# Towards Damage Tolerant Design of Laminated Components

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Laminated composites are becoming increasingly attractive as primary structural components of aerospace vehicles. The possibility of small undetectable manufacturing defects and the development of damages during service has become an important concern in designing such structures. Analogous to the fracture mechanics approach to design of metallic structures, damage tolerant design of laminated composites is discussed in this paper. Some of the available analytical tools are briefly reviewed and a global-local approach has been indicated as a possible procedure for damage tolerant design of laminated components.

## 1. INTRODUCTION

In the design of metallic structures, it is well understood that it is not the yield strength or the ultimate tensile strength that would determine the life of the structure, but rather, formation of cracks and their growth that influences all subsequent events. With the growing tendency to utilise layered composite materials for critical applications, a new need to understand the behaviour of cracks in controlling the strength of laminates has arisen. Unfortunately the crack pattern in a laminate is so complex, it defies any possibility of mathematical modeling analogous to the metallic structures. This is because, several forms of cracks representing fibre breaks, matrix cracks, fibre disbonds, voids, delaminations etc., simultaneously co-exist. This kind of situation may arise during manufacture or during service. It is usually called a 'defect' if it occurs during manufacture and 'damage' if it occurs during service. Familiarly the word 'damage' is used here to represent this kind of situation and each form of crack constitutes an important component of the damage. Failure mechanisms [1] and the influence of damages [2] on the failure process in laminates play a paramount role in evolving suitable design criteria. The impracticability of considering the damage in detail in theoretical analysis, opens a search for simplified models. This paper is an attempt in this direction.

In the design of laminated components, the designer is confronted with a question as to what will be the effect of damage on the performance of the components. Often, there is a threshold level of damage before it can be detected and this is usually identified from the inspection procedures adopted.

Therefore, a methodology to evaluate the effect of a certain specified size of damage, on the behaviour of laminates is necessary. Presently, there has been considerable emphasis on coupon studies. Such studies are important in generating the necessary understanding of the inherent failure mechanisms in laminates. Their extension to components is not always easy, because of complex geometric shapes and the loading patterns. The objective of this approach paper is to indicate a simple possible methodology for damage tolerant design of composite components to form a basis for further discussion at the workshop.

The influence of various components of damage or pattern of cracks, on the laminate properties differs. Fibre disbonds, fibre breaks, matrix cracks etc., are relatively small in size. Highly detailed modeling is perhaps, the only way to obtain precise effect of each of this kind of damage. The desire to examine feasibility of simplified models, to account for gross behaviour atleast, is highly tempting. One possibility is to degrade ply properties (stiffness elements) in a manner, it reflects various components of damage. While this approach has been considered in different contexts [3], the precise methodology for implementing this concept for the purpose of damage tolerance analysis of laminated components does not seem to be available yet. However we consider this as a viable possibility hoping that systematic experimental/theoretical future studies will establish the precise procedure of degrading plies. Thus, for modeling purposes, it is suggested that this form of damage may be accounted by appropriately degrading the ply properties.

All components of damage may not be accounted for in a similar manner. It is essential in some cases to consider the geometric changes brought out by the crack. Delamination deserves such a consideration. Single delamination may not always occur in practice; multiple delaminations of different sizes are also not uncommon. Nevertheless, the single delamination case may represent the initial formative situation and hence can be considered as representative beginning of the failure process. Thus a laminate with degraded ply properties and delamination located at a specified location, may be considered as a model for obtaining the damage tolerance capability of a laminate, i.e. to evaluate whether or not a specified delamination grows under the design loading conditions and affects the performance of the structure.

The location and the size of the delamination is of prime importance, in this approach. Interlaminar zone is usually the weakest link in the composite and the failure in this zone may be expected to be initiated by interlaminar stresses. Table 1 shows a set of typical strengths of laminates. A perusal of this table clearly brings out that the interlaminar zone is the weakest link in the system, as the interlaminar shear strength (ILSS) and the inter-laminar normal strength (ILNS) are low compared to the ply strength. Using interlaminar stresses in an appropriate interlaminar failure criteria such as the quadratic stress criterion [4], it is proposed to identify probable delamination sites. Thus, theoretical models with ability for interlaminar stress estimation will serve the purpose of identifying potential damage sites. Several methods are proposed in the literature for estimation of interlaminar stresses [5]. In general,

**Table 1. Typical Strengths of Laminates**

(all values in MPA)

| Material | $V_f$ | $X$ | $X'$ | $Y$ | $Y'$ | $S$ | ILSS |
|---|---|---|---|---|---|---|---|
| GFRP | 0.6 | 780 | 500 | 28 | 118 | 75 | 40 |
| CFRP | 0.6 | 1620 | 1300 | 34 | 206 | 90 | 80 |
| KFRP | 0.6 | 1400 | 235 | 30 | 90 | 53 | 60 |
| RESIN | | 40–100 | 40–100 | 40–100 | 40–100 | 25–60 | |

$V_f$ = Volume Fraction; $X$ = Longitudinal tensile strength; $X'$ = Longitudinal compressive strength; $Y$ = Transverse tensile strength; $Y'$ = Transverse compressive strength; $S$ = In-plane shear strength; ILSS = Interlaminar shear strength.

most of these theories perform well except in critical regions such as free edges and rivet holes.

