A SUMMARY AND REVIEW OF COMPOSITE LAMINATE DESIGN GUIDELINES

Task 22

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FOREWORD

This report was prepared under NASA Contract NAS1-19347, "Composite Commercial Aircraft Primary Sructures (CCAPS)," administered by the NASA Langley Research Center and addresses the analytical work performed under Task 22 of this contract.

The NASA technical monitor for the project is Mr. Juan R. Cruz. The CCAPS program manager at Northrop Grumman Corporation is Dr. Steven G. Russell. The task manager at Northrop Grumman Corporation is Dr. Robert P. Ley, who also reviewed the manuscript. The manuscript was researched and prepared by Dr. J. Kim Bailie and Dr. Arun Pasricha. The authors express their appreciation to Ms. Helen Richardson for her assistance in editing this manuscript.

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NOMENCLATURE

A_{ij}	=	Laminate extensional stiffness matrix
B_{ij}	=	Laminate extensional-bending coupling stiffness matrix
D_{ij}	=	Laminate bending stiffness matrix
M_x	=	Bending moment resultant per unit width about laminate y axis
M_y	=	Bending moment resultant per unit width about laminate x axis
M_{xy}	=	Twisting moment resultant per unit width
N_x	=	Stress resultant per unit width in laminate x direction
N_y	=	Stress resultant per unit width in laminate y direction
N_{xy}	=	Shear stress resultant per unit width
Q_{ij}	=	Engineering reduced stiffness constants
z_k	=	Distance of kth ply's inner surface, from laminate midsurface
α_{11}	=	Thermal coefficient of thermal expansion along "1" axis
α_{22}	=	Thermal coefficient of thermal expansion along "2" axis
ε_x^o	=	Midsurface direct strain along x axis
ε_y^o	=	Midsurface direct strain along x axis
γ_{xy}^o	=	Midsurface shear strain
κ_x	=	Bending curvature about y axis
κ_y	=	Bending curvature about x axis
κ_{xy}	=	Twisting curvature
θ	=	Ply angle
τ	=	Shear stress
υ	=	Poisson's ratio

<u>Subscripts</u>

x	=	Midsurface coordinate along the 0° axis
y	=	Midsurface coordinate along the 90° axis
z	=	Through-thickness coordinate
k	=	Ply number
1	=	Ply coordinate along the fiber for tape plies or along the warp for cloth plies

SECTION 1

ABSTRACT AND OBJECTIVES

The purpose of this study is to review many of the available design guidelines for unidirectional tape, laminated aerospace composite panels. Guidelines for bonded and bolted joints, cutouts, and durability and damage tolerance are also presented, as they strongly influence designs for production aircraft. These guidelines are accompanied by explanations of why each one was generated and the influence each one has on the structural performance of various aircraft components. Most of these guidelines were developed during actual construction of relatively simple aircraft components in the late 1960s and early 1970s. Unfortunately, generally available literature detailing the derivation of these guidelines is scarce; hence, it was made necessary to obtain information directly from various aerospace engineering organizations and notes presented in lectures. The scarcity of formal documentation may also be due, in part, to the fact that many lessons were learned when unpredicted failures occurred during early development programs that are only now being declassified.

The present review is focused on composite laminates made of graphite fibers embedded in a polymer matrix since use of such laminates is increasing in highly loaded aerospace primary structures. Simple analyses and data are presented to illustrate the basis for many of the guidelines.

The objective of this review is to (1) gather the design guidelines currently used for structural design and analysis of unidirectional tape laminates, (2) review their derivation, and (3) explain their ranges of application. Many of these guidelines have served the aerospace industry for close to three decades as they were developed for fighter/attack aircraft structural components being designed in the late 1960s. Attention was directed towards production aircraft that were to be certified for operating lives on the order of 6,000 flight hours.

By gathering together these guidelines and critically evaluating their derivation, it is feasible to assess situations under which they can be safely relaxed or even ignored. Such an assessment is performed for a spar cap composed entirely of unidirectional plies proposed for unmanned air vehicles under development for NASA's ERAST program.

SECTION 2

INTRODUCTION

Sophisticated analyses have played a pivotal role in the development of composite aerospace structures. However, as with all engineering materials, practical, albeit less sophisticated, design guidelines have been developed that help the designer exploit the material's strengths while mitigating the adverse effects of the material's weaknesses. Many of these guidelines evolved as a result of prototype development programs and industrial experience rather than being derived from first principles. Proprietary rights and market-place pressures have hindered the emergence of these guidelines into the public domain through formal documentation. Niu provides abundant useful design advice and guidelines along with supporting rationale [Ref. 1]. Tracing the origin of and supporting data for these guidelines is difficult since many of the constraints were imposed to accommodate the composite materials and manufacturing processes used more than 20 years ago. These processes are continually being improved so guidelines having a sound basis for application to certain components, materials, and fabrication techniques of a bygone era may not be universally applicable to components being designed today.

Laminated composites made from unidirectional tape must be designed so that fibers carry most of the applied loading. Primary loading of the resin matrix leads to strengths an order of magnitude less than those in which fibers dominate load transmission. Many of the guidelines have evolved to ensure advantage is taken of the material's inherent strengths while care is taken to mitigate the effects of the material's weaknesses. Hence, the first step towards an understanding of these guidelines is to review the strengths and weaknesses of typical laminated composite materials.

SECTION 3

STRENGTHS AND WEAKNESSES OF LAMINATED COMPOSITES

Well-designed composites exhibit the following highly desirable characteristics:

1. Excellent strength-to-weight and stiffness-to-weight ratios.
2. Very good fatigue properties.
3. Tailorability of stiffness and/or strength so that the resulting structure meets the design requirements most efficiently.
4. Corrosion resistance that is far superior to aluminum alloys and other metals.

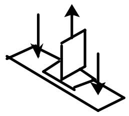
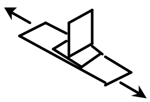
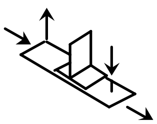
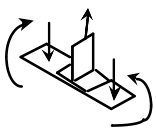
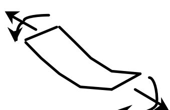
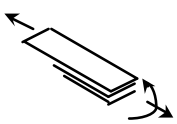
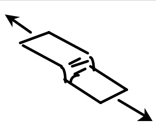
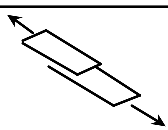
However, composites have several weaknesses. Some of these weaknesses are:

Very Low Interlaminar Tension Strength. This low strength makes composite laminates vulnerable to small out-of-plane loads and eccentricities. Numerous examples of design details that give rise to these types of loads are illustrated in Figure 3-1.

Nonlinear, Rate-Dependent Response of Most Polymer Resins. Laminates carrying significant loads in resin matrix-dominated directions may creep, particularly at elevated temperatures, and exhibit highly nonlinear stress-strain behavior. Use of resin matrix-dominated laminates can lead to excessive deflections and, after a sufficient number of loading cycles, operational or structural failures.

Microcracking of the Polymer Matrix. Resin matrix cracks may not always be immediately structurally degrading, but their propagation can lead to pressure cabin and fuel tank leaks as well as degraded durability and damage tolerance. Such cracks also allow ingression of moisture and other liquids. If cyclic freezing/thawing of these liquids occurs, crack propagation is likely to be accelerated.

Order of Magnitude Differences in Coefficient of Thermal Expansion (CTE). This CTE mismatch parallel and transverse to the fiber orientation which can lead to unacceptable warping and thermal stresses. As shown in Section 8, use of nonsymmetric lay-ups can result in laminates that warp upon cool down from the cure temperature. This can result in serious assembly problems and other functional deficiencies.

Configuration	Description	Application
 **Flatwise Tension**	Skin/Stiffener Separation Due to Normal Pressure Loading	Wing Skin/Spar Flatwise Tension Due to Internal Fuel Pressure
 **Transverse Tension**	Stiffener Web Splitting Due to Transverse Tension Loading	Wing Skin/Spar Transverse Tension Due to Chordwise Loads
 **Lateral Bending**	Skin/Stiffener Separation Due to Lateral Stiffener Bending	Lateral Spar Bending Due to Asymmetrical Fuel Pressure
 **Postbuckling**	Skin/Stiffener Separation Due to Post-buckling Deformation and Loads	Stiffened Panels Subjected to Compressive and/or Shear Buckling
 **Curved Panel Bending**	Interlaminar Stresses Due to Panel "Beam-Column" Effects	Fuselage Skins and Frames Subjected to Bending Loads
 **Thickness Transitions**	Interlaminar Stresses Due to Combined Loading and Local Bending	Ply Drop Offs, Build-Ups, and Doublers
 **Irregular Loading**	Interlaminar Stresses and Local Bending Due to Axial Loading in the Presence of Eccentricity	Joggles and Kinks
 **Bonded Joints**	Interlaminar Stresses Due to Local Bending Arising From Eccentricity	Single and Double Lap Bonded Joints

MS97-359

FIGURE 3-1. TYPICAL OUT-OF-PLANE LOADS IN COMPOSITE STRUCTURES [REF. 2]

Reduction in Strength Due to Impact-Induced Damage. Such damage is often not visible from the impacted surface. This non-visible damage is a primary cause for concern in all phases of certification of composite structures for durability and damage tolerance. The literature related to the damage tolerance of composites contains hundreds of references on the effects of impact.

Environmental Sensitivity of Polymer Resins. Significant strength reductions of resin matrix-influenced properties such as compression and shear strengths can arise due to exposure to elevated temperatures and moisture. These strengths at the “elevated temperature, wet” (ETW) design conditions are usually a limiting factor for high performance military aircraft.

Reduced Thermal and Electrical Conductivity Relative to Commonly Used Structural Metals. The reduced thermal conductivity can result in the presence of higher thermal gradients than those that occur in higher conductivity metal structures. These gradients may cause unacceptable thermal/structural responses. The lessened electrical conductivity of composites influences the response to lightning strikes in two ways: (1) direct effects primarily affecting structure, and (2) indirect effects primarily affecting electronics and electrical subsystems.

Direct effects are those where the energy density is large enough to cause local structural damage or failure. At lightning strike attachment points on the exterior of the flight vehicle, damage may range from superficial burn marks to penetration through skins. Additionally, overheating problems can arise at low conductivity (high resistance) joints. An unacceptably dangerous situation can arise between fasteners. Charge built up within the fasteners is not carried away by the low conductivity composite and instead dumps or arcs from sharp points to adjacent fasteners. This arcing can ignite vapors in a fuel tank. This is illustrated in Figure 3-2.

Indirect effects of lightning strike on composite structures are those related to aircraft performance rather than structural damage. Composite structures provide less electromagnetic shielding of interior electronic components than metal equivalents. These components can be seriously damaged by static discharge.

One technique used to mitigate the effects of lightning strike is to install metallic mesh outer plies to increase conductivity and shielding albeit with some weight penalty. Niu provides a review of the influence on composite structures of lightning strikes [Ref. 1].

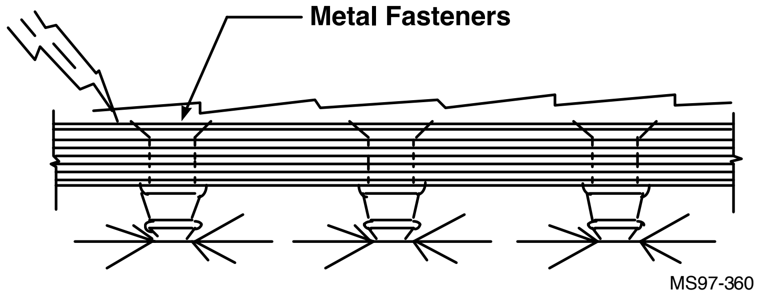


FIGURE 3-2. ARCING BETWEEN UNPROTECTED METAL FASTENERS IN A COMPOSITE FUEL TANK [REF. 1]

Galvanic Incompatibility Between Graphite Fibers and Some Metals Such As Aluminum Alloys. This incompatibility results in corrosion damage when the materials are in contact. To prevent corrosion damage, guidelines such as the required use of titanium fasteners (versus steel or aluminum) have been established. Layers of glass instead of graphite composite need to be interposed between graphite plies and aluminum surfaces. This is most efficiently accomplished by cocuring a glass composite ply as the outer ply of the laminate.

The eight weaknesses associated with composite laminates can easily outweigh the four strengths in an improperly designed composite structure. One way to prevent this from happening is to follow a set of fundamental design guidelines. These fundamental guidelines are the topics of the following section.

SECTION 4

FUNDAMENTAL LAMINATE DESIGN GUIDELINES

To take advantage of the strengths of composite structures and preclude their weaknesses from preventing the attainment of structural goals, a number of guidelines have been developed. In the remainder of this text, unnotched means free of all holes, cutouts, impacts, and other strength-degrading damage that is measurable on a macroscopic scale. While the general guidelines presented in this section focus on flat laminated panels to ease analyses and discussions, they also apply to curved laminates unless specifically stated otherwise. Stiffeners and other small, complex parts require additional, more specific guidelines and, hence, are not addressed in this section. The general guidelines are most applicable to laminates made of unidirectional graphite fibers embedded in polymer resin matrices. Guideline modifications required for cloth (fabric) laminated structures and laminates built using other fibers and resin matrices are briefly discussed in Section 17.

Guideline 1 Laminates Are to Be Symmetric About Their Middle Surfaces.

There are two reasons why this guideline is representative of good practice: (1) to uncouple bending and membrane response, and (2) to prevent warping under thermal loading. Clearly this guideline cannot always be rigorously enforced such as in zones where thickness is tapered. However, any asymmetry existing due to manufacturing constraints should be minimized.

To understand coupling of membrane and bending response, laminate equations are required. These equations are derived and discussed by Jones [Ref. 3] and others; hence, they are only briefly outlined here. Notation is used that is most commonly found in analyses and computer codes employed by development engineers in the aerospace industry. Notation for a ply is shown in Figure 4-1. The lamina 1 and 2 material axes are, respectively, along the fiber direction and at right angles to it within the plane of the ply. The laminate 0° axis lies along the structural or load-oriented axis with the ply angles θ, shown in Figure 4-1, measured from it.

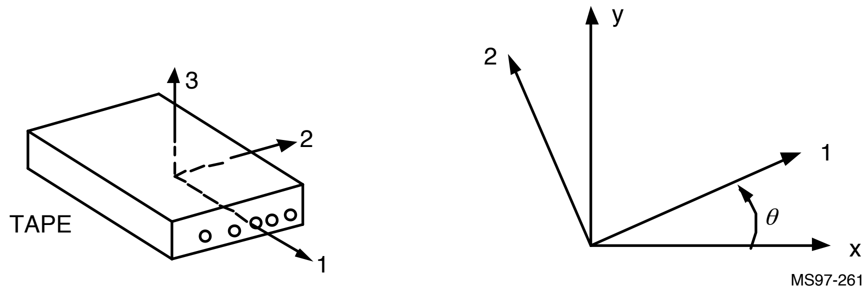
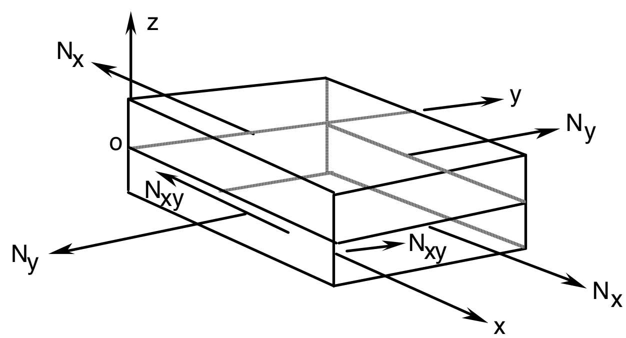


FIGURE 4-1. PLY (LAMINA) AXES

The laminate x and y axes also lie within the plane of the laminate. These two axes form a mutually orthogonal set with the z axis in the thickness direction. The origin is at the middle surface of the laminate. The laminate axes are illustrated in Figure 4-2, along with the three membrane stress resultants N_x, N_y, and N_{xy}.



x-y Plane is Laminate Midsurface

FIGURE 4-2. LAMINATE AXES AND STRESS RESULTANTS

A laminate cross section is illustrated in Figure 4-3 to show ply (lamina) locations and numbering. Note that the terms "ply" and "lamina" are used interchangeably throughout this manuscript.

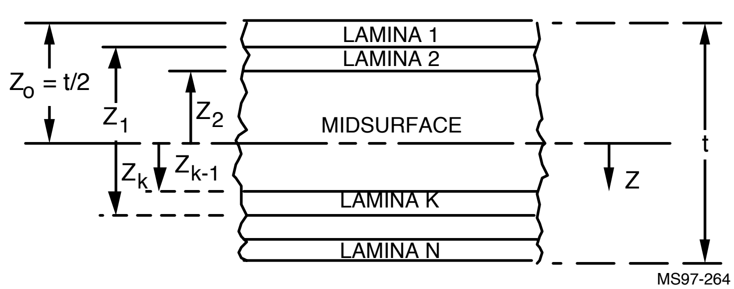


FIGURE 4-3. NOTATION FOR LAMINATE THICKNESSES AND PLY LOCATION

The equations for the stress and moment resultants in terms of the middle surface strains and curvatures are:

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^{\circ} \\ \varepsilon_y^{\circ} \\ \gamma_{xy}^{\circ} \end{Bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{16} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix} \tag{4.1a}$$

$$\begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{16} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^{\circ} \\ \varepsilon_y^{\circ} \\ \gamma_{xy}^{\circ} \end{Bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{16} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{Bmatrix} K_x \\ K_y \\ K_{xy} \end{Bmatrix} \tag{4.1b}$$

where

$$A_{ij} = \sum_{k-1}^{N} \left(\overline{Q}_{ij}\right)_k \left(z_k - z_{k-1}\right) \quad B_{ij} = \sum_{k-1}^{N} \left(\overline{Q}_{ij}\right)_k \left\{z_k^2 - \left(z_{k-1}\right)^2\right\}$$

$$D_{ij} = (1/3) \sum_{k-1}^{N} \left(\overline{Q}_{ij}\right)_k \left\{z_k^3 - \left(z_{k-1}\right)^3\right\} \tag{4.1c}$$

where

$i, j = 1,2,6$

The $\overline{Q}_{ij}$ are the ply stiffnesses in the laminate (x-y) coordinate system reduced to reflect the assumption of lamina plane stress [Ref. 3, Equation 2.61]. The A_{ij} are the membrane or extensional stiffnesses. The D_{ij} are the flexural or bending stiffnesses. The B_{ij} are responsible for the coupling between bending and membrane response. When the B_{ij} are non-zero, a membrane loading induces bending curvatures and bending moments induce membrane strains. It is to be noted that the B_{ij} involve squares of the z coordinate; hence, they vanish when the laminate has geometric and material symmetry about its middle surface.

Uncoupling the membrane and bending response is a significant advantage for two reasons. First, it grossly simplifies the measurement of laminate membrane and bending strengths and stiffnesses by test. Inducing curvatures by application of membrane loading and membrane strains when moments are applied complicates the gathering of empirical data to a considerable degree. An example of the complications caused by unsymmetric laminates is the measurement of laminate open-hole tension (OHT) strength. While the OHT of most symmetric, fiber-dominated laminates can be characterized using a relatively small number of coupon tests, the complexity added by allowing unsymmetric laminates would increase both the number of tests required as well as the cost of a single test due to the more complex, non-intuitive response exhibited by each specimen as illustrated in Figure 4-4.

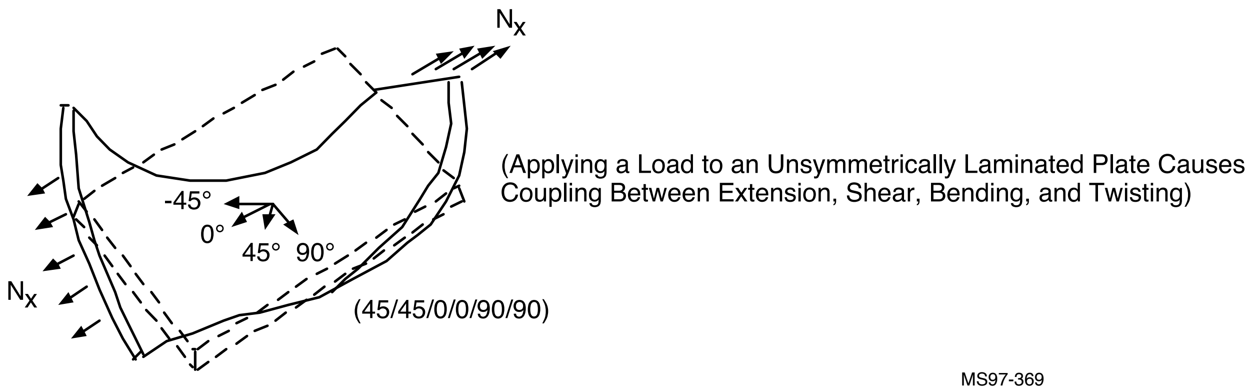


FIGURE 4-4. AN EXAMPLE OF COUPLING INTRODUCED BY UNSYMMETRIC LAMINATES [REF. 1]

Uncoupling laminate bending and membrane response also eliminates warping due to changes in temperature owing to the difference in CTE between the 1 and 2 directions of a composite lamina described in Section 4 (see Figure 4-1). Nonsymmetric laminates warp in response to an applied uniform temperature change across their thickness. This warping can occur during cool-down from the cure temperature and during in-service operations.

