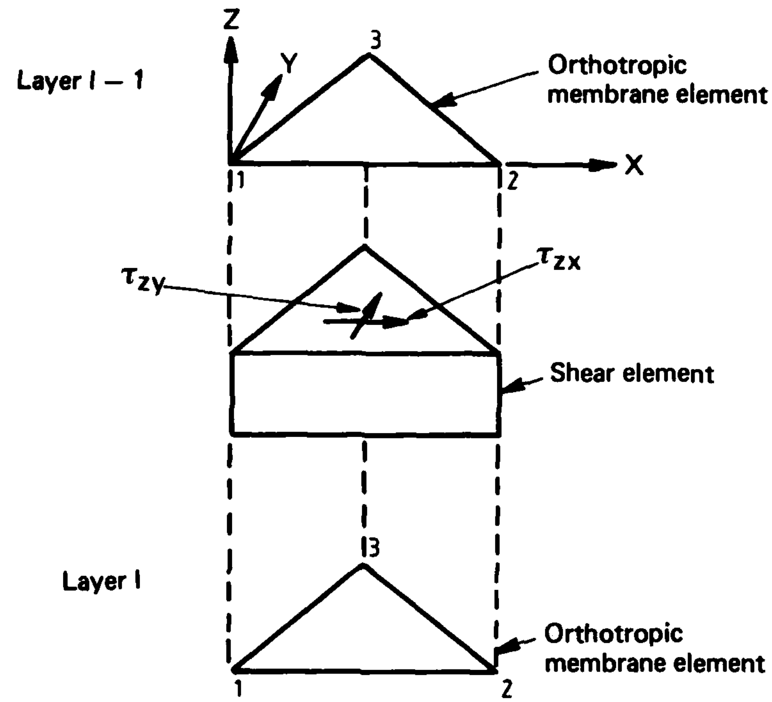

FIG. 1 COMPOSITE ELEMENT

a realistic fibre composite structure. Hence there is a tendency to use elements similar to those developed in [6, 7] whenever possible in preference to the fully three-dimensional elements. To some extent this trend has been encouraged by references [11, 12] which indicate that, even when the peel stresses are important, they may in certain circumstances be accurately estimated from a two-dimensional stress analysis. Furthermore, although it has been shown that significant interlaminar stresses exist around holes [3, 13, 14, 15], it has also been established [16, 17, 18, 19] that a workable failure criterion exists for the failure of composites containing holes or cracks and that this criterion only needs a knowledge of the two-dimensional stress state around the crack or hole. However this is not thought to be true for the growth of disbonds between the layers of a laminated composite. Indeed references [20, 21], which specifically deal with the definition and modelling of flaws in composite laminates, when dealing with disbonds pay particular attention to the interlaminar stresses. As a result the behaviour of disbonds in composite laminates requires, unlike cracks or holes, a detailed knowledge of the interlaminar stresses. This in turn requires either the fully three-dimensional elements [5, 10], the plate bending elements [6, 7]; or the "shear" elements [2, 3] to be used. However the simple "shear" elements should only be used when the stresses in the laminate are primarily membrane stresses. This is often the case in the wing skins of aerospace vehicles.

Whilst, in the above discussion, a number of elements have been described, these are by no means all of the available elements. Other approaches worthy of note are given in references [22, 23, 24, 25, 26]. Of these the work of Reddy [24] is particularly valuable in as much as it compares the effects that reduced integration, mesh size and changing interpolation functions within an element have on the stresses for the case of plate bending elements with shear deformation.

3. ADHESIVE ELEMENT

A similar approach to that of Levy *et al.* [2, 3] was used in [27, 28, 29] for the development of an "adhesive" element for the analysis of adhesively bonded repairs to thin metal or composite sheats and where the overlay (repair) is a composite laminate.

In this approach it is assumed that the in-plane displacements vary quadratically through the thickness of the sheet and the overlay and linearly through the thickness of the adhesive. The resultant distribution of the transverse shear stresses through the structure is identical with that used in [30] for the analysis of sandwich structures with thin faces and is in close agreement with the stress distribution obtained in [1] from a full three-dimensional stress analysis. With these assumptions [27, 28, 29] develop the following expressions for the adhesive shear stresses τ_{zx} and τ_{zy}, for the case of a repair on one side of the sheet only, viz.,

$$\tau_{zx} = (u_0 - u_s)/(t_a/G_a + 3t_0/8G_0 + 3t_s/8G_s) \tag{3}$$

$$\tau_{zy} = (v_0 - v_s)/(t_a/G_a + 3t_0/8G_0 + 3t_s/8G_s) \tag{4}$$

Here t_a, t_0 and t_s are the thicknesses of the adhesive, overlay and sheet respectively while, to enable a comparison with equations (1) and (2), we have considered the case when $G_{zx} = G_{zy} = G_0$ in the overlay, $G_{zx} = G_{zy} = G_s$ in the sheet and have defined G_a as the shear modulus of the adhesive. In addition we have denoted the (x, y) displacements in the sheet as (u_s, v_s) and the corresponding displacements in the repair as (u_0, v_0).