Free edges and rivet holes are known to act as interlaminar stress raisers and initiate delaminations. In general, it is not possible to account for this phenomenon adequately even through higher order plate theories. Although the Global-Local model of Pagano and Soni [6, 7], generalized laminate plate theory of Reddy [8, 9] and ply dependant laminate models of Vijayakumar et al [10] are highly promising, currently they are not available in finite element form to enable application to a general situation. Therefore the use of three-dimensional theory of elasticity in finite element form, becomes unavoidable. On the other hand, use of three-dimensional finite element scheme is not always practical, because of high level of refinement required. Even in the case of simple problems such as the laminated coupon or perforated composite plate, three dimensional analysis requires a very high number of degrees of freedom and hence for practical situations such as the wing of an aircraft, complete three dimensional analysis is prohibitive.

Thus, with the use of either the 'Laminated Plate Theory' alone or 3-D analysis only, it does not appear to be feasible to study the damage tolerance capability. Use of both is desirable. Laminated plate theory could be used to identify potential damage sites and local 3-D analysis to bring out damage tolerance capability. Several basic studies to facilitate implementation of this approach are already available in literature. Here under, we briefly describe some of the studies in progress at the Indian Institute of Science.

### *Choice of Probable Damage Sites*

Currently finite elements based on single layer laminated plate theory are available in several computer software packages and it is also possible to incorporate the capability for interlaminar stress estimation in some of them. Accurate estimation of interlaminar stresses is necessary to identify potential damage sites. Generally it is believed that higher order theories are necessary to predict interlaminar stresses. Fortunately, reasonably good estimates to interlaminar stresses can also be obtained from the first order theory. As Reddy says [5], "among these higher order theories, the first order theory seems to provide the best compromise between accuracy and computational efficiency". In general, inter-laminar stresses may be obtained either by using constitutive relations, or by integrating local equilibrium equations. The former are called "consistent estimates" while the latter may be called "statically equivalent estimates". A recent comparative study of some higher order models with regards to the estimation of interlaminar stresses in a simply supported rectangular laminated plate subjected to sinusoidal static transverse loading has indicated that the statically equivalent estimates of inter-laminar stresses in classical laminated plate theory are better than the consistent estimates in some higher order theories [11].

Recognising that the procedure for getting statically equivalent estimates gets very cumbersome in higher order theories, it may be considered worthwhile to use the classical first order theory for obtaining interlaminar stresses. Recent studies with the aid of two bench problems bring out the performance of the first order theory in estimating interlaminar stresses. The first one is the classical free-edge problem in a tensile coupon. The first theory [12], based on the displacement field

$$\hat{U} = \varepsilon_0 x$$

$$\hat{V}(y, z) = V(y) - \frac{h\xi^2}{2} W_{,y}$$

$$\hat{W}(y, z) = \xi W(y)$$

where

$$\xi = \frac{z}{h} \tag{1}$$

nterlaminar

which represents the Quasi-3D state in the coupon, has been used to obtain the inter-laminar stress distribution in laminates. Figures 1–3 shows typical results.