Hence, the primary design guideline suggested for composite laminates is to make them symmetric unless there is a very sound reason doing otherwise. Symmetry simplifies analysis, testing, definition of allowables, and manufacturing. If a locally nonsymmetric laminate is essential, the asymmetries should be kept as close to the laminate middle surface as feasible to minimize the warping. Warping during cure cycle cool-down of a part that is to be assembled to others presents a potential assembly tolerance problem. The stiffer the laminate and the greater the asymmetries, the more preload or shimming is required to provide adequate fit-up. This can result in significant stresses being induced in the part during assembly that may degrade the part's structural integrity. These assembly-related issues tend towards practical insignificance for very thin laminates. Thin laminates may be so flexible they can be easily assembled without building significant assembly stresses into them.

Guideline 2 Laminates Are Required to Be Balanced.

In this context, balanced means that angle plies (those at any angle θ other than 0° or 90°) should occur only in balanced pairs. For the 0/±45/90 laminate family, any +45° ply is to be accompanied by a -45° ply. A typical example of the difference between balanced and unbalanced symmetric laminates is shown in Figure 4-5.

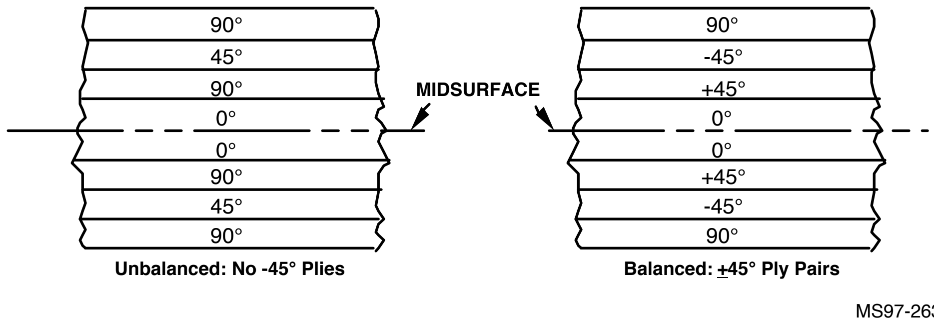


FIGURE 4-5. ILLUSTRATION OF BALANCED AND UNBALANCED SYMMETRIC LAMINATES

The significance of balanced laminates is two-fold. First, the membrane coupling between in-plane normal and shear behavior is removed since the stiffness coefficients A_{16} and A_{26} are both zero. This can be explained by looking at the equations for membrane loading of a symmetric laminate. They are extracted from Equation 4.1 as:

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^{\,o} \\ \varepsilon_y^{\,o} \\ \gamma_{xy}^{\,o} \end{Bmatrix} \tag{4.2}$$

Coupling between in-plane normal and shear response is due to non-zero A_{16} and A_{26} terms. Using Equation 4.1 and Jones' Equation 2.80, the normal-shear coupling terms can be written in full as:

$$A_{16} = \sum_{k=1}^{N} \left(\bar{Q}_{16}\right)_k t_k \tag{4.3a}$$

where

$$\left(\overline{Q}_{16}\right)_k = \left(Q_{11} - Q_{12} - 2Q_{66}\right)_k Sin\theta Cos^3\theta \quad + \left(Q_{11} - Q_{22} - 2Q_{66}\right)_k Sin^3\theta Cos\theta \qquad (4.3b)$$

and

$$A_{26} = \sum_{k=1}^{N} \left(\overline{Q}_{26}\right)_k t_k \qquad (4.4a)$$

where

$$\left(\overline{Q}_{26}\right)_k = \left(Q_{11} - Q_{12} - 2Q_{66}\right)_k Sin^3\theta Cos\theta \quad + \left(Q_{11} - Q_{22} - 2Q_{66}\right)_k Sin\theta Cos^3\theta \qquad (4.4b)$$

The Q_{ij} are the ply stiffness terms defined by Jones [Ref. 3, Eq. 2.65] and the t_k are the ply thicknesses. It is noted that A_{16} and A_{26} both contain only products of odd powers of both Sinθ and Cosθ. Hence, 0° and 90° plies do not contribute to A_{16} and A_{26}. Furthermore, A_{16} and A_{26} can be reduced to zero in any laminate, if every ply at an angle θ, has a matching one at -θ.

Equations 4.2, 4.3, and 4.4 can also be used to show that for a single angle θ, the degree of anisotropy, quantified by the magnitude of A_{16}, A_{26}, D_{16}, and D_{26}, are inverse functions of the number of plies [Ref. 3]. This fact is demonstrated for angle-ply laminates by Jones [Ref. 3, Section 4.4.4]. Such laminates are defined as having plies stacked alternately at θ and -θ.

Satisfying this “balancing” guideline results in the following simplified stress resultant-strain equation, with in-plane normal and shear responses uncoupled.

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & 0 \\ A_{12} & A_{22} & 0 \\ 0 & 0 & A_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^{\,o} \\ \varepsilon_y^{\,o} \\ \gamma_{xy}^{\,o} \end{Bmatrix} \qquad (4.5)$$

Use of unbalanced laminates complicates the interpretation of coupon test data required for the determination of certain parameters such as open-hole strength. Furthermore, it adds to the expense of such testing. Unbalanced laminates also exhibit behavior that is not intuitive to most engineers in industry accustomed to working with metals. Oversight of this non-intuitive behavior can result in unwanted deformations and stresses being induced in built-up structure. Consider thermal response.

A ply will have only two primary CTEs, α_{11} and α_{22} with respect to its material (1-2) axes. When working with a ply oriented at angle θ with respect to the laminate x axes ($0 < \theta < 90$), a thermal shear coefficient of expansion (α_{xy}) arises in the laminate x-y coordinate system. This can be shown by using the strain transformation [Ref. 3, Eq. 2.68]. Hence, an unrestrained, initially rectangular ply skews into a parallelogram at temperatures other than the stress-free temperature. This skewing of an angle ply can result in skewing of the laminate it was built into. This skewing is undesirable in most applications and can be negated if these plies occur only in balanced pairs.

The laminate bending response may also be simplified if the laminate is balanced. The terms D_{16} and D_{26} of Equation 4.1 couple bending and twisting. The D_{16} and D_{26} of Equation 4.1 are always non-zero in symmetric laminates containing unidirectional plies at angles other than 0° or 90°. Hence, application of a bending moment produces twisting and a twisting moment causes bending curvatures. However, if the angle plies are dispersed within the laminate in ± pairs, the D_{16} and D_{26} terms are relatively small compared to other terms in the D matrix for all but the thinnest laminates.

While adherence to Guideline 4.2, requiring the use of only balanced laminates, is prudent in the vast majority of situations, there is at least one important exception. The exception occurs in the case of aeroelastic design of wings. In many situations, there is a clear advantage to using unbalanced lay-ups to produce extension/compression-shear coupling in the skins of these wings [Refs. 5,6]. One specific case where this coupling has proved beneficial is that of forward swept wings (FSW). It has been shown that for forward sweep angles above ~20°, the FSW is not practical when the wing skins are made of metal [Refs. 7,8]. The problem with the FSW design is that bending and twisting responses to airloads are coupled resulting in an angle of attack that increases towards the tip. This increasing angle of attack under load can lead to divergence and structural failure of the wing at high dynamic pressure. Using composites, it is possible to lay-up the skins so that a torsion-bending coupling arises that counteracts the increasing angle of attack. Lynch, et al., explain how the use of unbalanced lay-ups produced the desired warping or twist of the wing when subjected to bending loads [Refs. 6-8]. This design philosophy culminated in the success of the Northrop Grumman X-29 Technology Demonstrator aircraft. Its structural design is reviewed by Dastin, et al. [Refs. 9,10].

Guideline 3 Do Not Extrapolate Test Data.

In spite of past time and effort spent on the development of aerospace laminated composite structures, there are still many applications where novel concepts cannot be designed and analyzed with sufficient confidence to commit them to flight vehicles. These concepts must always be validated by design development testing. A few cases require special attention.

1. Typical design guidelines specify fibers be placed in at most four different orientations, namely 0°, ±45°, and 90°. There are likely to be situations where the plies are not all at 45° to each other and/or are positioned in more than four angles.
2. Failure strains are developed almost exclusively for uniaxial loading. Validated failure criteria, applicable to notched laminates subject to combined loading, are scarce.
3. Composite materials databases are fiber and matrix specific. Extrapolating them to new materials is, at the least, very risky.
4. Environmentally-induced material property changes can be very important. The type of changes and their magnitude depends on the severity and duration of the part's service life. For airframe structures, exposure to elevated temperatures and moisture results in serious reductions in strength. For satellites, outgassing, and microcracking due to thermal cycling in vacuum must be addressed.
5. Laminated composites are notoriously sensitive to out-of-plane such as those that arise due to eccentric load paths, such as those shown in Figure 4-1. Load path eccentricities should always be minimized during the design process. The failure analysis of composite structures subject to out-of-plane loads is complex and configuration-specific. Furthermore, the scatter in measurements of out-of-plane properties is large. Adequate confidence in the performance of composite structures subject to significant out-of-plane loads is unlikely without extensive testing.

SECTION 5

DESIGN/ANALYSIS AIDS FOR UNNOTCHED SYMMETRIC, BALANCED LAMINATES UNDER IN-PLANE LOADING

Some of the phenomena that arise in the practical design of flight quality hardware are discussed sequentially to explain why design ground-rules were created. To avoid complications that hinder simple explanations, it is assumed, unless stated otherwise, that only in-plane loading (N_x, N_y, and N_{xy}—see Figure 4-2) is applied and that plies will be laid up only in the 0°, ±45°, and 90° directions, defined as shown in Figure 4-1.

To illustrate the influence of lay-up on laminate strength and stiffness, it is convenient to use carpet plots of strength and stiffness of ideal, undamaged laminates subject to uniaxial or shear loading. Carpet plots for a typical high strength graphite epoxy at room temperature are presented in Reference 4. These plots are reproduced in Figures 5-1 through 5-5. The strengths are predictions based on first ply failure theory under the assumption of linear stress-strain behavior of the laminate up to ultimate failure. Looking at Figure 5-1, it can be seen that the tension strength for 100% 0° plies is 180 ksi, for 100% 90° plies it is ~8 ksi, and for 100% ±45° plies it is ~22 ksi. Shear strength in Figure 5-2 is almost linear with respect to percentage of ±45° plies beyond ~20% ±45° plies. Figures 5-3 and 5-4 show similar dependency on lay-up for Young's and shear moduli. The dependence of Poisson's ratio on lay-up is illustrated in Figure 5-5. Depending on lay-up, Poisson's ratio can be more than twice the values typically associated with isotropic metals.

In theory, plies can be laid up at any angle the designer chooses leading to a nearly infinite number of possible ply combinations. As discussed in Section 4, however, such freedom comes at the price of increased testing and analysis. Furthermore, standardizing laminates using a small number of ply angles simplifies manufacturing and the training of factory labor. Such manufacturing and economic constraints restrict permissible ply angles in most practical laminates to a handful of discrete values. By far the most common laminates are those of the four angle, 0°/±45°/90° family. Carpet plots, such as those of Figures 5-1 through 5-5 are most widely available for this family of laminates.

Figures 5-1 through 5-5 show a very large range of properties available with one single material layed up at only 0°, ±.45°, or 90° angles. Similar design aids can be created for other materials and laminations.

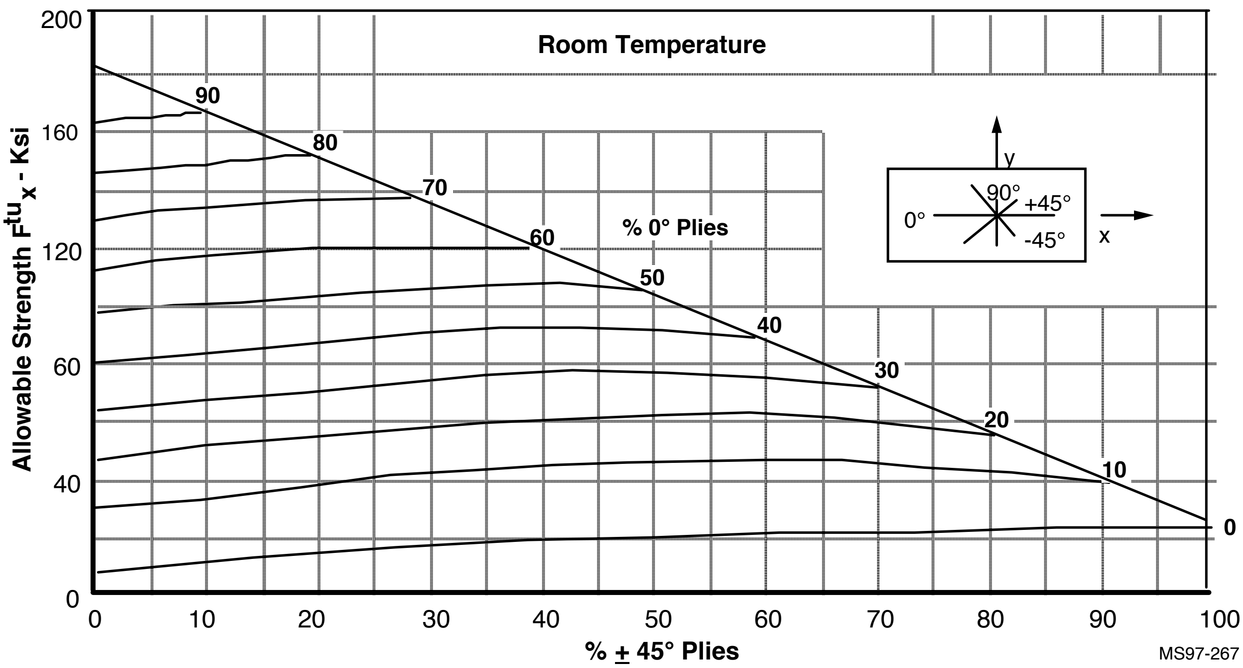


FIGURE 5-1. ROOM TEMPERATURE ULTIMATE TENSION STRENGTH (F^{TU}) FOR HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

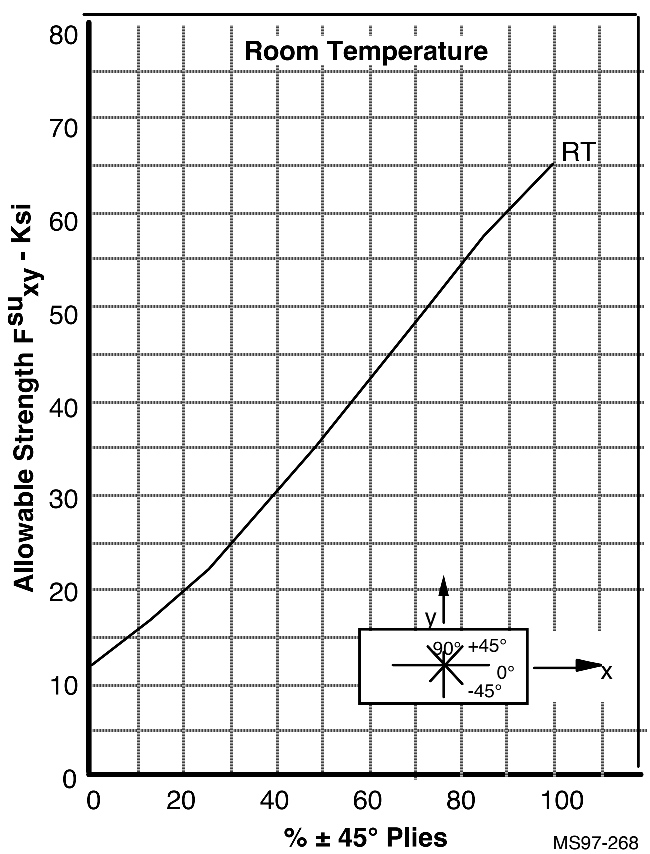


FIGURE 5-2. ROOM TEMPERATURE ULTIMATE SHEAR STRENGTH (F^{SU}) FOR HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

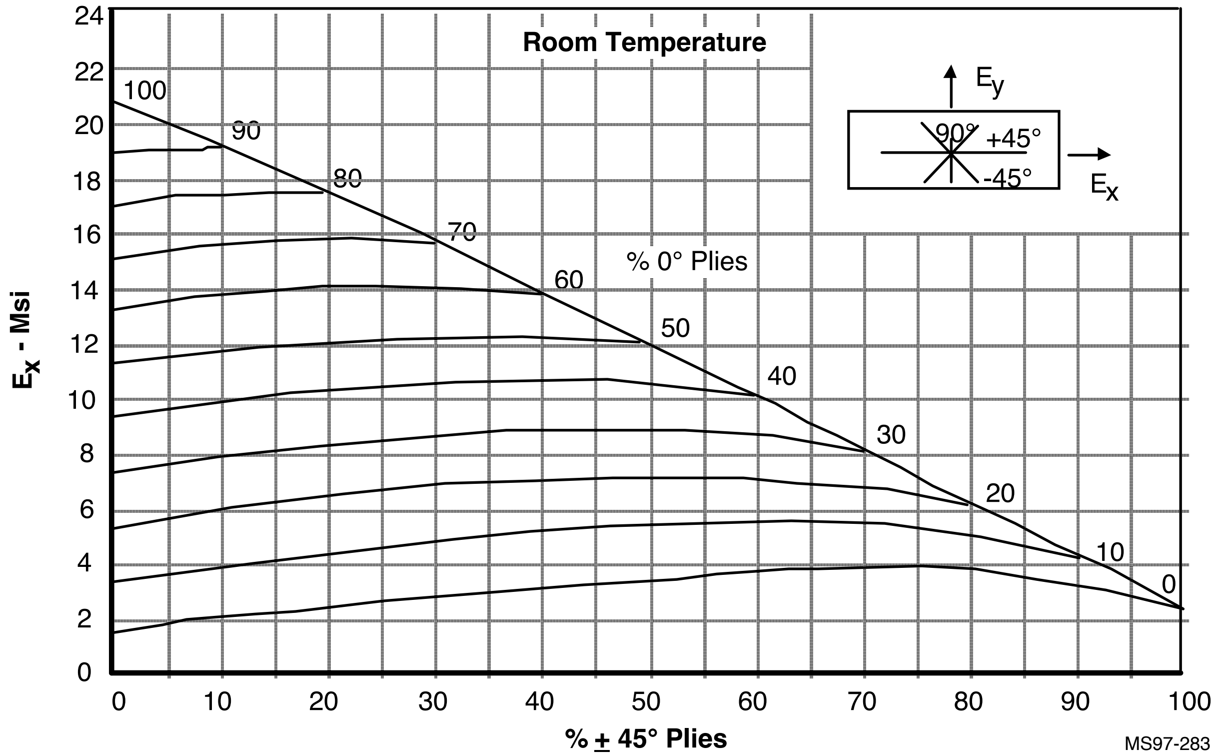


FIGURE 5-3. ROOM TEMPERATURE EXTENSIONAL MODULUS (E_x) FOR HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

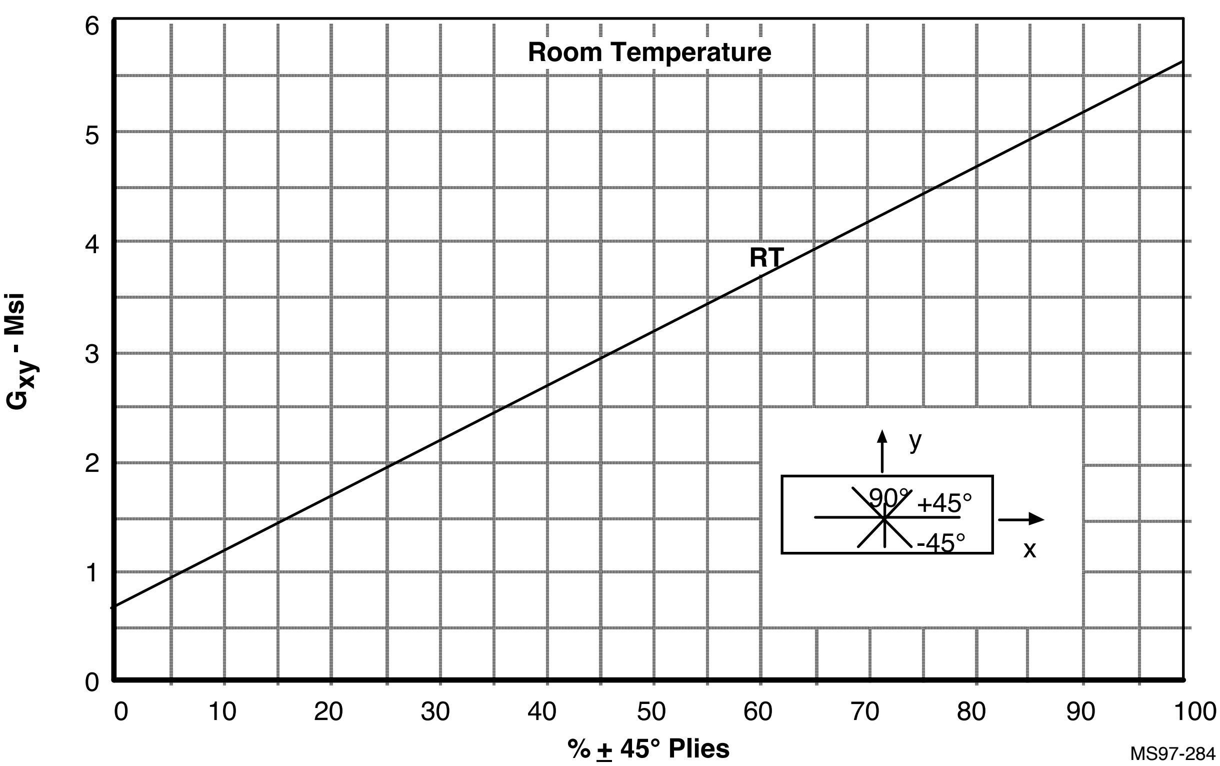


FIGURE 5-4. ROOM TEMPERATURE IN-PLANE SHEAR MODULUS (G_{XY}) FOR HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

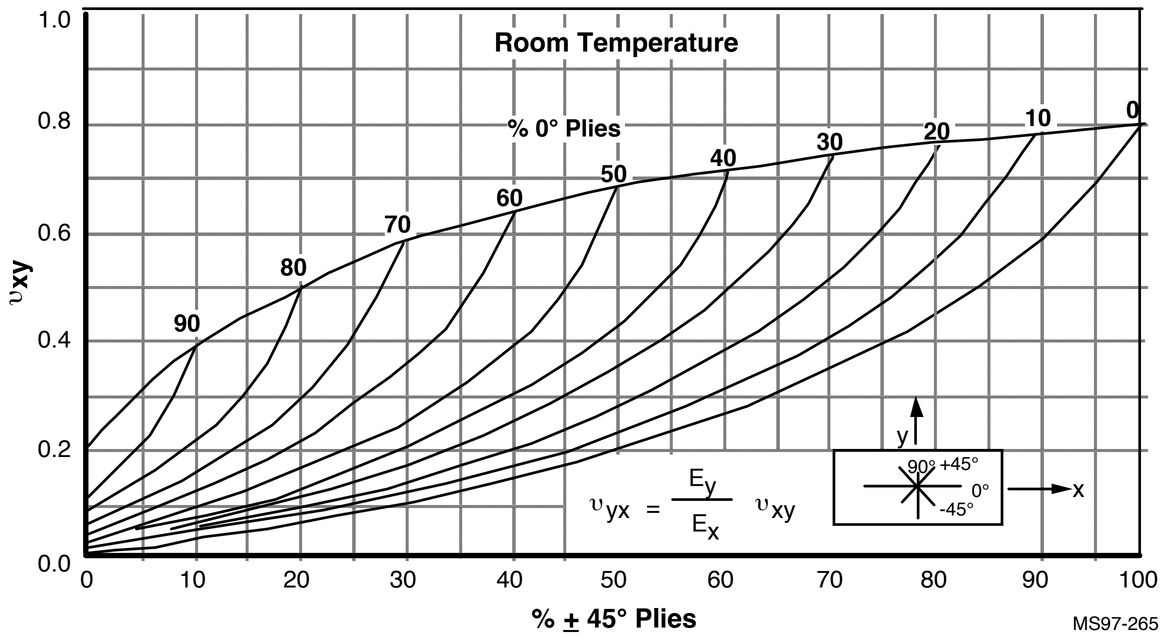


FIGURE 5-5. ROOM TEMPERATURE POISSON'S RATIO (υ_{xy}) FOR HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