This approach has been shown in [29] to give strains, on the surface of a repair to a circular cut out and a crack in a metallic sheet, in excellent agreement with measured strains. It has also been used [31, 32] in the design of repairs to the lower wing skin of Mirage III-o aircraft in service with the Royal Australian Air Force. These repairs have been installed on the Mirage fleet and to date the stresses predicted in [31, 32] have been validated by the inservice performance of the repair and by a series of laboratory tests [32] carried out at the Aeronautical Research Laboratories, Australia. For a detailed account of other bonded repair schemes developed at the Aeronautical Research Laboratories the reader is referred to [33].

In the above we have been primarily concerned with a particular method of analysis which allows for shear deformation in the adhesive, repair and sheet. However other methods have also been developed [34–40] which do not allow for shear deformation in the repair or the sheet. As a result, as stated in [34, 38] these methods underestimate the stress intensity factors of cracks in the repaired structure and overestimate the adhesive shear stress (which is basically an interlaminar stress). Nevertheless they are very useful contributions to repair technology.

The works of Hart-Smith [37] and Swift [36] are particularly interesting since, unlike earlier work, they allow the adhesive to behave plastically. Let us now turn our attention to the design of bonded repair schemes. Detailed design charts have been presented by Davis [41] for the repair of cracks using bonded stringers. Design charts have also been given by Jones *et al.* [28] for the repair of cracked aluminium sheets with a unidirectional boron-epoxy laminate for a crack 38·1 mm long and where the repair completely covers the crack. However the scope of this work has recently been extended by Rose [42] who showed that for the range of thicknesses considered in [28] the value of the stress intensity factor, after repair, remains constant for all crack lengths greater than a critical length which is approximately 5 mm. This means that the design charts given in [28] for a 38·1 mm crack may now be used for all crack lengths greater than 5 mm.

Another design study into the bonded repair of cracks is given by Ratwani in [43]. This study and that of [28] are complementary in as much as whereas in [28] the authors considered the stress intensity factor, the maximum adhesive stress and the peak fibre stresses as the key design quantities reference [43] concentrates on the observed crack growth rate after repair which is an area omitted in [28]. Another important feature of Ratwani's work is that in [34] he gives a simple formulae for the neutral axis offset effect which occurs when repairing a structure on one side only.

So far we have only been considering the use of fibre composite material to repair metallic structures. However the repair of damage to composite structures is an important consideration [44] and is now receiving considerable attention. A detailed discussion of the actual repair procedures for composites is given in [45], which is specifically concerned with sandwich structures. Several theoretical investigations into the repair of composites have also been undertaken [46, 47, 48, 49]. References [46, 47, 49] are primarily concerned with the repair of holes in composites while [48], using an extension of the bonded element developed in [27], considers the repair of cracks and holes. Whereas [46] only considered a bonded metallic repair, [47, 48] consider both composite and metallic repairs and conclude that for holes in composites a metallic repair has certain advantages over a composite repair scheme. This is not true for the repair of cracks in composites as shown in [48]. Reference [49] is valuable both for the comparison between theory and experiment which it gives and also for the list of references which it contains.

Limited research [50, 51, 52, 33] has been undertaken into the repair of surface cracks or cracks in thick metal sections. The only known such repair, actually in use, is the repair to cracks in the main landing wheel of the AerMacchi jet trainer in service with the Royal Australian Air Force. This repair was developed at the Aeronautical Research Laboratories, Australia and is fully documented in [33]. The development of a proposed similar repair for use on main landing wheels of Mirage III aircraft, in service with the Royal Australian Air Force, is described in [50]. In [50] it is shown that the repair of surface cracks in thick sections by means of a bonded overlay of composite material is really only applicable if the cracking is primarily caused by inclusions and/or by high residual stresses rather then by the externally applied load. Reference [51] is concerned only with the repair of surface scratches to thin sheets and makes use of the reduction in the stress intensity factor obtained for a through crack using the two-dimensional analysis methods to predict the growth rates of a repaired surface scratch. These predictions are in reasonable agreement with experimentally determined crack growth rates. The remaining work, [52], is concerned with the variation, along the crack front, of the stress intensity factor for a sandwich structure with a through crack in one face only.