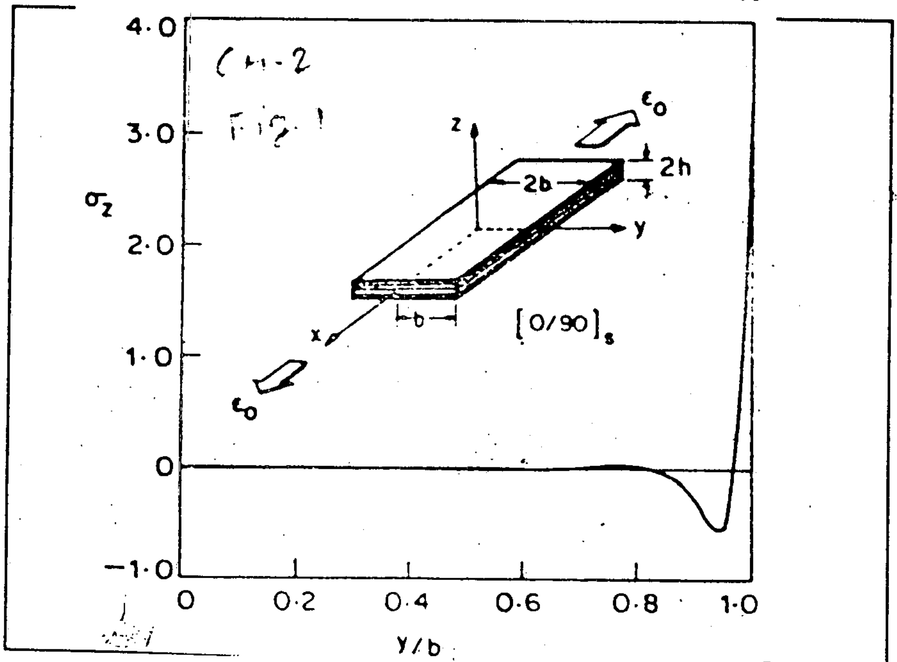


Fig. 1. $\sigma_z$ along $z = 0$ for $b = 20$.

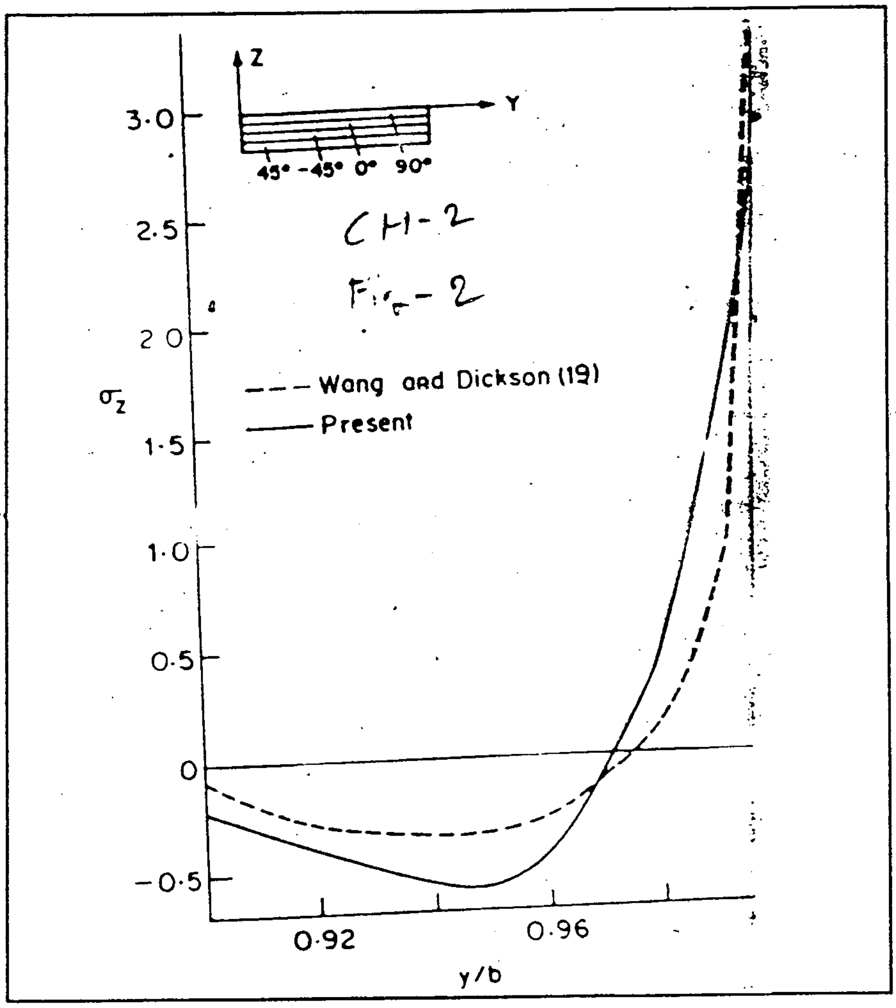


**Fig. 2.** $\sigma_z$ **along** $z = 0$ **for** $b = 20$.

Clearly, this example makes a critical test for this simple theory. It is interesting to note that it has been possible to predict the steep raise of the interlaminar normal stress by this first order theory which is analogous to the classical laminated plate theory of bending. The second example is a coupon containing delaminations [13]. The first order theory applicable for this case is based on the displacement field.