Using the carpet plots, appropriate laminates can be selected for any set of design conditions. However, some of these laminates would exhibit one or more of the weaknesses discussed in Section 3. To preclude this, a minimum percentage of fibers in each of four directions is often specified to ensure fibers carry most of the in-plane loads. Guidelines and reasons for this fiber-dominated design philosophy are presented in the sections that follow.

SECTION 6

FIBER-DOMINATED UNNOTCHED LAMINATES

In theory, ply fractions (percentages) at each orientation angle may be specified as anywhere from 0 to 100%. For example, if the loading is defined as being uniaxial along the x axis, there might be a temptation to specify a laminate with 100% 0° plies. Using Figures 5-2 through 5-5, the following properties are derived.

$F^{tu}_{x} = 180.$ ksi: $F^{tu}_{y} = 8.$ ksi: $E_{x} = 20.8.$ msi: $E_{x} = 1.8.$ msi:

$F^{su}_{xy} = 11.5$ ksi: $G_{xy} = 0.7$ msi: $\upsilon_{xy} = 0.2$:

These are the properties of a very orthotropic laminate, having small strength and stiffness with respect to transverse (y-direction) and shear loadings.

Specification of such a highly orthotropic laminate is undesirable for the following reasons.

1. While the primary loading is dictated to be uniaxial, secondary loadings in other directions often exist that are not accounted for in the design. Consider the following example witnessed by one of the authors. A strut designed primarily to transmit axial load for a communication satellite payload failed catastrophically during a thermal-vac test meant to expose the delicate electronics payload to the space environment—not to test the structure. Failure occurred when the atmospheric pressure contained inside the strut split it due to hoop stresses. Had only a single hoop ply been provided, this very expensive failure would have been avoided.
2. For non-zero, in-plane shear forces, a situation similar to that of item 1 above develops. The shear strength and stiffness of such panels is small so cracks in the resin matrix can develop easily.
3. The stress-strain relationships of laminates having matrix-dominated characteristics can be highly nonlinear. This is illustrated by the shear stress-strain curve shown in Figure 6-1. The nonlinear stress-strain behavior complicates the prediction of structural response.

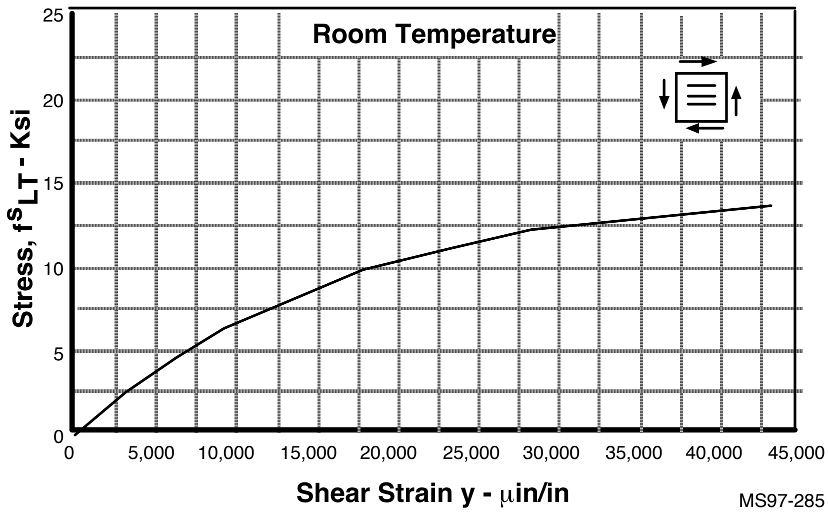


FIGURE 6-1. TYPICAL ROOM TEMPERATURE IN-PLANE SHEAR STRESS-STRAIN CURVE FOR HIGH STRENGTH GRAPHITE EPOXY [REF. 4]

The in-plane normal stress-strain curve in the resin matrix dominated 2 direction is similarly nonlinear (see Figure 6-1). Hence, any laminate containing only 0° plies would exhibit a significantly nonlinear response to transverse or shear loading. This adds considerably to the complexities of the analysis and the experimental characterization of such laminates. There are strong pragmatic reasons for keeping most major structural global response linear, at least up to design limit load. Foremost among these reasons is the increase of resources consumed in the analysis and testing of significantly nonlinear structures. Local nonlinearity is acceptable in specific situations, such as thin skins in stiffened panels designed to operate into the postbuckled range. However, even in this situation, the overall load-displacement relation of the stiffened panel is still more or less linear. Further examples of the complexities arising due to nonlinear, resin matrix-dominated behavior are discussed by Eckstrom and Spain [Ref. 11].

4. Loadings in the transverse (2) direction of laminates with 100% 0° plies could result in creep. Excessive creep can result in unacceptably low fatigue life.

5. Laminates with plies in only one direction are susceptible to crack propagation. Once a crack develops in the resin matrix, resistance to its propagation is minimal. A great contributor to the excellent fatigue lives exhibited by fiber-dominated laminates is that the bridging of the fibers across the crack significantly hinders its propagation. In axial tension testing of laminates with saw-cuts, it was found that failure due to splitting (cracking along the main load-bearing fibers, parallel to the principal load direction) occurs whenever the percentage of 0° plies exceeded 60%. The minimum

percentage of 0° plies at which splitting occurs is a strong function of matrix toughness.

6. Resin matrix microcracks also allow fluid and gas leaks that are unacceptable for pressure cabins and fuel tanks. Also, moisture ingress can cause structural damage during the freeze/thaw cycles that occur as the vehicle altitude changes.

For these reasons, Guideline 4 was established.

Guideline 4 Laminates Will Be Fiber Dominated, Having at Least 10% of Their Plies in Each of the 0°, ±45°, and 90° Directions.

This guideline is intended to preclude the types of problems just described. It is often referred to as the 10% rule. There is no formal documentation substantiating the rigor, uniqueness, or validity of this guideline. However, the guideline has been followed with great success on a number of production programs and has, hence, survived to the present day. The constraints imposed on laminate performance by the 10% rule can be illustrated by indicating, using cross-hatching, disallowed laminate designs in the design carpet plots which are reproduced from Figures 5-1 through 5-5 in Figures 6-2 through 6-6. Use of this smaller subset of designs leads to usable laminates that are more robust in the sense that they are less susceptible to the weaknesses associated with highly orthotropic laminates.

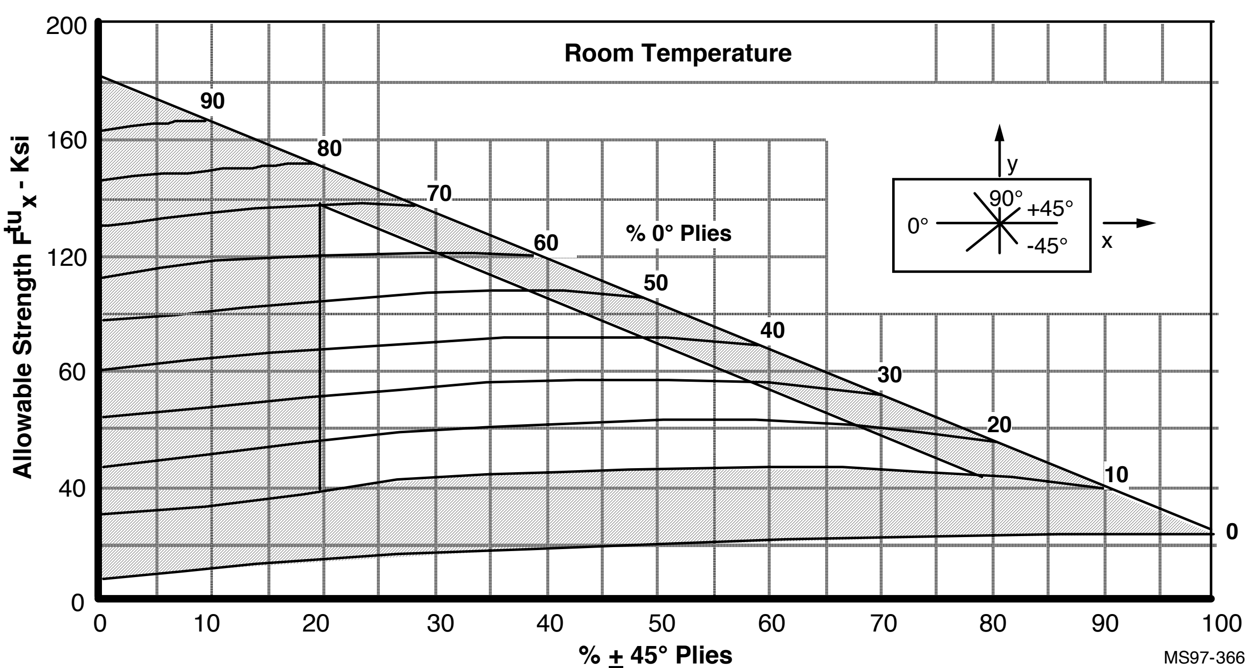


FIGURE 6-2. ROOM TEMPERATURE ULTIMATE TENSION STRENGTH (F^{TU}) OF USABLE HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

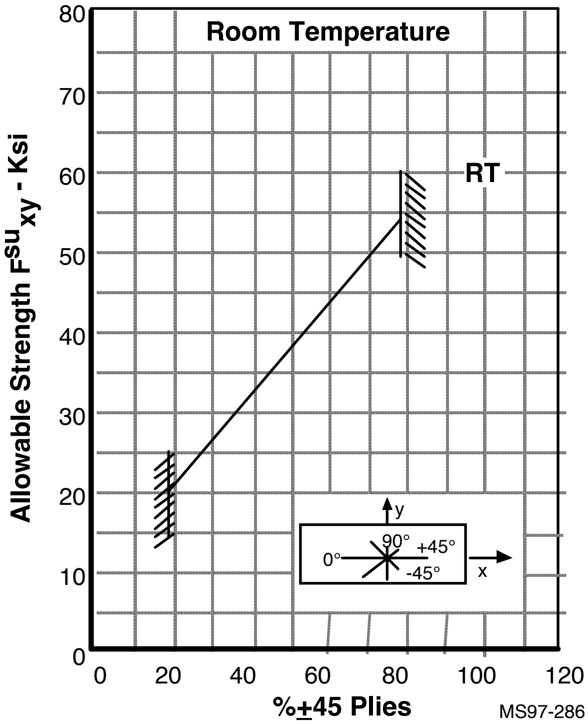


FIGURE 6-3. ROOM TEMPERATURE ULTIMATE SHEAR STRENGTH (F^{su}) OF USABLE HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

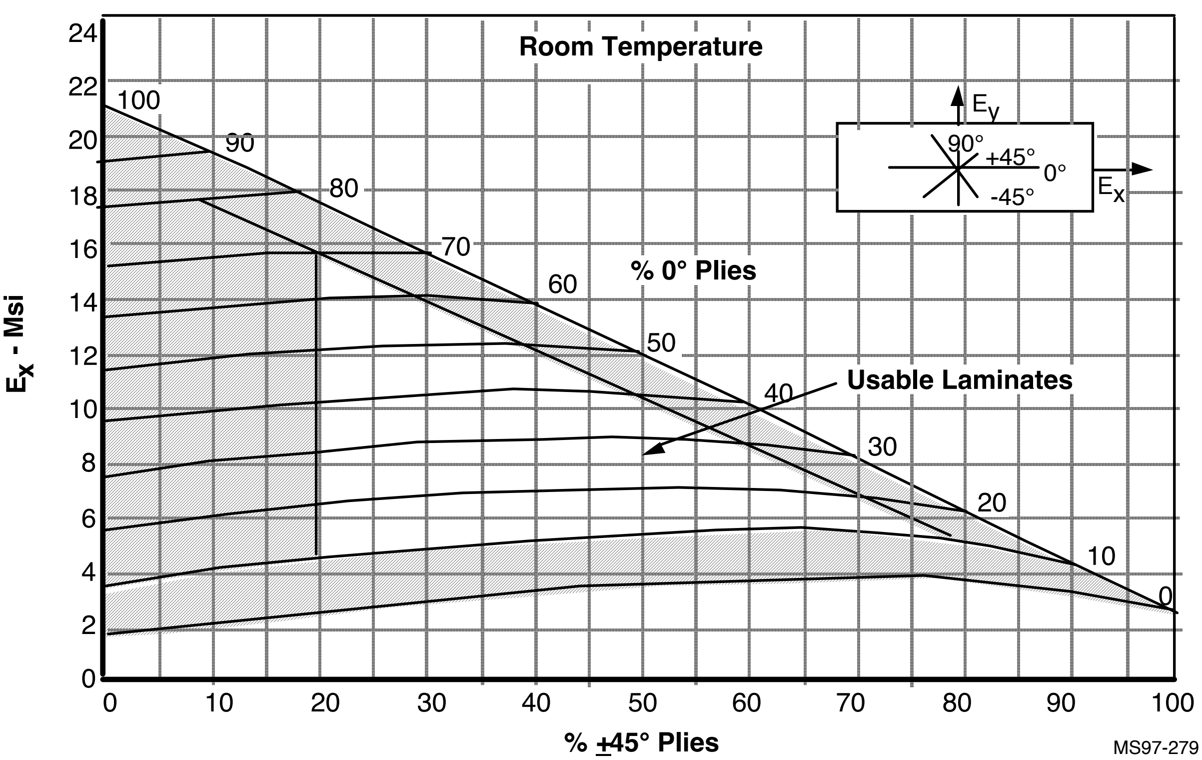


FIGURE 6-4. ROOM TEMPERATURE EXTENSIONAL MODULUS (E_x) OF USABLE HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

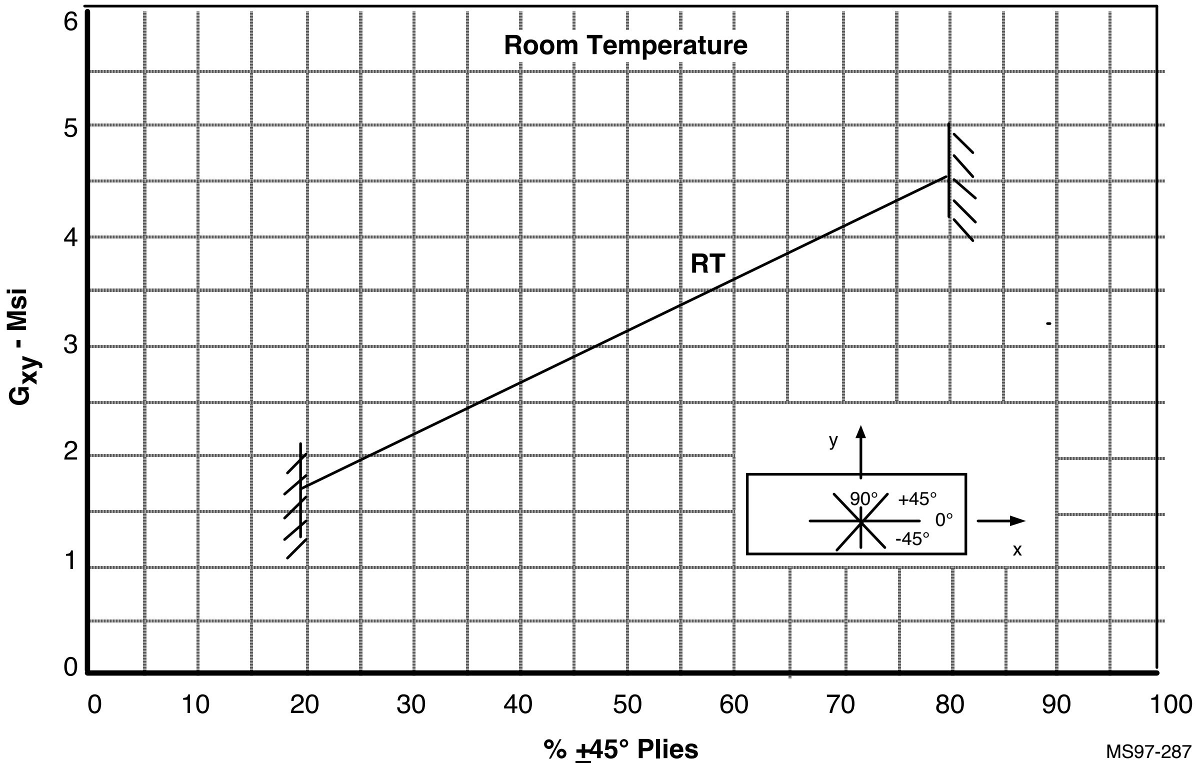


FIGURE 6-5. ROOM TEMPERATURE IN-PLANE SHEAR MODULUS (G_{XY}) OF USABLE HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

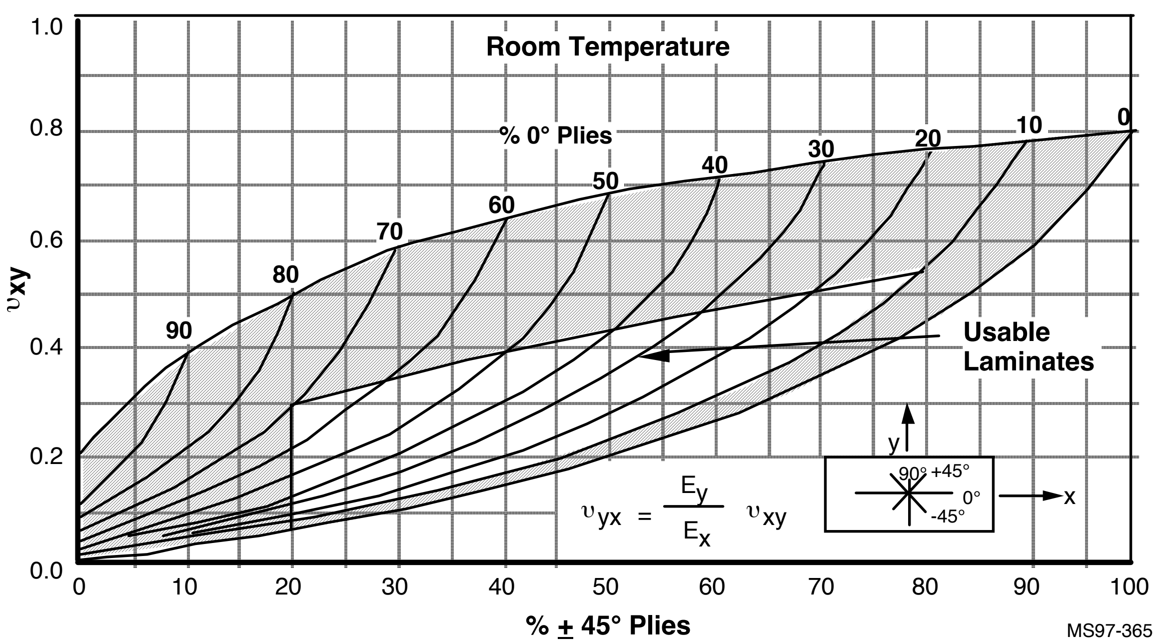


FIGURE 6-6. ROOM TEMPERATURE POISSON'S RATIO (υ_{XY}) OF USABLE HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

Using Figures 6-2 through 6-6, the laminate properties for the 80/10/10 laminate obeying the 10% rule that replace those of the original 100/0/0 laminate are:

F^{tu}_{x} = 150. ksi: F^{tu}_{x} = 34. ksi: E_x = 17.4 msi: E_x = 4.0 msi:

F^{su}_{xy} = 15.0 ksi: G_{xy} = 1.2 msi: υ_{xy} = 0.2:

It is important to note that the 10% fiber-dominated guideline is often interpreted differently with regard to the ±45° plies. Some project directives require there be least 10% +45° and 10% -45° plies, rather than 5% of +45° and -45° plies. There are no guidelines that establish a rigorous differentiation between these two alternative minimum 45° ply contents. Other projects have issued guidelines requiring at least 6% (rather than 10%) 90° plies <u>provided</u> there are at least 20% ±45° plies.