4. A UNIFIED APPROACH

In the previous two Sections we saw how a large amount of work has been undertaken into the analysis of fibre composites and fibre composite repair schemes. In the begining of Section 2 we paid particular attention to the simple "shear" elements developed by Levy and coworkers [2, 3] for the analysis of the interlaminar stresses developed in composite laminates. We subsequently saw in Section 3 that an adhesive element very similar to the "shear" element of Levy has been developed [27, 28, 29] to account for the transverse shear stresses developed in the adhesive layer for bonded repairs to metallic or composite structures. Yet despite the similarity of these two approaches it appears, at first glance, that the expressions for the shear stresses used in [2, 3] and [27, 28, 29] differ due to the difference in the denominators in equations (1), (2) and (3), (4). However it will now be shown that equations (3), (4) may also be used to model the shear stress distribution in a composite laminate.

To do this we first interpret t_a as the "interlaminar resin layer" which separates two plies, or bundles of plies of composite material, the thickness of these plies, or bundles of plies, being t_s and t_0 respectively. With these definitions the interlaminar shear stresses in the "interlaminar resin layer" may, using the approach outlined in [27, 28, 29], now be obtained from equations (3) and (4).

Let us now compare this model with that of Levy *et al.*, as given by equations (1) and (2) for the case of the $(\pm 45)_s$ laminate subjected to uniaxial extension which was initially considered in [2]. In this case we will compare the interlaminar shear stresses in the resin separating the $+45°$ and the $-45°$ plies. If we consider, as in [2], that these plies have the same thickness (i.e., $t_s = t_0$) and, in accordance with [1, 2], the same transverse shear moduli (i.e., $G_0 = G_s = G$) then the separation of the mid-surface of the plies is equal to $t_a + t_0$ and equations (1) and (2) reduce to

$$\tau_{zx} = G\,(u_0 - u_s)/(t_a + t_0) \tag{5}$$

$$\tau_{zy} = G\,(v_0 - v_s)/(t_a + t_0) \tag{6}$$

Here in order to enable a simple comparison between the two approaches we have adopted the notation $u^l = u_0$, $u^{l-1} = u_s$ etc.

In general it is difficult to exactly determine t_a. However, it has recently been suggested [52], and confirmed by examination under the microscope that t_a, the thickness of the interlaminar resin layer, is approximately one tenth the thickness of an individual ply. Thus in our case $t_a = 0{\cdot}1\ t_0$ and so equations (5), (6) reduce to

$$\tau_{zx} = G\,(u_0 - u_s)/1{\cdot}1\ t_0 \tag{7}$$

$$\tau_{zy} = G\,(v_0 - v_s)/1{\cdot}1\ t_0 \tag{8}$$

Alternatively, if we use equations (3) and (4) to determine the shear stresses τ_{zx} and τ_{zy} we obtain

$$\tau_{zx} = G\,(u_0 - u_s)/t_0(0{\cdot}75 + t_a\,G/G_a\,t_0) \tag{9}$$

$$\tau_{zy} = G\,(v_0 - v_s)/t_0(0{\cdot}75 + t_a\,G/G_a\,t_0) \tag{10}$$

We thus see that one approach yields $1{\cdot}1\ t_0$ as the denominator while the other yields $t_0(0{\cdot}75 + t_a\,G/G_a\,t_0)$.

The value of G_a given in [53] for a typical epoxy resin is $1{\cdot}28$ GPa while in the problem under consideration the value of G used in [2, 3] is $4{\cdot}98$ GPa. These values give

$$0{\cdot}75 + t_a\,G/t_0\,G_a = 1{\cdot}13$$

As a result we see that in practice there is very little difference between the two approaches. Indeed since the value of t_a is to some extent uncertain it would be best to set the ratio $t_a\,G/t_0\,G_a$ to be $0{\cdot}35$, where t_0 is the thickness of an individual ply, so that the two models would coincide exactly.

5. CONCLUSION

In this paper we have outlined the various finite element methods which are available for the analysis of composite laminates and fibre composite repair schemes. In addition we have attempted to indicate when a detailed analysis is required and when simple methods will suffice. We have also presented a summary of the developments in use of bonded repair schemes and have provided an insight into the manner in which the present numerical methods, used to design bonded repairs, are linked to the numerical methods used to analyse composite laminates.

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