$$\hat{U} = \varepsilon_0 x + U(y)$$

$$\hat{V}(y, z) = V(y) - h\xi \overline{W}_{,y} - \frac{h\xi^2}{2} W_{,y}$$

$$\hat{W}(y, z) = \overline{W}(y) + \xi W(y) \qquad (2)$$

(see (Fig. 4). The solution is obtained by treating the problem in three zones

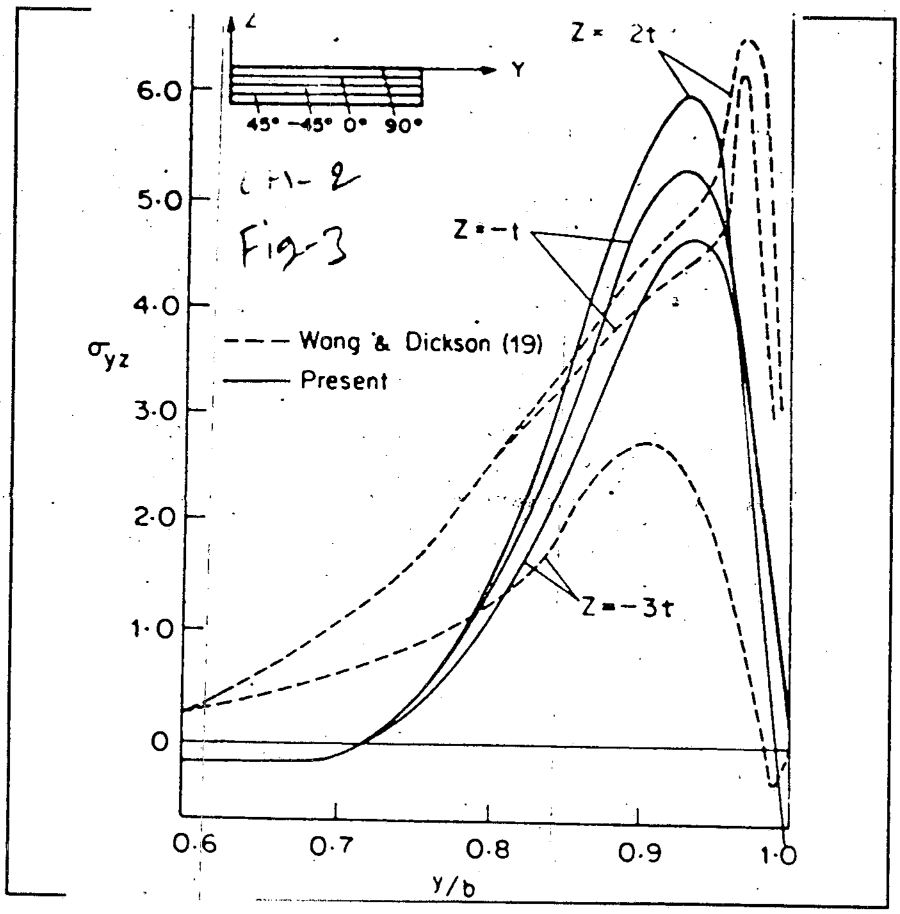


Fig. 3. $\sigma_{yz}$ in [± 45°/0°/90°].

and matching the displacement and force variables at the boundaries. These studies have also indicated that, with the first order theory it is possible to predict reasonably well, stress variations ahead of the delamination tips (Fig. 5). Figure 6 shows the variation of $\sigma_z$ at the delamination tip for various values of the size of delaminations. A comparision of the strain energy release rates is given in Table 2. Figure 7 shows the variation of the strain energy release rate with delamination size. The simplicity of the model with a reasonable level of agreement shown in the case of these two problems encourages further investigations to explore the first order theory. Currently finite elements based on single layer laminated plate theory are available in several computer software packages and it is also possible to incorporate the capability for interlaminar stress estimation in some of them. A general discussion of modeling considerations of laminates is available in [14] and the first order theory appears to represent a viable model for global analysis of composite panels for identifying potential delamination sites.