Guideline 5 Keep a Reasonable Number of Primary Load Carrying Plies Away From the Outer Surfaces.

This is done so that these critical plies are not easily damaged by minor impacts. The outer plies of moderately thick laminates, damaged by the impact, protect those underneath them. Cloth plies are sometimes specified for outer surfaces [Ref. 19] for the following reason quoted directly from the reference:

> "A prime advantage of eight harness satin weave cloth is its resistance to splintering. Damage remains very confined and, therefore, easy to repair."

Guidelines 1, 2, 4, and 5 have been developed to improve overall laminate strength and stiffness. Section 7 contains additional guidelines for improving laminate stability.

SECTION 7

STABILITY OF FIBER-DOMINATED LAMINATES

Guideline 6 Laminates Should Be Symmetric and Balanced to Maximize Buckling Strengths.

Due to the large number of parameters involved in buckling-resistant composite panel design, specific guidelines valid for all combinations of planforms, lay-ups, and loadings are not easily identified. However, the following comments by Jones summarize the motivation behind Guideline 6 [Ref. 3, Section 5.6].

> "The presence of coupling between bending and extension in a laminate generally increases deflections. Hence, coupling decreases the effective stiffness of a laminate. At the same time, this coupling reduces buckling loads and vibration frequencies significantly. Similarly, for laminates with twist-curvature coupling, the deflections are increased, the buckling loads decreased, and the vibration frequencies decreased."

The desirability of symmetric, balanced laminates in buckling-critical structures is also discussed in Reference 12.

In symmetric, unbalanced laminates and some thin, symmetric balanced laminates D_{16} and D_{26} are significant with respect to other terms of the [D] matrix leading to a flexurally anisotropic rather than a flexurally orthotropic panel. The equations governing stability of anisotropic panels do not admit closed-form solution. This lack of closed-form solutions necessitates the use of numerical solution techniques such as the finite element method. This method was employed by Fogg [Ref. 13] to produce the results reviewed by Leissa [Ref. 12, p. 123 et seq.]. The finite element method was used to obtain buckling loads for an axially-loaded, simply supported panel. The influence of bend-twist coupling on the buckling load of this panel is summarized in Figure 7-1.

Figure 7-1 is an example of how bend-twist coupling reduces the buckling load relative to the buckling of the same panel with D_{16} and D_{26} set to zero. In Figure 7-1, N_x (ANISO) is the buckling load of the panel having $D_{16} = D_{26}$ nonzero. These curves are plots of a subset of results listed by Leissa [Ref. 12, Table 5-3]. These results substantiate his following quotation:

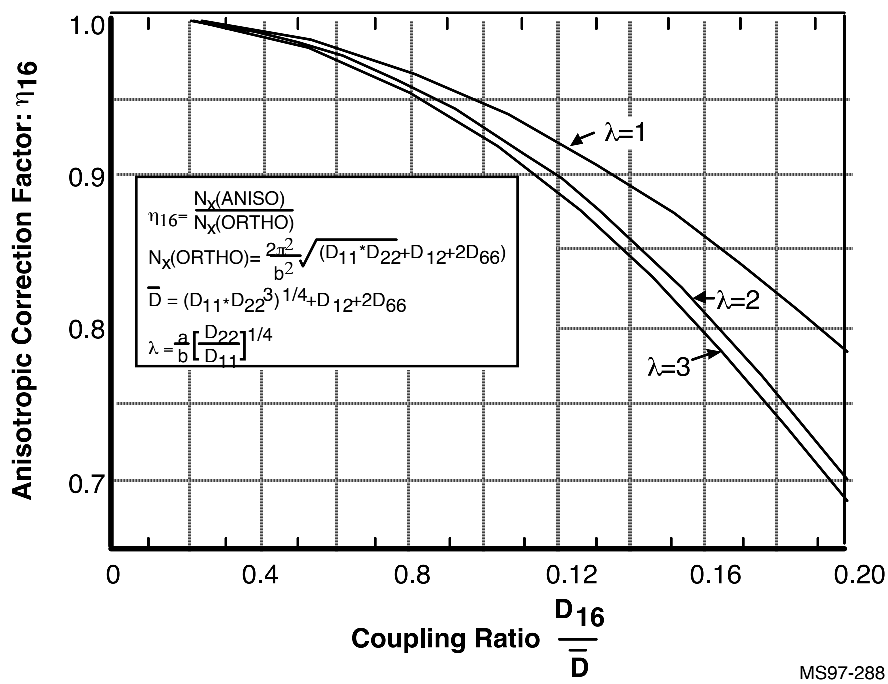


FIGURE 7-1. BUCKLING CORRECTION FACTOR DUE TO DIRECT-SHEAR COUPLING [REF. 13]

"1. The buckling loads are always decreased.

2. In the case of the uniaxial compression, longer plates (larger λ) always have larger decreases.

3. Shear buckling loads are more greatly decreased than those due to uniaxial compression."

Recent work by Nemeth, et al., has confirmed the results documented by Leissa and Fogg and quantified the effects of flexural anisotropy on the buckling for numerous rectangular panels subjected to a wide range of loadings [Refs. 12-18]. The degree of anisotropy is quantified by the two parameters γ and δ defined in Figure 7-2. The effect of γ and δ on the buckling load is shown as the ratio of anisotropic buckling coefficient to that for the corresponding specially orthotropic plate with $D_{16} = D_{26} = 0$. Anisotropy reduces buckling loads in all cases. The greater the anisotropy, as indicated by the magnitude of γ and δ, the greater is the buckling load reduction.

Guideline 7 Use of ±45° Plies on the Outer Surfaces Is Recommended for Stability Critical Laminates.

The stacking sequence may be influenced by the need to maximize one or the other major bending stiffnesses (D_{11} or D_{22}), depending on the loading direction(s) [Ref. 12]. However, the

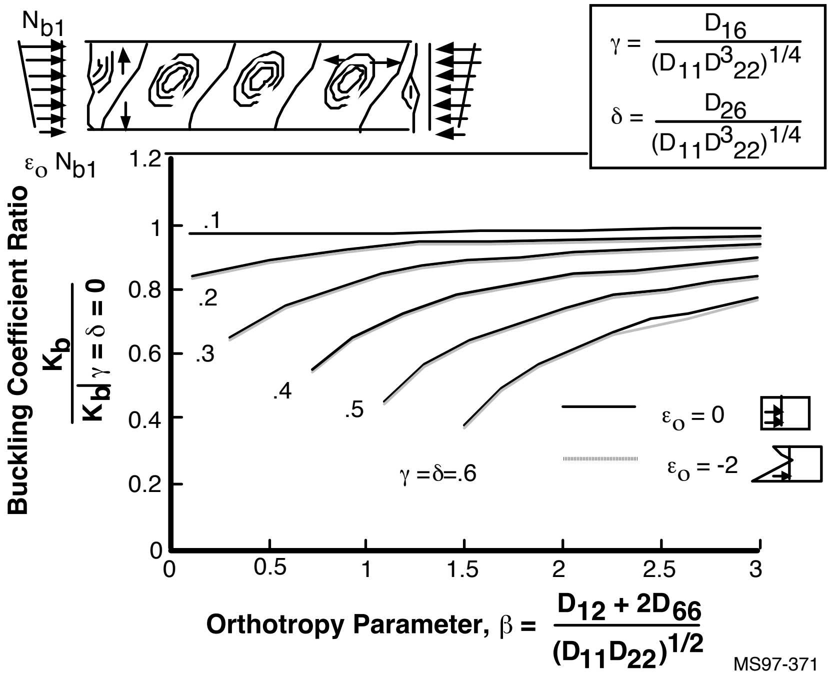


FIGURE 7-2. EFFECT OF ORTHOTROPY PARAMETER β AND ANISOTROPY PARAMETERS γ AND δ ON BUCKLING COEFFICIENTS FOR SIMPLY SUPPORTED PLATES SUBJECTED TO LINEARLY VARYING EDGE LOADS [REF. 16]

buckling resistance is maximized in most cases by locating the ±45° plies on the outer surfaces. The reason for this can be explained by examining the equation for the buckling load of a long orthotropic panel loaded axially [Ref. 3, p. 261].

$$N_{xcr} = \pi^2 \left[D_{11} \left(\frac{m}{a}\right)^2 + 2\left(D_{12} + 2D_{66}\right)\left(\frac{1}{b^2}\right) + D_{22}\left(\frac{1}{b^4}\right)\left(\frac{a}{m}\right)^2 \right]$$

where a and b are the panel length and width and m is the number of half waves in the buckling mode shape along the lengthwise direction.

Experimental observations indicate that the value of m typically yielding a minimum value of N_{xcr} results in (m/a) approximately equal to (1/b). Hence, N_{xcr} is a function of four times D_{66} while only a function of one times D_{11} (D_{22}). Adding ±45° plies remote from the middle surface, thus increasing D_{66}, is four times more effective in raising the buckling load than adding 0° or 90° plies.

SECTION 8

THERMAL RESPONSE OF LAMINATES

Guideline 8 Significant CTE Mismatches Among Bonded or Cocured Structures Must Be Avoided.

Laminate CTEs are a strong function of lay-up due to severe differences in CTE in the lamina 1 and 2 directions. Significant residual stresses can be built up during the cool-down phase of the cure cycle. Hence, CTE must be carefully considered. The following guidelines have been established to help the designer manage CTE-related problems.

Guideline 9 Laminates Should Have Greater Than 10% 0° and 90° Plies to Avoid Excessive Thermal Coefficients of Expansion.

For symmetric, balanced laminates, the laminate membrane thermal strains depend linearly on the CTE of the laminate. In Figure 8-1, it is seen that the CTE is large in laminates with a very small percentage of 0° plies. Furthermore, α_y is large when there are few 90° plies. (In fact, Figure 8-1 can be used to determine α_y if the percentage of 0° plies is read as the percentage of 90° plies.) Excessive values of CTE can be avoided by enforcing the fiber-dominated laminate philosophy. In Figure 8-2, this philosophy has been imposed resulting in a usable subset of laminates exhibiting moderate CTEs. Control of laminate CTE is particularly important if the laminate is bonded or bolted to a metal structure and operates in a severe thermal environment since thermal loading of the bonded or bolted joint is sensitive to CTE mismatches.

Guideline 10 Resin Matrix Toughness Must Be Great Enough to Prevent the Occurrence of Intralaminar Cracks During Cool-Down From the Stress-Free Temperature.

It is instructive to look at the intralaminar thermal stresses generated when 90° and 0° plies are laid up in contact as shown in Figure 8-3. It is assumed that boundary effects can be ignored and that each ply is homogeneous. It is also assumed that a stress-free temperature exists from which all temperature excursions (ΔT) are measured.

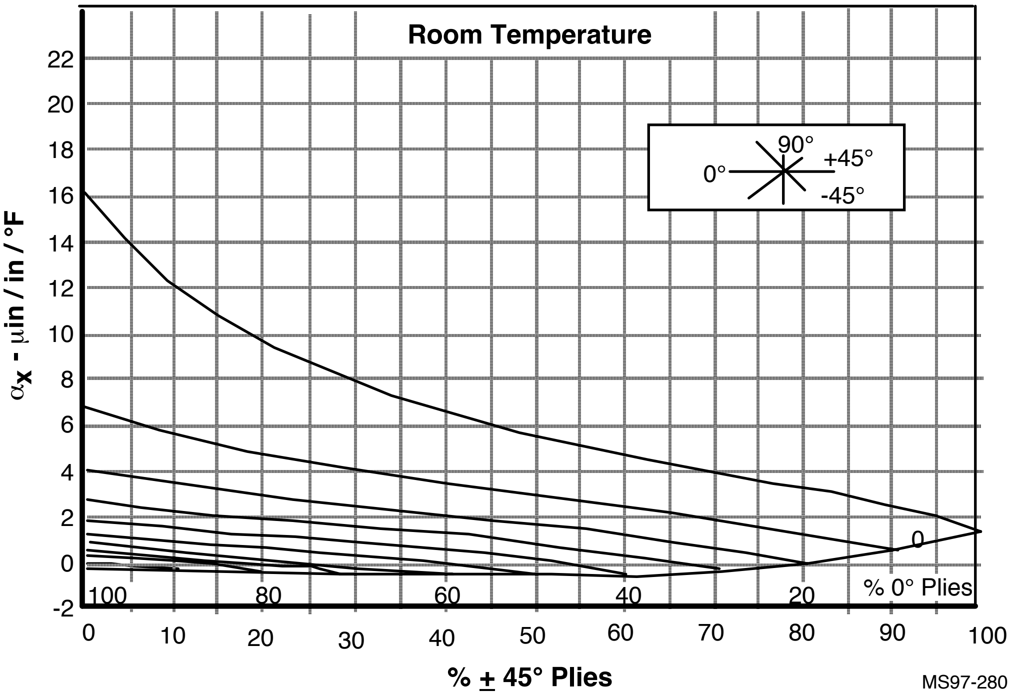


FIGURE 8-1. ROOM TEMPERATURE COEFFICIENT OF EXPANSION (α_x) OF HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

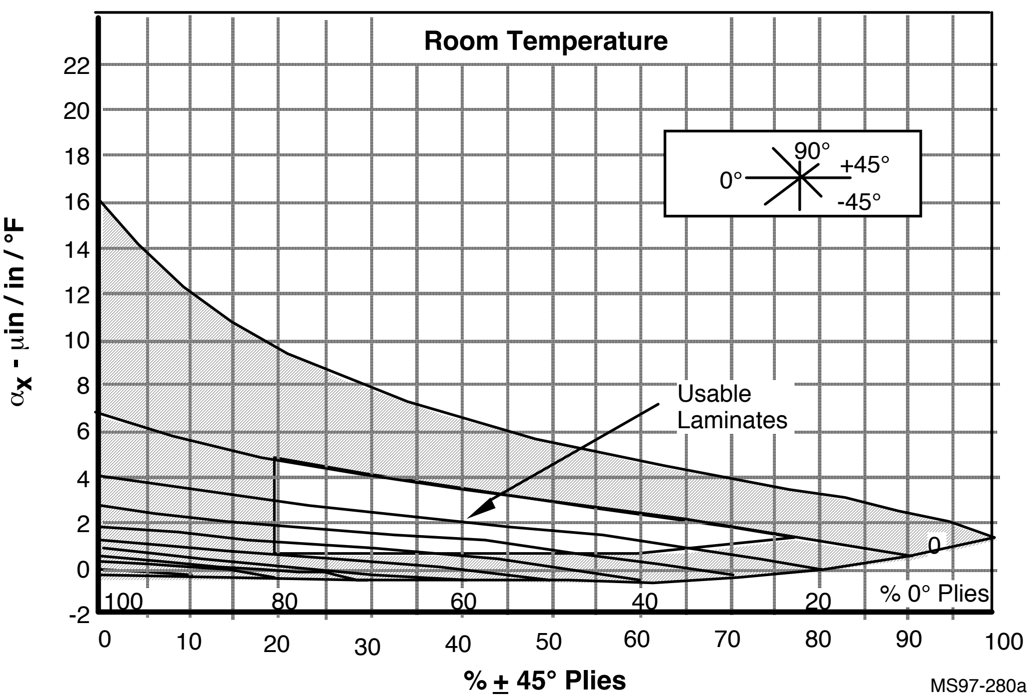


FIGURE 8-2. ROOM TEMPERATURE COEFFICIENT OF EXPANSION (α_x) OF USABLE HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]

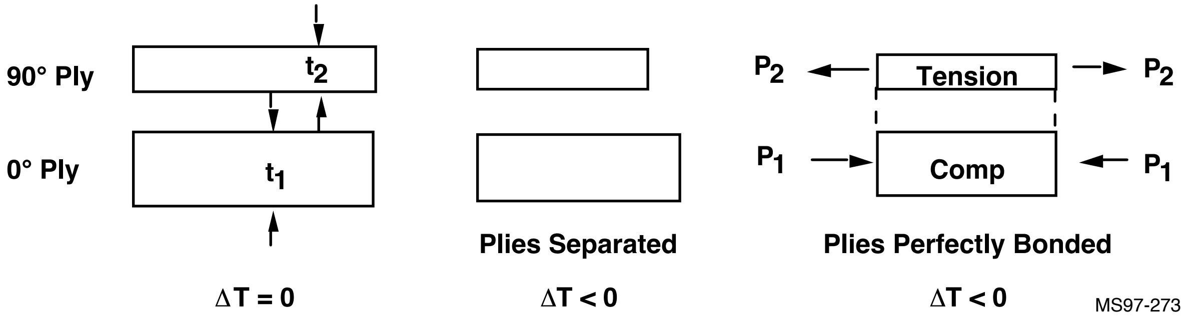


FIGURE 8-3. INTRALAMINAR THERMAL STRESSES IN 90° AND 0° PLIES

Consider first these two plies disconnected from each other and subjected to a uniform temperature drop (ÆT < 0), well away from the free edges. Each ply will contract in proportion to its individual CTE. Now consider the two plies rigidly connected. Their final displacements must be equal—the 90° ply contracting less than it would on its own, the 0° ply contracting more. As no external forces are applied, the resulting internal forces generated must be equal in magnitude and opposite in direction.

Using the simple analysis of a bimetallic strip outlined by Popov, predicated on the assumption that the strains in the two plies $\varepsilon_{0°}$ and $\varepsilon_{90°}$ are equal [Ref. 20].

$$\varepsilon_{0°} = \alpha_{11}\Delta T + P_1 / \varepsilon_{11}t_1 = \alpha_{22}\Delta T - P_2 / \varepsilon_{22}t_2 = \varepsilon_{90°} \tag{8.1}$$

where t_1 and t_2 are the two ply thicknesses. For self-equilibrating forces, $P_1 = P_2$. Hence,

$$P_1\left[1/ \varepsilon_{11}t_1 + 1/\varepsilon_{22}t_2\right] = \left(\alpha_{22} - \alpha_{11}\right)\Delta T \tag{8.2}$$

As typical data, assume:

E_{11} = 20.0 Msi $\alpha_{11} = 0.1 \times 10^{-6}$ in/in/°F

E_{22} = 2.0 Msi $\alpha_{22} = 10. \times 10^{-6}$ in/in/°F $t_1 = t_2 = 0.05$ in

which are typical values for a single ply of graphite epoxy. Substituting these values into Equation 8.2 yields

$$P_1/\Delta T = (10. - 0.1)\ 10^{-6} / [1/(20 \times .05) + 1/(2.0 \times .05)]\ 10^{-6} = 0.9 \text{ lb/°F}$$

resulting in equal but opposite ply stresses of 18.0 psi /°F.

For a laminate peak curing temperature of 350°F (typical for graphite/epoxy), cooling down to 75°F results in a ΔT of -275°F and thermal stresses of ±4,950 psi (assuming that the peak curing temperature is approximately equal to the stress-free temperature). The ply mechanical strains resulting from this temperature change are

$$\varepsilon_{11}^{m} = \left(\sigma_{11} / E_{11}\right) = \left(-4,950 / 20.\text{x}10^{6}\right)$$

$$= -247.5 \ \mu\text{strain}$$

$$\varepsilon_{22}^{m} = \left(\sigma_{22} / E_{22}\right) = \left(-4,950 / 2.\text{x}10^{6}\right)$$

$$= +2,475 \ \mu\text{strain}$$

This idealized, elastic, homogeneous ply analysis shows a high tensile strain exists in the 90° ply. This high strain has the potential to cause cracking of the 90° ply resin matrix. Next consider the case where the first ply (t_1 = .05 in) is broken into several layers .01-inches thick and placed in various configurations about the second ply. With bending effects neglected, the calculated ply mechanical strains are the same no matter what lamination scheme is specified. Near free edges, however, induced thermomechanical strains are a strong function of lay-up. Free-edge effects are discussed in Section 9.

Guideline 11 The Maximum Operating Temperature (MOT) of the Laminate Should Be at Least 50°F Below the Wet Glass Transition Temperature.

The properties of fiber-dominated laminates loaded in tension are not particularly sensitive to operating temperature. The reason for this lack of sensitivity is that load transmission is predominantly through the fibers whose processing temperatures are an order of magnitude above the peak laminate curing temperature. However, at elevated temperatures, in the presence of moisture, the compression and shear properties of the resin matrix degrade seriously. This degradation is due to plasticizing (softening) of the resin matrix exposed to a hot/wet environment that reduces its ability to support the fibers and increases the likelihood of fiber microbuckling.

Matrix softening occurs due to the effect of moisture on glass transition temperature (T_g). The T_g is the measure of a limit beyond which resin matrix stiffness and strength drop precipitously. An example of the dependence of T_g on moisture content for AS4/5250-3 graphite/BMI is provided in Figure 8-4. The dry T_g (zero moisture content) is nearly 160°F above the post-cure temperature of 425°F. The drop in T_g is close to 100°F over the range of laminate moisture contents from 0 to 1.2% moisture. A literature survey failed to produce similar data showing T_g dependence on moisture content of graphite/epoxy laminates; however, the trends are believed to be similar for most composites made with polymer resin matrices.

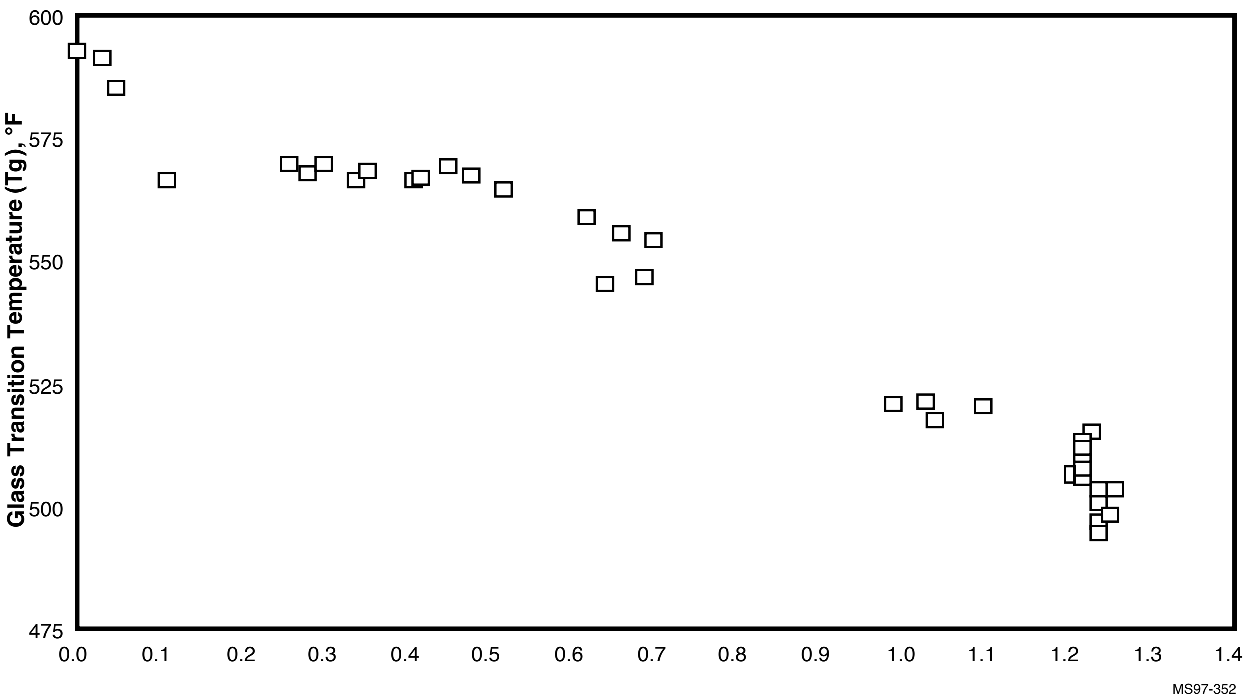


FIGURE 8-4. EFFECT OF MOISTURE CONTENT ON T_G FOR AS4/5250-3 GRAPHITE BMI UNIDIRECTIONAL TAPE LAMINATES (NORTHROP GRUMMAN DATA)

The results depicted in Figure 8-4 indicate a sensitivity of T_g to operating environment. Use of 50°F below the T_g at maximum operational moisture content as the maximum operating temperature (MOT) is suggested in References 21 and 22 on the basis of work that was focused on the certification of graphite epoxy composites. The reason for specifying a 50°F difference in MOT and wet T_g is that as the temperature approaches T_g, the slope of the strength versus temperature curve is very steep. Hence, having the wet T_g no closer to the MOT than 50°F adds a margin of safety against the effects of statistical scatter in the measured value of T_g. The drop-off in compression strength, relative to room temperature strength, for a well-known 350°F cure graphite epoxy is depicted in Figure 8-5.

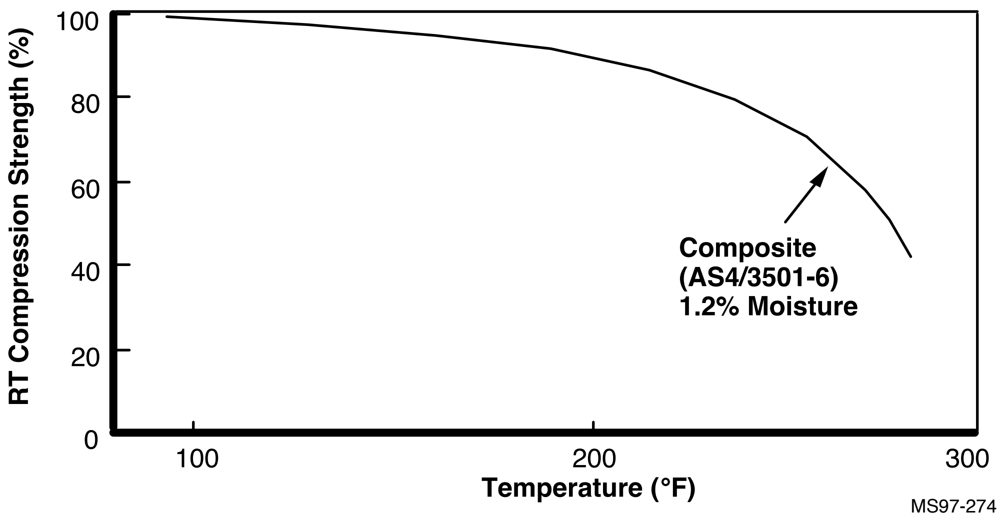


FIGURE 8-5. COMPRESSIVE STRENGTH DEGRADATION OF AS4/3501-6 FIBER-DOMINATED LAMINATES AT ELEVATED TEMPERATURES [REF. 21]

From a preliminary design standpoint, the decades of industry experience with aerospace composites suggests that maximum regular use temperature for production aircraft should be roughly 80°F to 100°F below the maximum cure temperature. Hence, 250°F curing epoxy composites on production aircraft are usually limited to use at 180°F, 350°F cure epoxies to use at 250°F, and 450°F post-cure BMIs to use at 350°F.

SECTION 9

STACKING SEQUENCE AND INTERLAMINAR FREE EDGE STRESSES

The guidelines discussed so far refer only to overall ply orientation percentages and how these plies may be distributed about the laminate mid-plane. So far, no attempt has been made to dictate ply grouping or how many plies at the same orientation angle can be laid up in contact with each other. To address the issue of stacking sequence, it is necessary to investigate interlaminar stresses at the free edges of laminates. These are ignored in classical lamination theory but discussed by Pipes, et al. [Refs. 23-28].

Guideline 12 Free Edge Interlaminar Stresses Due to CTE and Poisson's Ratio Mismatches Between the Plies Must Be Considered an Integral Part of the Laminate Design Process.

All discussions up to this point have been predicated on the assumptions of classical lamination theory. Classical lamination theory is invalid within a boundary or edge zone roughly one laminate thickness from any free edge. Interlaminar stress in laminates subjected to membrane loading arise in this edge zone from differences in Poisson's ratio between adjacent plies that have different orientation angles, and differences in the CTE of these plies.

Guideline 13 Limit Layer Thicknesses Within Laminates to 0.020 in. or Less.

The existence of interlaminar normal and shear stresses in laminate edge zones is shown by Pipes, et al., to depend on the stacking sequence [Refs. 23-28]. The existence of interlaminar stresses is the reason why some laminates fail at lower membrane loadings than others having the same percentage of plies at each orientation angle but different stacking sequences. The 0.02-inch thickness included in the guideline corresponds to four plies of typical graphite-epoxy material having a ply thickness of ~0.005 inches. Hence, this guideline is sometimes worded as allowing no more than four contiguous plies at the same orientation angle.

The existence of interlaminar stresses can be explained by considering a single layer made of n contiguous plies as a free body, as illustrated in Figure 9-1. It is assumed that the laminate is subjected to membrane load in the x direction and a uniform temperature change, ÆT, with respect to the stress-free temperature.

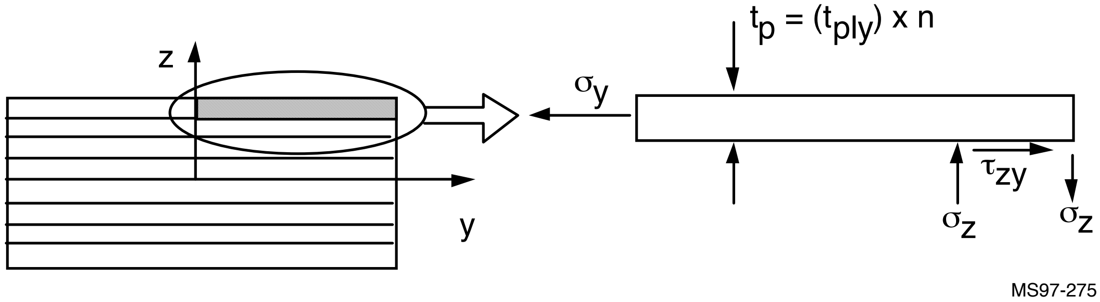


FIGURE 9-1. OUTER PLY AS FREE BODY TO ILLUSTRATE INTERLAMINAR STRESSES [REF. 23]

Based on CLT, the stress σ_y exists at the center (y = 0) of the ply. This stress arises because of the differences in Poisson's ratio and CTE (with respect to the laminate x-y axes) of adjacent laminae at different orientations. Clearly, membrane stresses must be zero at the edges. The normal force arising from σ_y is balanced by the shear force arising from the stresses τ_{zy} near the free edge.

The existence of interlaminar normal stress (σ_z) can be illustrated with the aid of Figure 9-1. The shear force arising from the interlaminar shear stress, τ_{zy}, is not collinear with the normal force arising from σ_y; hence, a moment (couple) arises. This moment must be balanced by an equal and opposite moment that arises near the free edge and induces the normal stress.

The magnitude of the interlaminar stresses σ_z and τ_{zy} are a function of the lamina thicknesses. Thicker laminae lead to greater interlaminar stresses. It has been demonstrated that laminae containing a large number of contiguous plies of older, brittle-matrix composite, equivalent to a combined thickness of about 0.02 in, can microcrack and delaminate near a free edge. Hence, the guideline restricting the number of contiguous plies within a lamina was created. In composites made of newer, tougher matrices, a greater lamina thickness may be allowed as experience with these new materials grows.

Theory explains and experiments confirm that interlaminar stresses decay within about one laminate thickness from any free edge [Refs. 23-24]. Precisely quantifying the magnitude and spatial distribution of free-edge interlaminar stresses requires sophisticated analyses. Furthermore, a predictive capability for local failure is needed. The energy release rate method embodied by Wang and Crossman is one such predictive capability [Ref. 26]. The work cited by Pipes, et al., demonstrates that lay-up has a strong influence on two different modes of failure at a free edge: (1) transverse cracks through the thickness of a group of interior plies, as illustrated in Figure 9-2, and (2) free-edge delamination along the midsurface, as shown in Figure 9-3 [Refs. 23, 24, 26].

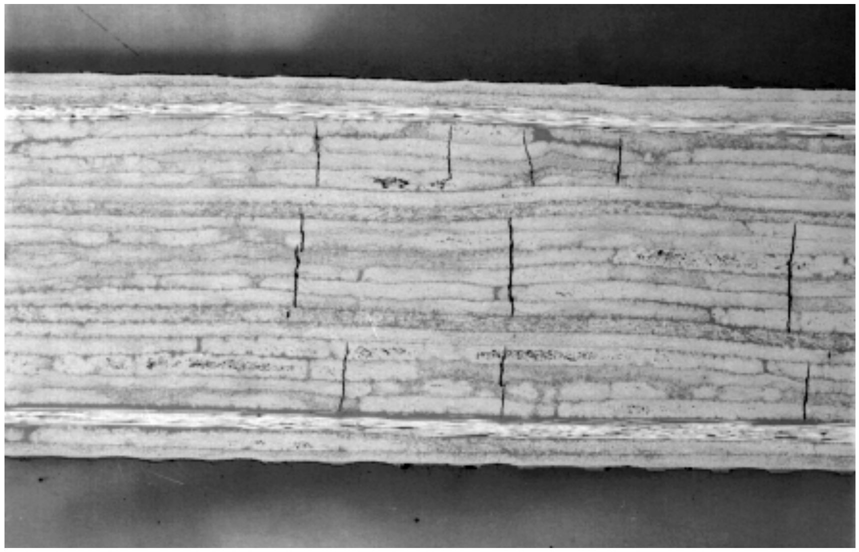


FIGURE 9-2. MULTIPLE TRANSVERSE CRACKS IN A $[0/90_n]_s$ LAY-UP [REF. 26]

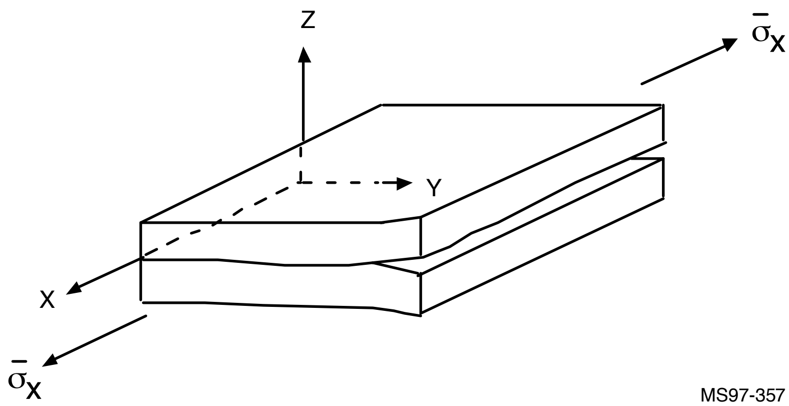


FIGURE 9-3. GEOMETRY OF A FREE EDGE DELAMINATION [REF. 26]

Note that theories put forth by Pipes, et al., are based on the assumption of homogeneous plies [Refs. 23, 24, and 26]. No distinction is made between fiber and matrix. In practice, the micromechanical interaction of fiber and matrix at ply boundaries is a complex nonlinear problem. Derivation of general failure criteria for all materials, lay-ups, and environments is not yet feasible. Reliance on experimental data is essential. As demonstrated by Long and Swanson, the newer, tougher epoxy resin matrices are effective in reducing strength loss associated with edge delamination [Ref. 28].

Based on the results documented by Wang and Crossman, the larger the thickness of centrally located 90° plies, the lower the applied σ_x stress at which transverse cracks of the type shown in Figure 9-2 appear [Ref. 26]. As this loading is increased, crack density becomes greater until a crack saturation level is reached. In Figure 9-4, total laminate strain at transverse crack initiation is listed for several cross ply graphite-epoxy laminates. The data indicates that the thicker the central 90° ply, the lower the laminate strain at transverse crack onset.

0° Ply Thickness (mm)	90° Ply Thickness (mm)	Laminate Strain at Onset of Transverse Cracks (%)
0.5	0.500	0.33
0.5	0.250	0.36
0.5	0.125	0.49
0.5	0.062	0.66

FIGURE 9-4. TRANSVERSE CRACK ONSET IN [0/90]$_s$ LAY-UP (FROM [26])

Plots of peak interlaminar normal stresses (σ_z) at the free edge of several laminates taken from Niu is shown in Figure 9-5 [Ref. 1]. This figure shows how laminates having the same number of plies at each orientation can, depending on stacking sequence, have very different interlaminar stress distributions. For example, laminate 2 exhibits a higher tendency to delaminate than laminate 3 due to the fact that the peak interlaminar normal stress in laminate 2 is tensile while that in laminate 3 is compressive.

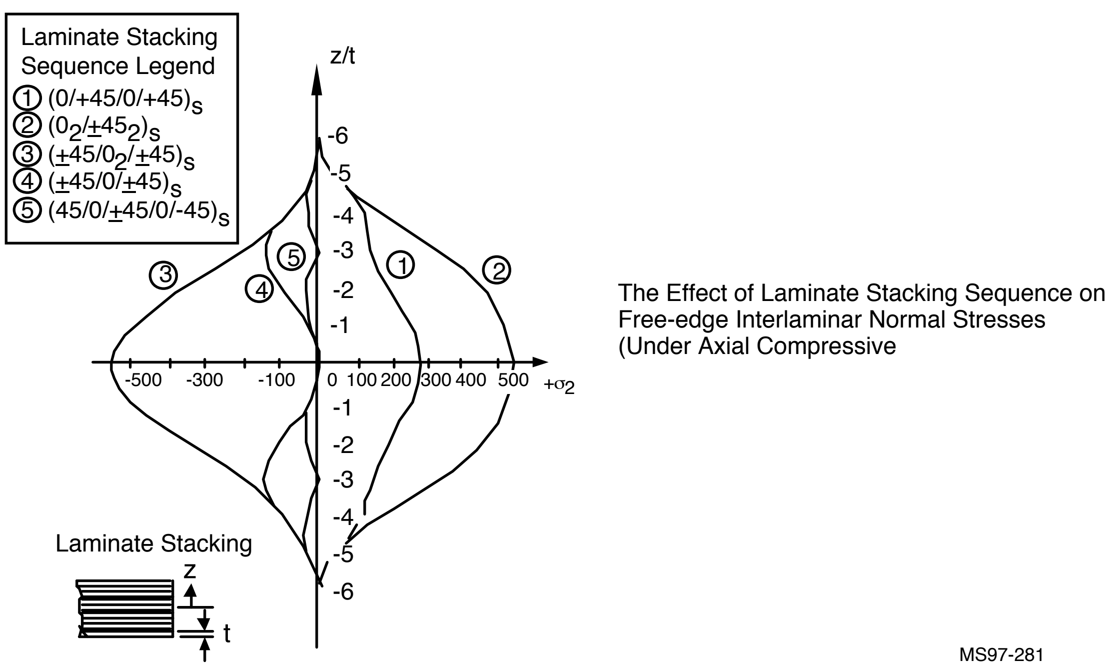


FIGURE 9-5. EXAMPLE OF INTERLAMINAR DIRECT EDGE STRESSES [REF. 1]

SECTION 10

POISSON'S RATIO MISMATCH BETWEEN LAMINATES AND BONDED OR COCURED STIFFENERS

Guideline 14 The Poisson's Ratio Mismatch Between a Stiffener and Panel That Are Bonded or Cocured Together Should Be Less Than 0.1.

As seen in Figure 6-6, a wide range of Poisson's ratios can arise in laminate design. A design feature where performance is sensitive to Poisson's ratio is a stiffener bonded or cocured to a laminate. There may be sound reasons for designing the stiffener with a fairly high percentage of 0° plies along its axis. There may be equally sound reasons for designing the skin laminate to be relatively rich in ±45° plies. The resulting situation for tension loads in the skin is shown in Figure 10-1. The Poisson contraction of the skin is resisted by the stiffener resulting in severe stresses at the skin/stiffener interface.

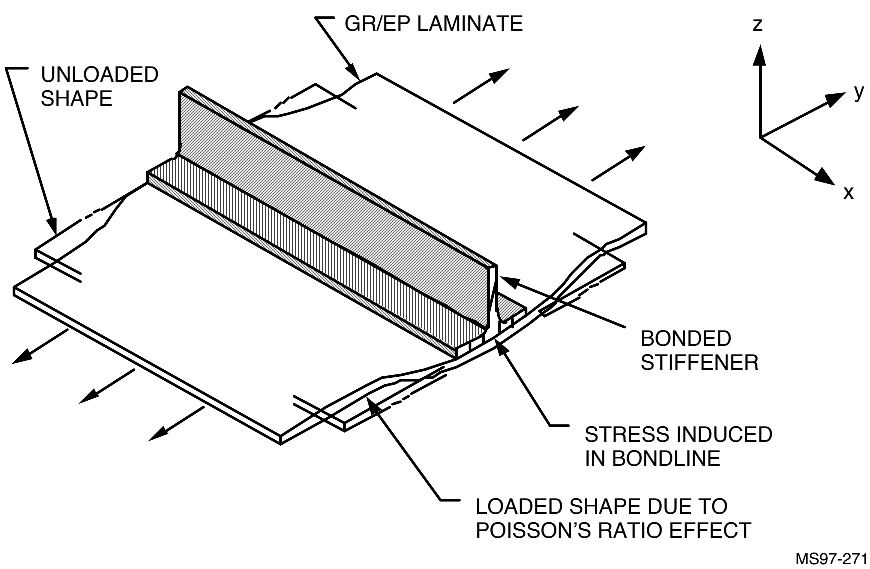


FIGURE 10-1. STRAIN DUE TO POISSON'S RATIO DIFFERENCES IN SKIN AND BONDED STIFFENER [REF. 1]

As an example, the initial designs could be as follows: Stiffener is a [60/20/20] lay-up. Enter Figure 6-6 assuming a [20/20/60] lay-up since the principal load in Figure 10-1 is in the y orientation. Hence, $\upsilon_{xy} = 0.21$. Skin laminate is a [10/80/10] lay-up. Based on Figure 6-6, $\upsilon_{xy} = 0.52$.