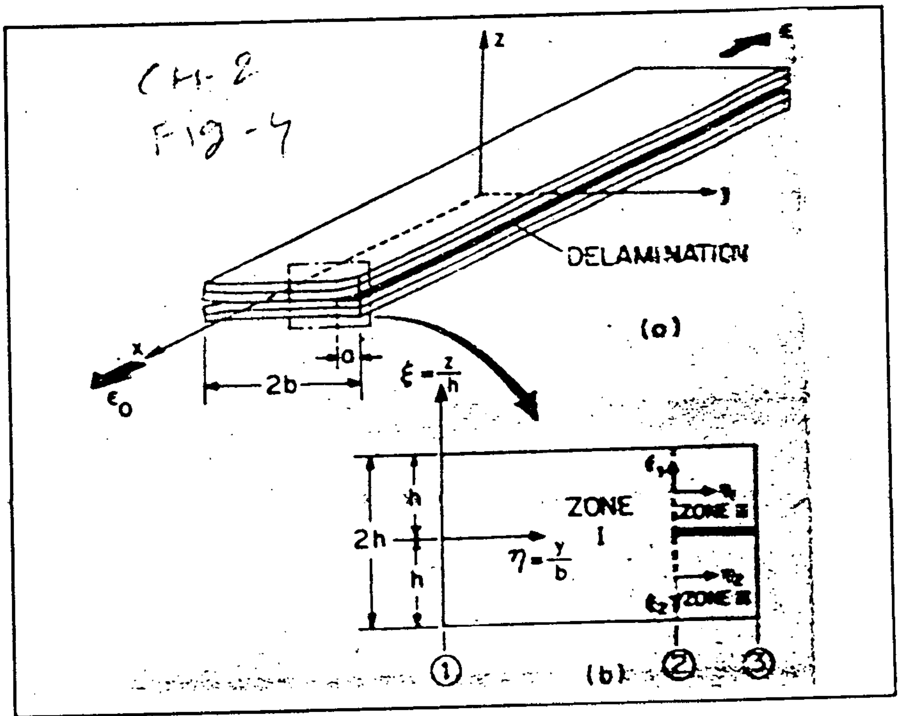


Fig. 4. A typical laminate with edge delamination.

*Local Analysis*

Currently, three dimensional finite element analysis appears to be well established to study local effects. The 20-noded isoparametric brick element is perhaps the most favoured element today for studying local stresses near free edges or delamination tips and to obtain the strain energy release rates. It appears that most of the available 3D studies aim at understanding the laminate behaviour rather than using it as a design tool. This is to be expected in view of the very high degrees of freedom required to model local effects. Further this element has no provision for interelement continuity of interlaminar stresses—a feature which might help in modelling the local effects. The case of the free edge stress analysis of a laminated coupon presents a simpler situation permitting the use of quasi three dimensional elements. This example served as a valuable bench mark problem for evaluating various strategies for solution.

Often a single stage 3D analysis is not favoured. Instead, a multi stage scheme is preferred wherein the finite element grid in a local region is successively refined and displacement boundary conditions obtained in the previous step are imposed on a boundary enclosing the local region [15]. Although the solution is attained in several stages, the procedure is attractive, particularly, when very large computers are not readily available.

In view of a very large number of degrees of freedom involved in the three dimensional analysis, the search for strategies to reduce the problem size is natural. The common approach is to use graded meshes and/or multistage procedure. With the use of transition element, it is possible to attempt a single

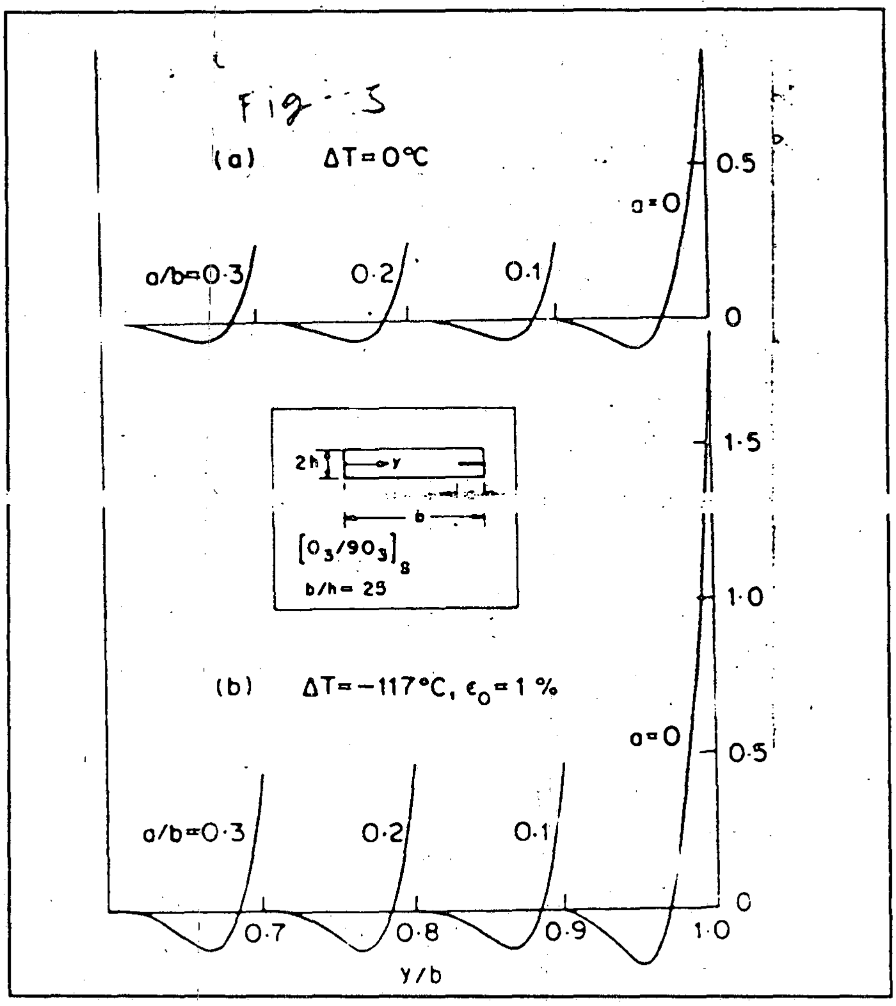


**Fig. 5.** $\sigma_z$ **variation ahead of the delamination tip.**

stage analysis. This concept has been implemented recently in the case of the free edge problem of a laminated coupon in extension, and very encouraging results have been obtained [16]. Figure 8 shows a finite element idealisation of a $[0/90]_s$ coupon under extension. In view of symmetries, a quarter of the cross-section is considered as an assembly of three regions. A large part of the problem (Region 1) is covered with simple elements and the minimum essential part (Region 3) is covered with Quasi-3D elements. Both the regions are interconnected by a transition element. Transition element is a simple super element with dissimilar nodes on the boundary to enable smooth connection with either side and it is obtained by first treating it as an assembly of Quasi-3D elements and then imposing appropriate constraints through a static condensation procedure. Figure 9 shows a comparison of $\sigma_z$ and $\sigma_{yz}$ near the free edge with complete Quasi-3D analysis. A substantial reduction in problem size with almost no loss of accuracy may be noted. Figure 10 shows the

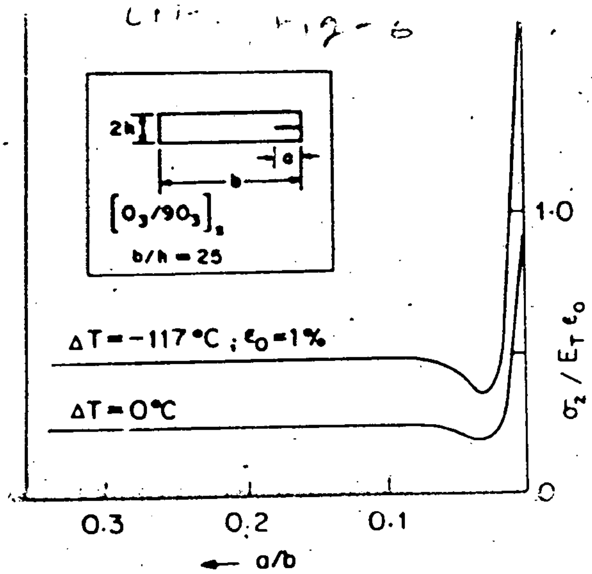


**Fig. 6. Variation of $\sigma_z$ at delamination tip.**

**Table 2. Comparison of strain energy release rates**

| Laminate | $\varepsilon_0$ (%) | ΔT (°C) | 3–D Stiffness | Modified Stiffness | Whitney* (Ref.18) | Q 3–D Fe Analysis | CLPT |
|---|---|---|---|---|---|---|---|
| $(30/-30_2/30/90_2)_s$ | — | 0 | 1.1379 | 1.2739 | 1.3 | 1.2506 | 1.2740 |
| | 0.3 | −117 | 2.0130 | 2.2432 | 2.0 | 1.9551 | 2.2435 |
| $(90_2/30/-30_2/30/)_s$ | — | 0 | 1.1384 | 1.2741 | 1.31 | 1.27 | 1.2740 |
| | 0.3 | −117 | 0.5105 | 0.5771 | 0.63 | 0.67 | 0.577 |
| $(0_3/90_3)_s$ | — | 0 | 0.0099 | 0.0155 | 0.016 | 0.0155 | 0.0155 |
| | 1.0 | −117 | 0.0455 | 0.0604 | 0.048 | 0.0605 | 0.0606 |
| $(45/-45_2/45/90_2)_s$ | — | 0 | 0.6803 | 0.7311 | 0.73 | 0.7285 | 0.7303 |
| | 0.3 | −117 | 0.8606 | 0.9232 | 0.92 | 0.9194 | 0.9222 |

*Values measured from graphs given in [18].

thickness wise variation of $\sigma_z$ and $\sigma_{yz}$. Table 3 shows a comparision of strain energy release rates with complete Quasi-3D analysis.

It may be mentioned here that the 3D finite element analysis of laminate is not without problems [17]. Often there were convergence problems and results depended on the mesh size. Therefore, it appears, that some more studies are required to establish the reliability of this approach for a general situation, before it is accepted as a standard tool for damage tolerance analysis of aircraft primary structures.