This Poisson's ratio mismatch of 0.34 is a cause for concern. The mechanical strains induced at the skin/stiffener interface by this mismatch can, depending on the applied load, be high enough to cause failure. Industry experience with these structures has taught that the difference in Poisson's ratio between a stiffener and a bonded or cocured skin laminate should be limited to a value of ≤ 0.1. The design of the example obviously fails this criterion. The numerical value of 0.1 does not appear to be substantiated by any quantitative, published supporting data. It is based on experience with older, brittle composites and adhesives. The more brittle the composite/adhesive, the smaller the mismatch needs to be to induce failure at the interface.

Returning to the example problem, it may be desirable to execute the following redesign. Stiffener is a [60/30/10] lay-up. Entering Figure 6-6 with a [10/30/60] lay-up, $\upsilon_{xy} = 0.36$. Skin laminate is a [10/70/20] lay-up. Based on Figure 6-6, $\upsilon_{xy} = 0.41$. This satisfies the criterion of a Poisson's ratio mismatch less than 0.1.

SECTION 11

HOLES, CUTOUTS, AND IMPACT DAMAGE

Holes, cutouts, and/or impact damage complicate the design of composite structures. Sections 5 and 6 illustrate a number of guidelines for fiber-dominated laminates free of holes, impact damage, and manufacturing imperfections. Typical carpet plots were presented from which the stiffnesses and strengths of these laminates can be determined. In the case of production airframes, the assumption of freedom from holes, cutouts, impacts, and imperfections is unrealistic. Composite laminates applied to real structures seldom achieve the properties predicted using the carpet plots because of their notch sensitivity. The question is not whether holes or damage exists, but to what level of damage (hole size) the structure should be designed. As a result of decades of research into the durability, damage tolerance, and certification of composite structures, a widely accepted consensus has been reached that can be illustrated with the aid of Figure 11-1 [Refs. 29, 37-45]. The influence of notches on compression strength is more significant than the influence of notches on tension strength. Hence, attention is restricted here to response to compression loading.

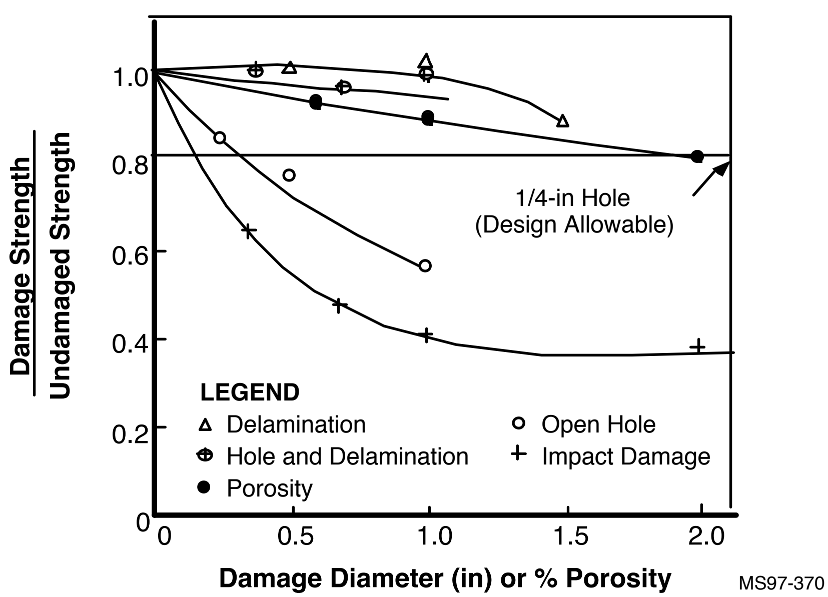


FIGURE 11-1. INFLUENCE OF DEFECT OR DAMAGE TYPE ON COMPRESSION STRENGTH OF TYPICAL FIBER-DOMINATED GRAPHITE EPOXY LAMINATES [REF. 29]

In Figure 11-1, the normalized static compression strength for laminates containing five types of damage is depicted. The results shown in Figure 11-1 demonstrate that minor manufacturing imperfections such as porosity and small delaminations have little effect on strength, relative to the effects of holes and impacts. Hence, attention is focused on the latter two types of damage. Major manufacturing anomalies are assumed to be detected prior to first flight so that structures containing these anomalies can be replaced or repaired before a failure can occur.

In selecting the notch type and size to be designed into the structure, the following four points should be considered.

1. A majority of the laminates in service today are less than 3/8-inches thick. Hence, a vast majority of fasteners used to join these laminates are approximately 1/4 inches in diameter. Therefore, mechanically-fastened laminates may contain a large number of 1/4-inch holes.
2. The fidelity of nondestructive inspection techniques lead to the conclusion that rogue (undetected) flaws had to be accepted in production flight hardware. These rogue flaws included porosity, damaged fibers, small inclusions, and impact damage from dropped tools.
3. There are many databases for military aircraft containing data on the relationship between strength degradation and damage delectability. These data support the idea that an impact causing barely visible impact damage (BVID) produced a level of strength loss approximating that caused by the presence of a 1/4-inch diameter hole. This similarity of the effect on strength of BVID and a 1/4-inch hole is by no means precise. Laminate thickness, lay-up, planform, edge support, fibers, resin matrices, impactor shape, impact location, and environment all influence the correlations between the effect on strength of BVID with that of a 1/4-inch hole.
4. Measurements of the strength of laminates containing a 1/4-inch hole are generally repeatable and consistent. Use of impacted laminates to measure strength requires careful control of many more independent variables and results in data exhibiting excessive scatter. Hence, the effects of damage are investigated separately on a case-by-case basis.

Based on the above considerations, basic design allowable laminate strengths used in industry for initial design are based on coupon tests of laminates with 1/4-inch holes. The strengths of the structural laminates with impact damage are then checked after the structural design matures. While the various certifying agencies (FAA, USN, and USAF) have different precise design requirements, their philosophies are similar. All agree that composite structural elements with 1/4-inch holes or barely visible impact damage must be designed to carry some factor greater than unity times design limit load (the greatest load the aircraft is expected to encounter in service). The FAA requires this factor to be 1.5.

When determining the strengths of laminates containing holes and impact damage, carpet plots such as those depicted in Figures 6-2 through 6-6 created for unnotched laminates, are not relevant. There is no universally accepted standard approach for estimating the strength of notched laminates subject to combined loads. What follows is a frequently used concept. Uniaxial strength, $N_{x,a}$, is considered. This strength is expressed as:

$$N_{x,a} = E_x \, t \, \varepsilon_{x,a} \tag{11.1}$$

where

$N_{x,a}$ = ultimate strength (force/unit length)

E_x = laminate Young's modulus, derived from carpet plots or classical lamination theory, of the pristine (unnotched, undamaged) laminate

t = laminate thickness

$\varepsilon_{x,a}$ = B-basis allowable strain for the appropriate material, lay-up, and environment derived from coupon tests of laminates containing a 1/4-inch diameter hole or BVID (should screening tests indicate BVID yields significantly lower $\varepsilon_{x,a}$).

Modulus and thickness are predictable with entirely acceptable accuracy. The main difficulty is in the estimation, using a minimum amount of experimental data, of the allowable strain. A proper accounting of lay-up, materials, environment, loading interactions, effects of stiffeners, and effects of ply drops across thickness changes requires considerable judgment. Much work remains to be done in this arena. It is important to note that accounting for a 1/4-inch hole is just the first, basic step in the structural design process. Larger holes and damage from higher energy impacts are dealt with as follows.

The curve depicted in Figure 11-2 shows experimentally derived "strength knock-down" factors for laminates containing large open holes. These factors are applied to reduce $\varepsilon_{x,a}$ in Equation 11.1. The curve of Figure 11-2 illustrates the well-known "hole size" effect on the strength of laminated composites. Since the peak stress concentration factor (SCF) at the edge of a circular hole is not a function of hole size, the reason for this effect is not immediately obvious. One explanation for the reduction in strength with hole size is as follows.

While the peak SCF at the edge of the hole is not a function of hole size, the stress distribution near the hole is. As illustrated in Figure 11-3, the stress distribution near the larger

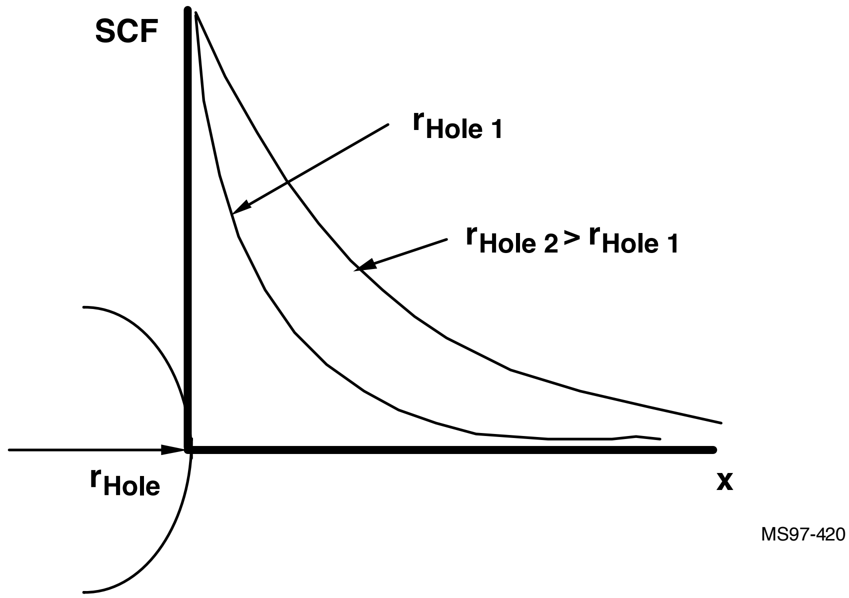


FIGURE 11-2. TYPICAL STRESS CONCENTRATIONS NEAR A CIRCULAR HOLE

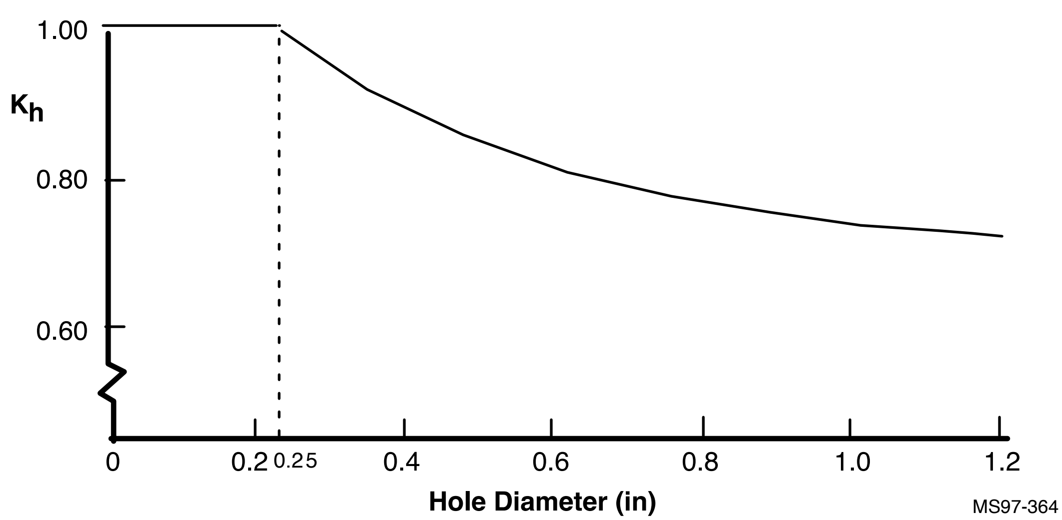


FIGURE 11-3. TYPICAL KNOCK-DOWN FACTOR FOR HOLES IN FIBER-DOMINATED GRAPHITE FIBER LAMINATES (NORTHROP GRUMMAN DATA)

hole results in a greater amount of material reaching a higher stress level than in the case of the smaller hole. With more material at a higher stress, redistribution of this stress due to local failure/delamination at the hole edge is less likely to be effective in halting further progression of this local failure. Furthermore, the probability of a small flaw where failure could initiate existing in a highly stressed region is higher for the laminate containing a larger hole.

The effect of extensive impact damage on laminate strength is seen in Figure 11-4 to be greater than that of holes for impact energies greater than 4 ft-lbs. Hence, care must be exercised when designing composite structure to withstand an impact of specified energy level. The specified energy level is configuration-specific and is typically arrived at through negotiations between the airframe builder and the certifying agency. Once the energy levels are determined and preliminary design using the 1/4-inch hole allowable data is complete, compression strength after impact (CSAI) data such as that of Figures 11-4 and 11-5 are determined and reviewed. Note that the expressions [42/50/0] in Figure 11-4 and [17/66/17] in Figure 11-5 refer to the percentage of 0°, ±45°, and 90° plies in the laminate, respectively.

When the CSAI strain to failure is less than the OHC allowable used for design, the CSAI strain must be used as the measure of structural integrity. Then in Equation 11.1, the CSAI allowable strain replaces $\varepsilon_{x,a}$, the design allowable strain derived with 1/4-inch holes. This reduction in allowable strain may result in a local resizing. Whether or not this is necessary depends on the structural configuration. For example, if the impacted skin is part of a stiffened panel, the stiffeners may provide an alternate load path and may be sturdy enough to carry load after the panel has been impact damaged.

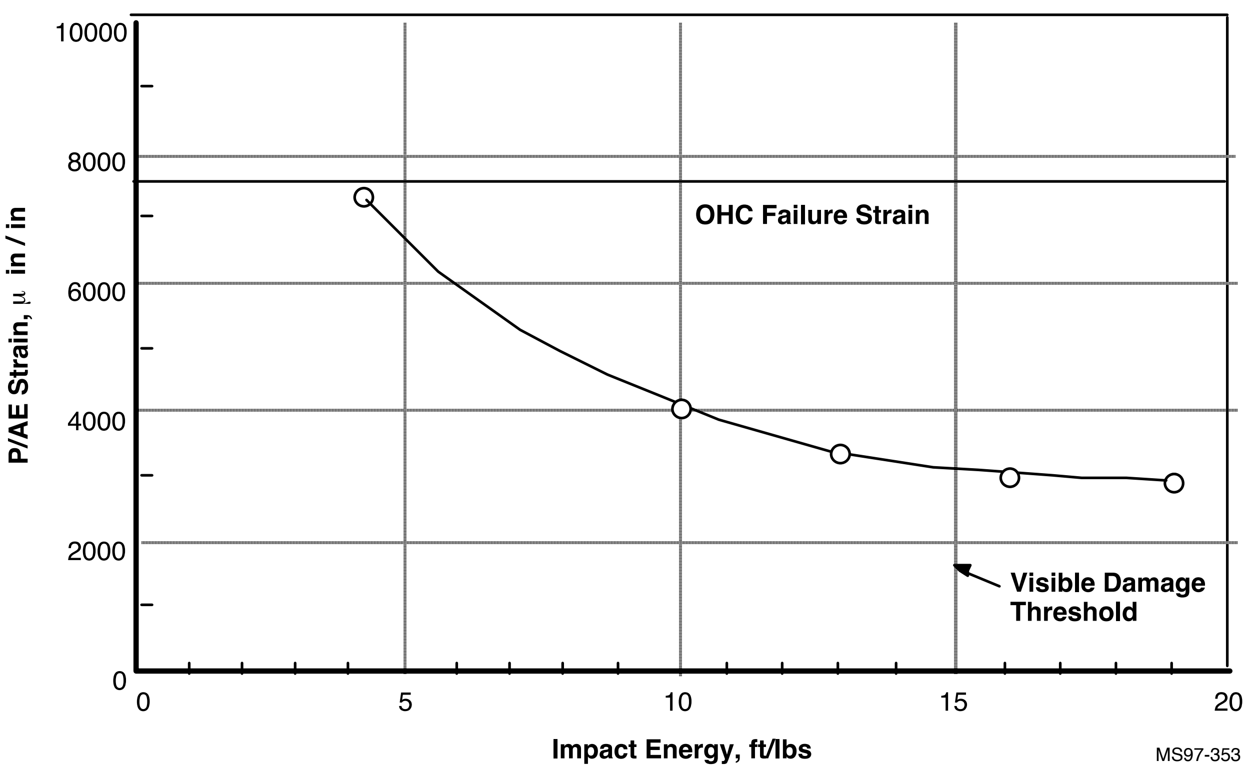


FIGURE 11-4. TYPICAL ALLOWABLE CSAI STRAIN VERSUS IMPACT ENERGY FOR 0.187-IN THICK [42/50/8] AS4/3501-6 TAPE LAMINATES (NORTHROP GRUMMAN DATA)

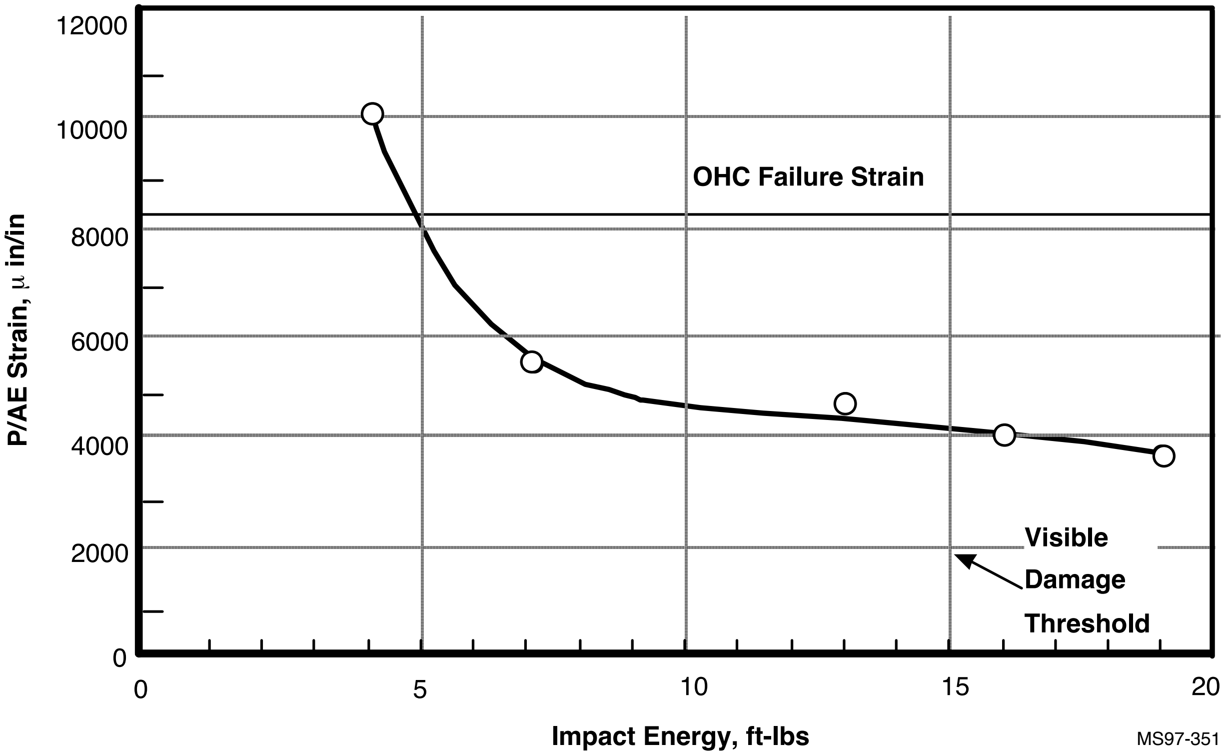


FIGURE 11-5. TYPICAL ALLOWABLE CSAI STRAIN VERSUS IMPACT ENERGY FOR 0.187-IN THICK [17/66/17] AS4/3501-6 TAPE LAMINATES (NORTHROP GRUMMAN DATA)

Given the preceding introduction to the general philosophy behind the design of damage tolerant composite structures, the following guidelines have been established.

Guideline 15 Initial Design Laminated Composite Structures Must Account for the Presence of Fastener Holes, Typically 1/4 in. in Diameter.

The rationale for this guideline has been presented in the preceding paragraphs.

Guideline 16 Final Design of Composite Laminates Must Provide Sufficient Post-Impact Strength.

The rationale for this guideline has been presented in the preceding paragraphs.

Guideline 17 The Maximum Percentage of Plies in Any Direction Will Be 60%.

This guideline prevents laminate splitting parallel to the principal loading axis at holes and cutouts. An area where this splitting failure mode may occur is bolted joints (see Section 12). Furthermore, most operational airframes contain cutouts with removable inspection panels. The relative brittleness of fiber-dominated composites and a virtual absence of ductility up to failure, relative to structural metals, makes composite laminates sensitive to high elastic stress concentrations. Furthermore, orthotropic materials can exhibit much higher stress concentrations at notches than metals. The effect of lay-up on stress concentration factors at holes is illustrated in Figure 11-6.