## 2. CONCLUSIONS

Assuming that the classical laminated plate theory, with some modifications if necessary, or other 2D theoretical models are adequate to represent the

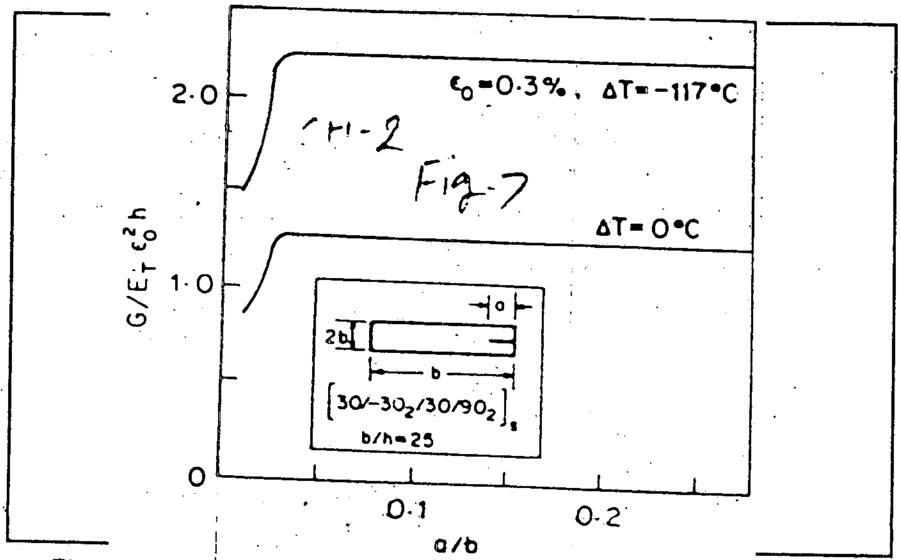


Fig. 7. Variation of strain energy release rates with delamination size.

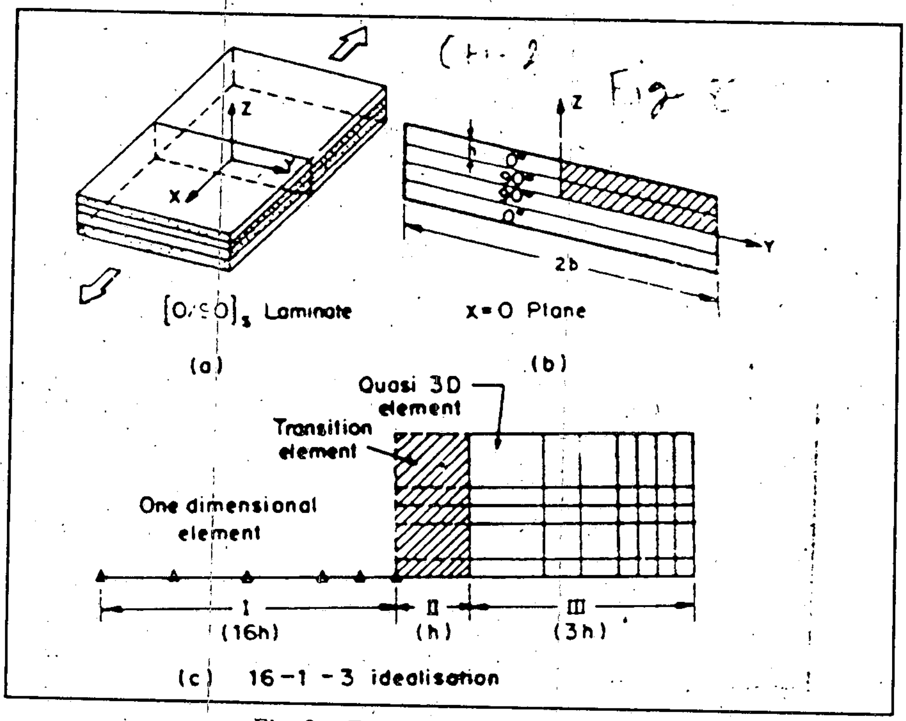


Fig. 8. Finite element idealisation.

gross behaviour of the laminate to identify potential damage sites and that it is feasible to obtain the effect of a damage at such locations by a local 3D analysis, Global-Local approach can be considered for use at the design stage of aircraft primary structures. Based on the constructional features and the known behaviour of laminates, certain critical locations such as rivet holes