It is clear from Figure 11-6a that for a laminate rich in 0° plies, the stress concentration is very high. At a circular hole in a unidirectional laminate, the stress concentration factor is roughly 7.2. The physical reason for this high SCF is related to the fact that the load in the plies that are terminated at the hole must be shed into continuous plies next to the hole through in-plane shear. The unidirectional laminate has low in-plane shear stiffness resulting in a significant amount of deformation near the hole as the load is shed around it. The splitting parallel to the loading axis is a manifestation of this deformation coupled with low shear strength. As pointed out by Hart-Smith, the stress concentration increases in laminates with high percentages of 0° plies as fast as the unnotched strength [Ref. 32]. Hence, the notched strength is relatively insensitive to the percentage of 0° plies in the laminate so that adding more 0° plies may do nothing but add weight.

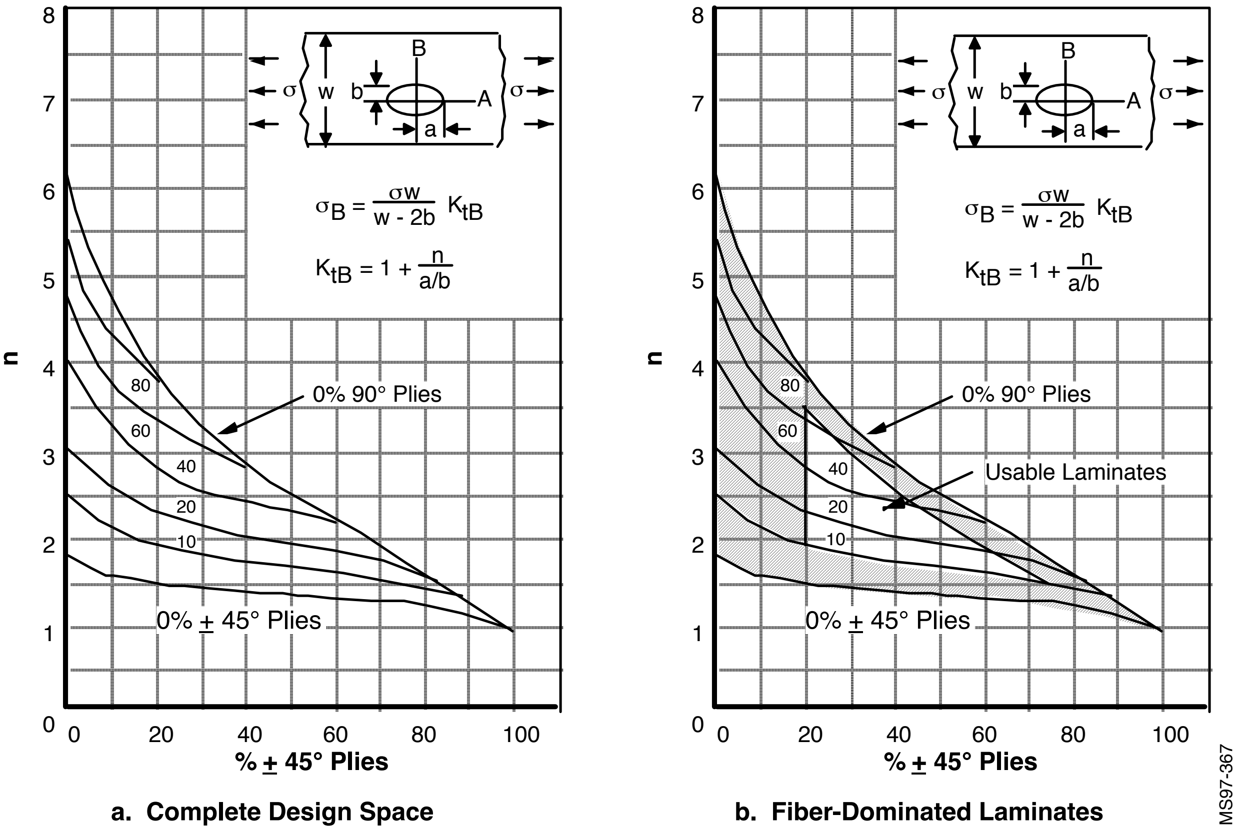
Enforcement of the fiber dominated design philosophy (≤60% 0° plies at any one orientation angle and at least 10% at each orientation angle) around cutouts leads to a set of usable laminates shown in Figure 11-6b where unusable laminate designs fall in the cross hatched regions. Use of the fiber-dominated design philosophy considerably reduces the maximum stress concentration factors at notches. Laminates having a significant amount of area containing cutouts should not be too far removed from quasi-isotropic.

Guideline 18 Use of Generic Design Charts to Size Laminates Containing Large Cutouts Should Be Avoided.

The state of stress adjacent to large cutouts is a complex function of cutout shape, reinforcement, and loading. For holes/cutouts in benign environments with planform dimensions less than about an inch, the use of simple design charts, as outlined previously in this section, is usually adequate. Large cutouts require detailed analyses validated by test.

Guideline 19 Reinforcing Plies Around a Cutout Should Be Interspersed With the Basic Laminate Plies.

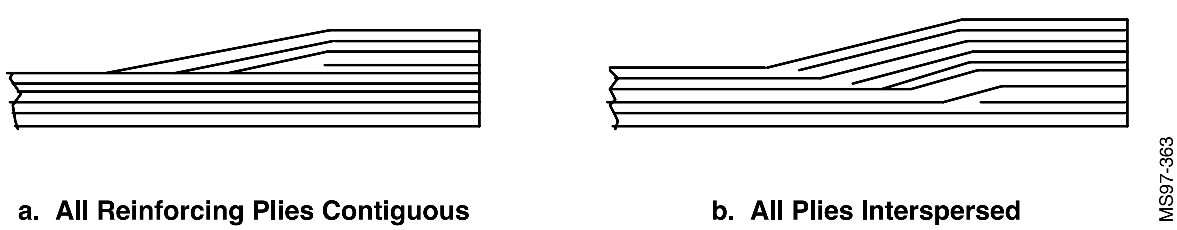
When adding reinforcing plies, or padups, around a cutout, there are two primary ways the pad-up may be created. They are illustrated in Figure 11-7. The pad-up can be laid up with all its plies contiguous, as shown in Figure 11-7a, or it can be laid up by interspersing the pad-up plies with the plies of the basic laminate as shown in Figure 11-7b. The latter method is preferable because transfer of load into the pad-up takes place over many ply interfaces rather than a single one.



a. Complete Design Space

b. Fiber-Dominated Laminates

FIGURE 11-6. STRESS CONCENTRATIONS AT LARGE HOLES IN HIGH STRENGTH GRAPHITE EPOXY LAMINATES [REF. 4]



a. All Reinforcing Plies Contiguous

b. All Plies Interspersed

FIGURE 11-7. INTERSPERSING REINFORCING PLIES AROUND A CUTOUT

SECTION 12

JOINTS

12.1 BOLTED JOINTS

Guideline 20 Laminates at Mechanically-Fastened Joints Should Be Fiber-Dominated, Contain No More Than 60% Plies at Any Single Orientation, and Contain No Less Than 35% ±45 Plies.

While many composite panels are initially installed without mechanical fasteners, they are often designed to allow for bolted repairs. This has lead to constraints being imposed on laminate designs to allow for bolted joints, regardless of the planned presence of such joints in the initial design. The influence of bolted joints on laminate design is presented in a number of papers [Refs. 30-32]. It is pointed out by Hart-Smith that an excessive percentage of 0° plies and deficit of ±45° and 90° plies in uniaxially-loaded joints can lead to cleavage and shear out failures at unacceptably low loads [Ref. 30]. The failures in highly orthotropic laminates involve splitting along the 0° axis, as illustrated in Figure 12-1.

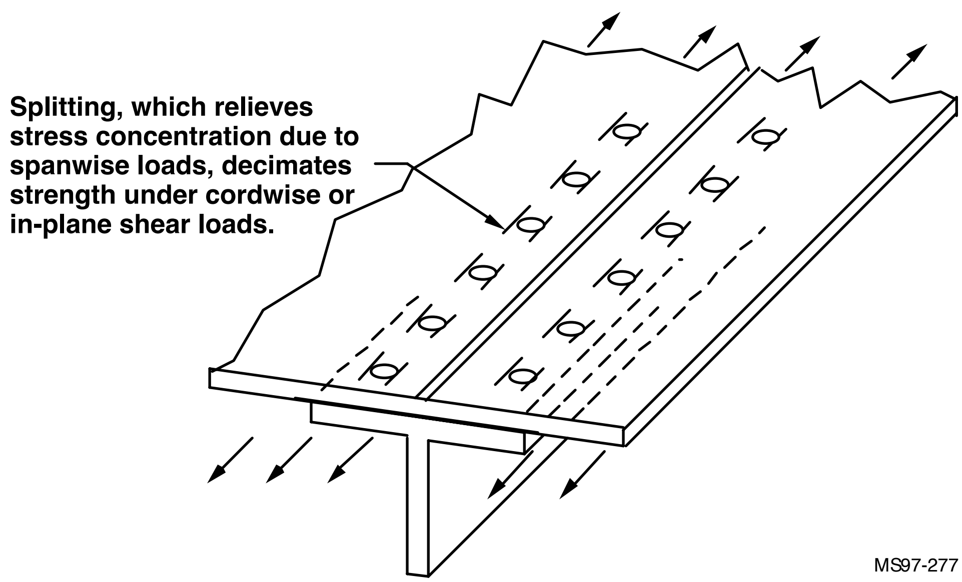


FIGURE 12-1. FAILURE OF HIGHLY ORTHOTROPIC LAMINATES [REF. 32]

Other guidelines have been imposed to enforce a fiber-dominated laminate design philosophy by requiring a minimum of 45% ±45° plies, which restricts the percentage of 0° plies to 45%. The basic theme of such guidelines is that laminates intended for use with bolted joints should not be too far removed from quasi-isotropic, as expressed graphically in Figure 12-2. Use of near quasi-isotropic laminates precludes cleavage and shear-out failures and results in more forgiving bearing-type failures.

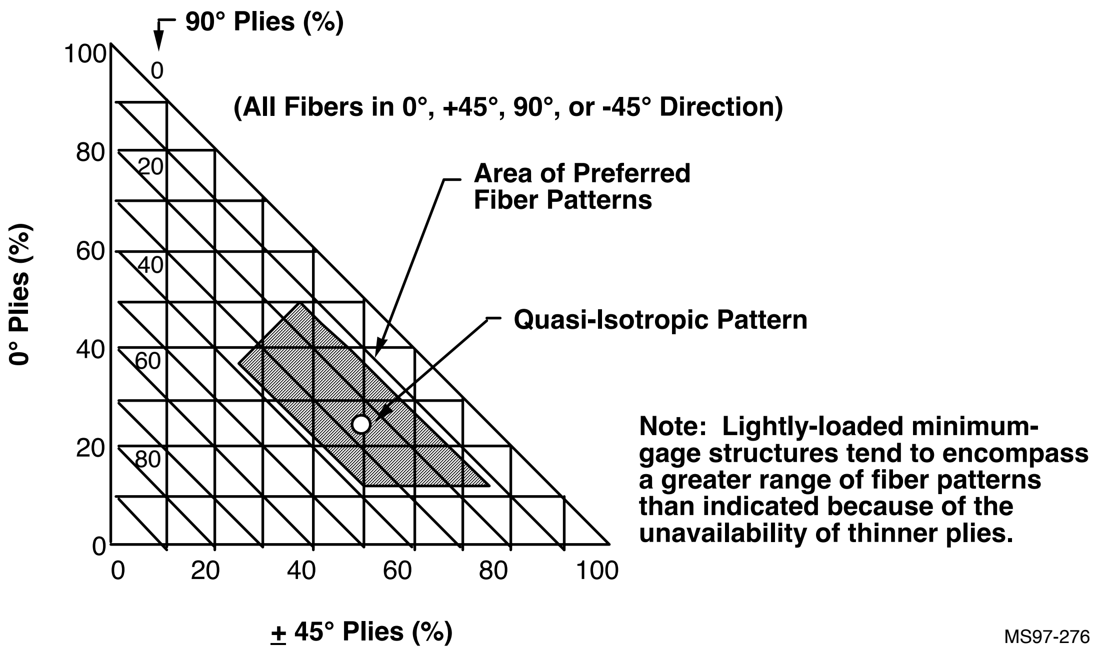


FIGURE 12-2. DESIRABLE LAY-UPS FOR BOLTED COMPOSITE JOINTS (FROM [30])

The problem of splitting of laminates with high percentage of 0° plies has also been observed during uniaxial tension tests of panels with 56% 0° plies and saw cuts normal the 0° axis [Ref. 33]. In these tests, highly orthotropic laminates were used as potential tear straps. Those laminates exhibited splitting after damage had propagated into the tear straps.

12.2 BONDED JOINTS

Numerous works authored by Hart-Smith, et al., provide guidelines, design advice, and experimental data for many bolted and bonded joint designs [Refs. 34-36]. The works should be consulted for a thorough understanding of good bonded joint design.

A few guidelines to bonded joint design are listed below. All of these guidelines are explained and substantiated in Hart-Smith's numerous reports and papers.

Guideline 21 Balance the Membrane Stiffness of the Adherends.

In this context "balanced" means the membrane stiffness of the adherends is the same on both sides of the joint. Significantly unbalanced designs suffer major strength loss relative to their balanced counterparts.

Guideline 22 Minimize Peel Stresses in Thicker Joints by Tapering Their Ends.

Most polymer matrix composite laminates cannot carry significant peel stress; hence, design of joints in such laminates must address this weakness. Peel stress is the interlaminar tension stress, that arises due to the moment generated near the end of a bonded joint. This moment balances the moment arising due to the eccentricity of one adherend middle surface with respect to the other. Tapering the ends is the most frequently used design solution to the problem of high peel stresses in bonded joints. If necessary, peel-resisting fasteners may be installed near the ends as well.

Guideline 23 Beware of Bonding Laminates With Significantly Different CTEs, Especially When Using High Temperature Cure Adhesives.

Residual strains present in the joint are proportional to the difference in CTEs of the adherends and cure temperature. As an example, attempts to bond aluminum members to graphite laminates can result in high residual stresses sufficient to fail the joint during autoclave cool-down. The higher the cure temperature of the adhesive, the more brittle, and hence the more susceptible to failure due to high residual strains it is likely to be.

Guideline 24 Use Step-Lap or Scarfed Joints in Highly Loaded Joints.

This guideline focuses on minimizing peel stresses for all but the thinnest laminates. This requires stepped lap or scarfed joints for any adherend greater than 0.1-inch thick. For composite-to-metal joints, step laps with the composites forming the outer members are preferred since machining steps interior to all but the thickest metal plate is difficult while the composite can be cocured to a metal part pre-machined with external steps. For composite-to-composite joints, scarf joints are likely to be preferred.

Guideline 25 Use the Most Ductile Adhesive That Satisfies Environmental Requirements.

Adhesive ductility is a singular virtue in joint design. Brittle adhesives lead to low joint strength and greater sensitivity to minor design details and tolerances.

Guideline 26 Use Adhesive Design Data Obtained From Thick Adherends.

Accurate adhesive material property data must be derived from tests with thick adherends (typically 3/8 in). The ASTM-D-1008 test, with its 0.063-inch thick adherends, does not provide meaningful adhesive structural data [Ref. 48].

Guideline 27 Lay-Up Outer Plies in Contact With the Adhesive at 0° or ±45° to the Principal Load Direction.

The outer surface plies in contact with the adhesive should be at 0° or ±45°, where 0° is the direction of the principal load. Having a ply at 90° to the principal load direction at the interface weakens the joint considerably since this ply has much lower transverse shear and in-plane strength than the adhesive but is just as critical for joint strength.

Guideline 28 Cocure Step Lap Joints.

This is strongly advised to simplify manufacturing. Machining a cured laminate to a close tolerance fit over a stepped member presents potentially serious tolerance problems.

Guideline 29 Ensure Adhesive and Laminate Curing Cycles Are Compatible.

In cobonded joints both laminate and adhesive are subjected to the same cure cycle. Hence, this cycle must result in a complete cure for both of them. In secondarily-bonded structures, the adhesive cure cycle must not degrade the properties of the precured laminates. For example, should the adhesive require a cure temperature near the T_g of the composite, that adhesive is likely to be unacceptable.

Guideline 30 Design Joints That Are Repairable.

This often means providing adequate space and edge distance to install mechanical fasteners, if the joint is severely damaged in service [Ref. 37].

Guideline 31 Correct Surface Preparation of the Adherends Is Essential.

Light abrasion of bonded surfaces is necessary. Tool release agents and removed peel plies can contaminate the surfaces to be bonded resulting in reduced joint strength.

Guideline 32 Provide a Corrosion Barrier Between Graphite Fiber Laminate and Aluminum Adherends.

In bonding laminates containing graphite fibers to aluminum, the galvanic corrosion potential is severe unless the adhesive is embedded in a glass fabric scrim cloth that acts as a barrier between the aluminum and composite surfaces.

SECTION 13

TAPERING OF SKINS AND FLANGES BONDED TO SKINS

Guideline 33 Changes in Laminate Thickness Normal to Primary Loading Directions Will Occur at a Taper Ratio of at Least 20:1. Thickness Changes Normal to Secondary Loading Directions Will Occur at a Taper Ratio of at Least 10:1.

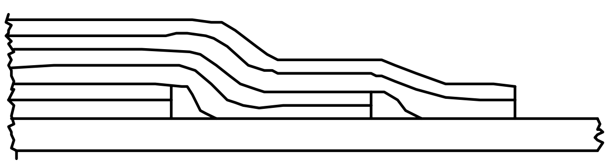
Steps in thickness exceeding approximately 0.02 inches introduce bending stiffness discontinuities and eccentricities that have been observed to cause interlaminar failures of parts in operational service. This observation has resulted in the specification of design guidelines for tapering zones where plies are dropped off. These guidelines are often stated in terms of number of plies that may be dropped over a given horizontal distance. For example, if the ply thickness is 0.005 inches, a region with a taper of 20:1 has no more than four plies dropped off every 20 x 0.02 = 0.4 inches.

Guideline 34 Angle Ply Pairs Should Be Dropped Off Together.

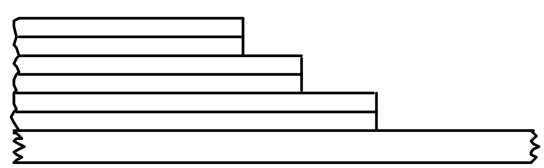
This guideline prevents the laminate from becoming locally unbalanced.

Guideline 35 The Outer Plies Should Cover All the Other Drop-Offs.

This is done as shown in Figure 13-1a to prevent edge delaminations that are more likely to occur at the drop-off shown in Figure 13-1b.



a. Outer Plies Protecting Others



b. Plies Liable to Delamination

MS97-272

FIGURE 13-1. PLY DROP-OFF DESIGN

Guideline 36 Stiffeners and Beam Flange Edges Should Terminate in a Minimum 10:1 Taper.

Untapered flanges create a discontinuity in lateral bending stiffness. The resulting eccentricity produces a moment from any lateral loading and a resulting tendency for the skin to peel away from the flange. A 10:1 taper is specified since such flanges are typically parallel to the primary load direction and normal to a secondary load direction. Such discontinuities also make the skin more vulnerable to impact damage. A flange termination is very vulnerable to failure by skin separation after an impact. Post-impact strength is improved by tapering the flange with the outer ply covering the inner ply drops as shown in Figure 13-1a.

SECTION 14

DAMAGE TOLERANCE, DURABILITY, AND CERTIFICATION

Certification of aircraft built of composite structures is a complex topic. Much of what was presented in Section 12 is relevant to the discussion of this topic. Real commercial or military aircraft structures must operate satisfactorily when damaged. If the damage is severe, the damage tolerance requirement may mean only completion of the flight in which the damage occurred without injury to passengers and crew. Required structural capability as a function of damage extent varies from one category of aircraft to another and must be negotiated between the original equipment manufacturer (OEM) and the certifying agency.

Damage tolerance is the ability of the structure to resist catastrophic failure in the presence of cracks or other damage without being repaired, for a specified number of operations (flights) or length of time in service. Damage tolerance is usually demonstrated by residual strength tests conducted after cyclic loading of a component that has been damaged in a well defined manner. Residual strength must be greater than limit load by a factor, defined by the certifying authority, that depends on the ability to detect the damage during an inspection.

Durability is the ability of the structure to resist damage initiation and/or growth for a specified length of time. How durable the structure should be designed to be is an economic issue. A highly durable structure requires fewer inspections and repairs. Durability is often demonstrated by showing that a specified damage does not grow significantly, if at all, during one or more spectrum fatigue tests.

Certification of composite components is inherently different from metals because of the large difference in behavior of composites and metals. Well-designed, fiber-dominated composites have excellent fatigue lives compared to metallic structures. However, their sensitivity to environments, impact damage and out-of-plane loading demand special attention. An added complication is that what are alleged to be “all-composite” structures almost always contain metal fittings whose fatigue lives require interrogation. Hence, aircraft certification via separate treatment of metal and composite components is seldom justified in practice. Such subtleties are extremely complex and deserve a separate review. Useful background information can be obtained from References 21, 29, and 38-47.

The following is a brief review of the fundamentals of how durability and damage tolerance certification of aircraft is addressed. Three levels of damage are considered: severe damage, clearly visible damage, and barely visible damage.

Severe Damage. This includes engine rotor burst damage in commercial aircraft and battle damage in military aircraft. Provision for adequate tolerance of severe damage must be considered in the conceptual design phase. The load level to be carried when such damage is present must be defined as well as the damage extent. Basic structural features such as spar, rib, frame, and stiffener configuration and location must be designed to satisfy severe damage tolerance requirements. Experimental verification of the required level of tolerance of severe damage requires large structural components and is inherently expensive. In transport aircraft, the ability to carry a specified load (some percent of design limit load (DLL)) with <u>at least</u> one major structural member severed in the wing and fuselage must be demonstrated. Depot level repairs are typically required after a severe damage event.

Clearly Visible Damage. In composite elements, this is typically induced by moderate impacts of sufficient energy to require structural repairs. Such damage may be caused by minor battle damage, large bird impact during flight, service vehicles, and dropped tool boxes. The damage is addressed analytically by considering <u>at most</u> one major structural element severed. Experimental validation of damage tolerance of structures containing clearly visible damage can often be accomplished using subcomponents (large panels) rather than complete aircraft components. Aircraft must be designed to carry limit load with clearly visible damage. While depot level repairs are not typically necessary after a clearly visible damage event, field level repairs may be necessary.

Barely Visible Damage. The likely presence of barely visible damage requires composite laminates to be designed to account for the presence of such damage by basing their allowable strengths on test coupons containing 1/4-inch diameter holes. This design approach is discussed in Section 12. Aircraft with barely visible damage are designed to carry ultimate load. Since the extent of barely visible damage is typically small, tests specimens used to validate barely visible damage tolerance can be proportionally smaller than those specimens needed to validate clearly visible or severe damage tolerance. Minor repairs are required should the damage be located; however, typically the airplane is completely airworthy even if this damage is left unrepaired.

Some of the fiber reinforced composite structures created for airframes in the 1960s and 1970s were developed with insufficient evaluation of their durability and damage tolerance. The certification requirements applied to airframes of that era were developed for metal structures, whose multi-decade durability and damage tolerance database and inherent ductility were well understood. Major durability issues in metals are initiation and growth of a through-thickness crack under tensile load. In composites, tension crack initiation and growth is often less important than propagation of impact-induced damage under compressive loading.

Guideline 37 Durability and Damage Tolerance Must Be Accounted for During All Stages of Design.

Today, laminated composite structure certification requirements and supporting databases are in place to ensure that durability and damage tolerance are considered throughout the development of these structures. The certification of composite structures is, because of the unique strengths and weaknesses characteristic of composite materials, markedly different from that of metal structures. Certification requirements for USN, FAA and USAF are, for sound reasons, different in some respects. These differences are discussed in References 21, 29, and 37-44. Structural design concepts must be chosen with full consideration of damage tolerance and durability requirements.

Thin laminated composite structures have been shown to be extremely susceptible to impact damage. Composite strength after impact (CSAI) is a critical design property. The life cycle costs (LCC) icrement due to inspecting for and repairing impact damage has been vastly greater than originally predicted for many components. To drive down these costs in new airframes, it is now common practice to require laminates to be immune to this type of damage for impacts below a specified energy level. Precluding the use of extremely fragile laminates is the intent of Guideline 38.

Guideline 38 Laminates Will Be at Least Thick Enough to Withstand Minor Impacts Without Damage.

Designers of production aircraft often invoke a minimum skin thickness of ~0.03 inches to ensure adequate durability and avoid the need for frequent repair. This guideline is sometimes phrased in terms of the requirement that no visible damage be produced by a specified low level impact, such as 4 ft-lb.

Many graphite fiber composite structures built before the mid-1970s were developed with relatively scant attention to their repairability. Most of the focus was on minimizing the weight of undamaged components and the fabrication of these components. The unfortunate result of this narrow focus was that the components proved to be more damage prone and their repair consumed far more resources than planned. This helped slow their acceptance in applications where life cycle cost was a serious consideration. To overcome this handicap, Guideline 39 is proposed.

Guideline 39 Design for Repairability.

Bonded repairs are usually restricted to thinner laminates [Ref. 37]. For thicker laminates, bolted repairs are often the preferred option. Many of the guidelines addressing laminate design are at least partially driven by the demands of cost-effective repair. For example, satisfying the requirement of allowing for bolted repairs anywhere in the structure is one motivation behind the user of the fiber-dominated design philosophy.

SECTION 15

RELEVANCE OF COMPOSITE DESIGN GUIDELINES TO SPAR CAPS IN UNINHABITED AERIAL VEHICLES MADE ENTIRELY OF 0° PLIES

The design guidelines presented in the previous sections are generic "rules of thumb" to be applied to the design of manned vehicles. Hence, the specific application of these guidelines to the design of lightweight spar caps for unmanned vehicles requires some evaluation. If spar caps having all plies aligned at a single orientation are to be used, the following ramifications must be considered.

1. The principal loading must be aligned along the fiber axis. Transverse strengths will be small. Hence, chordwise skin load transfer into the caps must be kept small.

2. No cutouts, sudden changes of section, or design details involving significant amounts of load transfer through shear can be tolerated.

3. It must be assumed that no mechanical fasteners will be used in these caps, either to attach skins or carry out repairs. The bolted joint strength will be small and splitting/cleavage failures are highly likely even at very low loads.

4. During cool-down of the cure cycle or during operations that may involve temperatures as low as -125°F, excessive matrix microcracking must be precluded. Tough, low temperature cure resin matrices are preferred. Close attention must be paid to the restraint of the structure to which the spar cap is attached to minimize the development of potentially harmful thermal stresses.

5. The damage tolerance of these spar caps will be low. Hence, protection against impacts on these spar caps will be required. Repairs, if at all possible, must be effected by bonding since bolt bearing strengths will be extremely low.

6. Attachment of the spar web warrants special care and attention. The attachment should be bonded or cocured. Interlaminar stresses developed due to shear load transfer from cap to web should be considered. A wrap-around configuration of web plies encasing the cap plies is recommended.

7. There may need to be constraints on spar cap cross section. Thin "T" sections are fragile and vulnerable to impact damage during handling and assembly. Low aspect ratio rectangles (approaching squares) are tougher.

These guidelines were never intended to apply to spar caps of unmanned aerial vehicles made entirely of 0° plies since the guidelines were created to aid the design of manned aircraft.

However, by understanding the background behind the design guidelines, the risks inherent in the use of 100% 0° ply spar caps can be understood and the need for more careful, detailed analyses can be evaluated.

A summary of composite guidelines as they relate to the design of spar caps made entirely of 0° plies is shown in Figure 15-2.

No.	Guideline	Rationale	Applicability to UAV Spar Caps	Reference
1, 6	Lay-up Symmetric About Midsurface	Uncouple bending and membrane response; maximize stiffness and buckling loads.	N/A	1, 3
2	Balance Laminates	Uncouple direct and shear responses; avoid thermal distortion.	N/A	1, 3
3	Do Not Extrapolate Test Data	Laminate properties can be highly nonlinear with respect to key parameters.	Yes	
4	Fiber-Dominated Laminates with ≥10% Fibers in at Least Three Directions	Linear response: low CTE to avoid microcracks, elevate crack resistance; provide excellent damage tolerance and durability.	N/A; near zero durability, sensitivity, and small transverse loads likely.	1, 3
5	Keep Primarily Loaded Plies Internal	Protect them from impact damage.	N/A; possibly low damage tolerance.	1, 46
7	±45° Plies on Exterior	Increase buckling loads; increased damage tolerance.	N/A	1
8	CTE Must Be Considered in Design/Analysis	Large differences in fiber and resin matrix CTE.	Yes	
9	≥10% 0° and 90° Plies	Avoid excessive CTE.	N/A	
10	Use Most Ductile Resin Matrix Satisfying Environmental Conditions	Increase joint strength; reduce sensitivity to manufacturing and assembly tolerances.	Yes	

FIGURE 15-21. DESIGN GUIDELINES RELATED SPAR CAPS MADE ENTIRELY OF 0° PLIES

No.	Guideline	Rationale	Applicability to 100% Spar Caps	Reference
11	Max Operating Temp ≤50°F Below T_g	Avoid intolerable drop in high temp compression and shear strength.	Yes	21, 22
12	Edge Stress Controlled in Design/Analysis	Avoid edge delaminations.	N/A	1, 23-28
13	Grouping of Plies at Same Angle ≤0.02" Thick (Four 0.005" Plies)	Avoid microcracking and edge delaminations.	N/A	1
14	Poisson's Ratio Mismatch Between Skin and Bonded Stiffener <0.1	Avoid bond failures.	Yes; skin must be bonded or cocured to spar.	1
15	Laminates are Designed Assuming 1/4" Dia Hole Anywhere in Them	Damage tolerance, durability and bolted repairs.	N/A, but must accept low damage tolerance.	
16	<60% 0° Plies at Cutouts and Bolted Joints	Avoid splitting and sensitivity to lateral loads.	Yes; cap must be free of notches.	30, 31
20	Guideline 21 and ≥35% ±45° Plies at Bolted Joints	Ensure acceptable joint behavior.	N/A	30, 31
21	Do Not Bond Laminates with Significantly Different CTEs	Avoid adhesive failures.	Yes, if skins bonded to spars.	
22	Design Joints as Stiffness Balanced as Feasible	Enhance joint strength.	Yes, if skins bonded to spars.	34, 35
23	Taper Ends of Thick Joints	Minimize peel stresses.	Yes, if spar cap has bonded joints.	34, 35
24	Scarf/Step Thick Joints	Enhance strength.	Yes, if spar cap has bonded joints.	34, 35
25	Cocure Step Lap Joints	Reduce manufacturing complexity and cost.	Yes, if spar cap has bonded joints.	34, 35
33	Taper Drop-Offs	Minimize stress concentration at ends.	Yes, if spar caps change thickness.	1
34	±45° Plies are Dropped Off Together	Ensure balanced laminate.	N/A	1
35	Outer Plies Should Cover Interior Dropped Plies	Prevent edge delamination.	Yes, if spar caps change thickness.	1
36	Stiffener Flanges Tapered at ≥10:1	Minimize bending discontinuity.	N/A	1

FIGURE 15-21. DESIGN GUIDELINES RELATED SPAR CAPS MADE ENTIRELY OF 0° PLIES (CONT'D)

No.	Guideline	Rationale	Applicability to 100% Spar Caps	Reference
37	Carry Specified Load After Impact	Satisfy damage tolerance requirements.	N/A; if very low damage tolerance acceptable.	1, 38-45
38	Satisfy Minimum Gage Requirement	Satisfy durability requirement.	N/A	1, 38-45
39	Ensure Repairs Feasible	Ensure satisfactory durability.	Yes	36

FIGURE 15-21. DESIGN GUIDELINES RELATED SPAR CAPS MADE ENTIRELY OF 0° PLIES (CONT'D)

SECTION 16

CLOTH (FABRIC) PLIES

Most of the guidelines in the previous sections were developed based on experience working with unidirectional tape composites. A wide range of woven fabrics are available and frequently used. They range from fabrics having similar structural properties in the orthogonal warp and weave directions to special weaves whose structural properties differ little from tape. The focus here is on the former category where $E_{11} \approx E_{22}$, $\upsilon_{12} \approx \upsilon_{21}$, $\alpha_{11} \approx \alpha_{22}$, since cloth plies having these properties are used most frequently in real structural applications.

Guidelines for laminates of cloth plies that relate to edge effects can be relaxed (though not ignored), due to the smaller differences in CTE and Poisson's ratios in the ply primary directions. Furthermore, because each ply inherently provides fibers in orthogonal directions, laying plies in only two directions automatically results in a fiber-dominated laminate with fibers in four directions. When tapering a laminate, it is still good practice to drop off no more than 0.014 inch of cloth thickness every 0.1 inch. This means that typical graphite-epoxy cloth having 0.014-inch thick plies should be dropped off singly.

There are other cases where most of the laminate is tape, but a few cloth plies are placed on its inner and outer surfaces to provide additional durability and damage tolerance. Guidelines for laminates made of unidirectional tape should be applied to these hybrid laminates.

SECTION 17

SUMMARY

A review of typical guidelines used in the aircraft industry for design of laminated composite structures on production aircraft has been conducted to document the reasons for their specification. Few, if any, of these guidelines are supported by rigorous analyses and experiments published in the generally available literature. Some were generated roughly 30 years ago, to apply to the designs, materials and fabrication methods of that era. Much of the early work that gave rise to these guidelines occurred during development of military aircraft primary and secondary structure. Hence, some information supporting these guidelines remains classified. Many guidelines are also often presented, without substantiation, in proprietary corporate design guides.

The design guidelines have been developed to maximize the benefits of using composite materials while minimizing the effects of their weaknesses. The need for tolerance of structural damage typically encountered during an aircraft's life was another key motivation behind development of these guidelines. Ease of manufacture and assembly also plays a big role in guideline development.

SECTION 18

REFERENCES

1. Niu, M. C. Y., Composite Airframe Structures, Conmilit Press 1993.

2. Paul, P. C., Saff, C. R., Sanger, K. B., Mahler, M. A., and Kan, H., "Out of Plane Analysis for Composite Structures," *Proc 8th DOD/NASA/FAA Conference on Fibrous Composites in Structural Design*, NASA CP 3087, 1989.

3. Jones, R. M., Mechanics of Composite Materials, McGraw Hill, 1975.

4. Anonymous, "Advanced Composites Design Guide," AFML, 1973.

5. Austin, F., et al., "Aeroelastic Tailoring of Advanced Composite Lifting Surfaces in Preliminary Design," *Proc 17th SDM Conference*, Valley Forge, PA, May 1976, pp. 69-79.

6. Lynch, R. W. and Rogers, W. A., "Aeroelastic Tailoring of Composite Materials to Improve Performance," *Proc 17th SDM*, May 1976, pp. 61-68.

7. Krone, N. J., "Divergence Elimination With Advanced Composites," AIAA Paper 75-1009, 1975.

8. Weishaar, T. A., "Divergence of Forward Swept Wings," J. Aircraft, Vol. 17, No. 6, June 1980, pp. 442-448.

9. Dastin, S., Eidenoff, H. L., and Armen, H., "Some Engineering Aspects of the X-29 Airplane," *Proc 29th National SAMPE Symposium*, 3-5 April 1984, pp. 1438-1449.

10. Hadcock, R. N., "X-29 Composite Wing", *Proc. AIAA "Evaluation of Aircraft/Aerospace Structure and Materials Symposium*," Wright-Patterson AFB, OH, 24-25 April 1985.

11. Eckstrom, C. V. and Spain, C. V., "Experiences in the Use of Composite Material for a Wing Skin," *J. Aircraft*, Vol. 20, No. 11, November 1983, pp. 913-919.

12. Leissa, A. W., "Buckling of Laminated Composite Plates and Shell Panels," AFWAL-TR-3069, 1985.

13. Fogg, L., "Stability Analysis of Laminated Materials," Lockheed Corporation Class Notes quoted as Reference 32 in [5].

14. Nemeth, M. P., "Buckling Behavior of Long Symmetrically Laminated Plates Subjected to Combined Loadings," NASA TP-3195, 1992.

15. Nemeth, M. P., “Importance of Anisotropy on Buckling of Compression-Loaded Symmetric Composite Plates,” *AIAA Journal*, Vol. 24, No. 11, Nov. 1986, pp. 1831-1835.

16. Nemeth, M. P., “Buckling Behavior of Long Symmetrically Laminated Plates Subjected to Shear and Linearly Varying Axial Edge Loads,” NASA TP 3659, 1997.

17. Jensen, D. W. and Lagace, P. A., “Influence of Mechanical Couplings on Buckling and Postbuckling of Anisotropic Plates,” *AIAA Journal*, Vol. 26, No. 10, October 1988, pp. 1269-1277.

18. Lagace, P. A., Jensen, D. W., and Finch, D. C., “Buckling of Unsymmetric Composite Laminates,” *Composite Structures*, Vol. 5, 1986, pp. 101-123.

19. Watson, J. C., “AV-8B Composite Fuselage Design,” *J. Aircraft*, Vol. 19, No. 3, March 1982. P 235.

20. Popov, E. V., Engineering Mechanics of Solids, Prentice Hall, 1990, p. 120.

21. Whitehead, R. S., “Lessons Learned for Composite Aircraft Structures Qualification,” *Proc. 1987 Aircraft/Engine (ASIP/ENSIP) Conference*, San Antonio, 1-3 December 1987.

22. Anonymous, “Polymer Matrix Composites,” MIL-HDBK-17-1D, February 1994, Section 2.2.1.

23. Pipes, R. B. and Pagano, N. J., “Interlaminar Stresses in Composite Laminates Under Uniform Axial Extension,” *J. Comp. Mat.*, October 1970, pp. 255-259.

24. Pagano, N. J. and Pipes, R. B., “The Influence of Stacking Sequence on Laminate Strength,” *J. Comp. Matls.*, Vol. 5, January 1971, p. 50.

25. Whitney, J. M., Daniel, I. M., and Pipes, R. B., Experimental Mechanics of Fiber Reinforced Composite Materials, Soc. Exp. Stress Analysis Monograph No. 4, 1982, p. 56 et seq.

26. Wang, A. S. D. and Crossman, F. W, “Initiation and Growth of Transverse Cracks and Edge Delaminations in Composite Laminates, Part 1: An Energy Method,” *J. Comp. Matls. Supplement*, Vol. 14, 1980, pp. 71-87.

27. Crossman, F. W, Warren, W. J., Wang, A. S. D., and Law, G. E., “Initiation and Growth of Transverse Cracks and Edge Delaminations in Composite Laminates, Part 2,” *J. Comp. Matls. Supplement*, Vol. 14, 1980. pp. 88-108.

28. Long, B. J. and Swanson, S. R., “Ranking of Laminates for Edge Delamination Resistance,” *Composites*, Vol. 25, No. 3, 1994, pp. 183-188.

29. McCarty, J. E. and Horton, R. E., "Damage Tolerance of Composites," paper presented at 15th Congress, International Council of Aeronautical Sciences, London, England, September 1986.

30. Hart-Smith, L. J., "Mechanically-Fastened Joints for Advanced Composites – Phenomenological Considerations and Simple Analyses," Fibrous Composites in Structural Design, Plenum Press 1980, p. 543.

31. Hart-Smith, L. J., "Design and Analysis of Bolted and Riveted Joints in Fibrous Composite Structures," *Proc. International Symposium on Joining and Repair of Fiber-Reinforced Plastics*, London, 10-11 September 1986.

32. Hart-Smith, L. J., "The Design of Efficient Bolted and Riveted Fibrous Composite Structures," Douglas Paper 8335, 1989.

33. Wolf, E. and Pasricha, A., "Sensitivity of Highly Orthotropic Laminated Composites to Saw Cut Damage," NASA/Boeing HSR program report performed under Boeing Purchase Contract ZA0851, March 1996.

34. Hart-Smith, L. J., "Design and Analysis of Adhesive-Bonded Joints," Douglas paper 6059, also published in *Proc Conference on Fibrous Composites in Flight Vehicle Design*, Dayton, OH, September 1972.

35. Hart-Smith, L. J., "Further Developments in the Design and Analysis of Adhesive-Bonded Structural Joints," *Proc ASTM Symposium on Joining of Composite Materials*, Minneapolis, MN, April 1980.

36. Hart Smith, L. J., "Designing to Minimize Peel Stresses in Adhesive-Bonded Joints, Delamination, and Disbonding of Materials," ASTM STP 876, W. S. Johnson, Ed., 1985, pp. 238-266.

37. Hart-Smith, L. J., "Design of Repairable Composite Structures", *SAE Trans 851830, SAE Aeropsace Technology Conference*, Society of Automotive Engineers, 1985.

38. McCarty, J. E. and Johnson, R. W., "Durable and Damage Tolerant Composite Commercial Aircraft Structure Design Approach," *J. Aircraft*, January 1978, pp. 33-39.

39. Horton, R. E. and McCarty, J. E., "Damage Tolerance of Composites," COMPOSITES, ASM, 1987.

40. Weinberger, R. A., Somoroff, A. R., and Riley, B. L., "US Navy Certification of Composite Wings for F-18 and Advanced Harrier Aircraft," AIAA paper #77-466, 1977.

41. Whitehead, R. S., "Certification of Primary Composite Aircraft Structures," *Proc 14th Symposium of International Committee on Aeronautical Fatigue*, Ottawa, 10-12 June 1987.

42. Soderquist, J. R., "Damage Tolerance Certification of Civil Composite Material Aircraft Structure," *Proc 6th Conf on Fibrous Composites in Structural Design*, AMMRC-MS-83-2, November 1983. pp. III-1 III-12.

43. Soderquist, J. R., "Design/Certification Considerations in Civil Composite Aircraft Structures," SAE paper 871846, presented at *Aerospace Technology Conference*, Long Beach, CA, 5-8 October 1987.

44. Lincoln, J. W., "Certification of Composites for Aircraft," *Proc USAF Aircraft Structural Integrity Conference*, Sacramento, CA, December 1986.

45. Kan, H., Whitehead, R. S., and Kautz, E., "Damage Tolerance Certification Methodology for Composite Structures," *Proc 8th DOD/NASA/FAA Conference on Fibrous Composites in Structural Design*, 28-30 November 1989.

46. Demuts, E. and Sharpe, P., "Tougher Advanced Composite Structures," *Proc 28th SDM Conference*, Monterey, CA, April 1987, pp 385-393.

47. Kan, H., Whitehead, R. S., and Kautz, E., "Damage Tolerance Certification Methodology for Composite Structures," *Proc 8th DOD/NASA/FAA Conference on Fibrous Composites in Structural Design*, Norfolk, VA, November 1989.

48. Anonymous, "Strength Properties of Adhesives in Shear by Tension Loading (Metal-to-Metal)," ASTM-D-1002, Vol 15.06, 1983, pp 60-64.