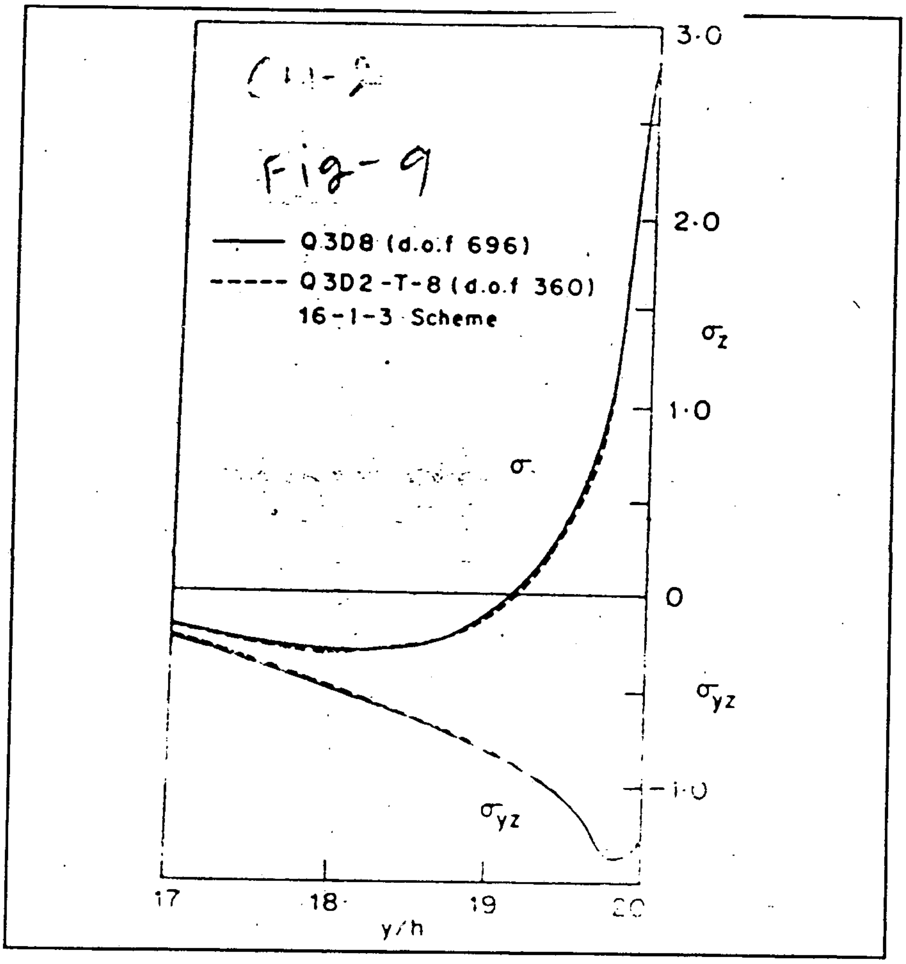


Fig. 9. $\sigma_z$ and $\sigma_{yz}$ variations along $y$ axis near free edge.

and free edges may be chosen for damage tolerance analysis. This is to be augmented by the results of the global analysis—which may bring out possibility of embedded damages, perhaps, around concentrated loads if any, and such other situations. A local analysis around these locations will provide the required damage tolerance capability. In this context, we may consider the structure to be damage tolerant if a specified size of delamination located at a critical location in the laminate with appropriately degraded plies does not grow under the limit load. Of course, the size of the delamination is to be based on inspection procedures and the NDE capability.

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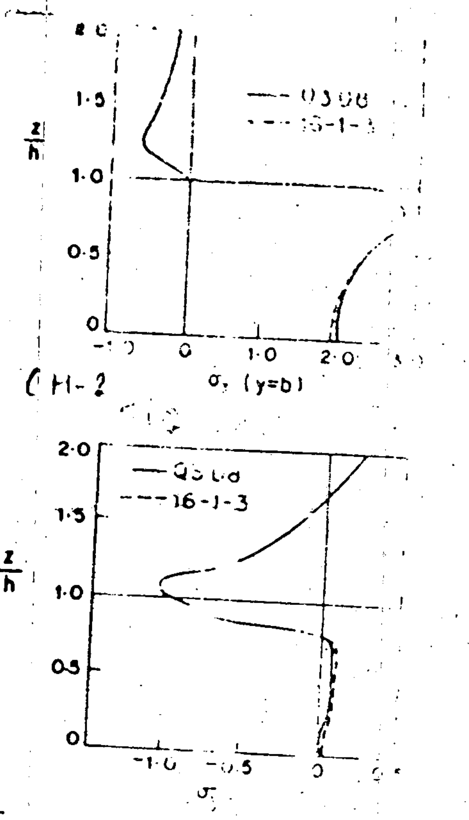


Fig. 10. $\sigma_z$ and $\sigma_{yz}$ variations along z axis.

**Table 3. Strain energy release rate in $[0/90]_s$ laminate**

| Delamination Location | | Q3D8 696 d.o.f | Q 3D 2F Q3 D8 360 d.o.f |
|---|---|---|---|
| Mid-Plane | $G_I$ | 0.129 | 0.132 |
| [0/90] interface | $G_I$ | 0.0726 | 0.0732 |
| | $G_{II}$ | 0.0571 | 0.0573 |

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