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RESIDUAL STRESSES IN ANGLEPLIED LAMINATES AND THEIR EFFECTS ON LAMINATE BEHAVIOR

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NASA Lewis Research Center research in the field of composite laminate residual stresses is reviewed and summarized. The origin of lamination residual stresses, evidence of their presence, experimental methods for measuring them, and theoretical methods for predicting them are described. Typical results are presented which show the magnitudes of residual stresses in various laminates including hybrids and superhybrids, and in other complex composite components. Results are also presented which show the effects of lamination residual stresses on laminate warpage and on laminate mechanical properties including fracture stresses. Finally, the major findings and conclusions derived therefrom are summarized.

STAR Category 24

Introduction

Determining the structural integrity and durability of structural components fabricated from angleplied fiber composite laminates requires use of various deterministic mathematical models. These models, for example, may be for relating stresses to applied forces, stress intensities at the tips of cracks to nominal stresses in the component, buckling resistance to applied force, vibration response to excitation sources, stress wave propagation to impact loads, etc. Deterministic models of such cases require knowledge of stress-strain-temperature-moisture relationships and strain dependent stress and failure criteria to relate stress to strength. Experimental data, for example, indicate that the presence of residual stress has significant influence on the initial tangent properties of boron/aluminum angleplied laminates. Analysis of experimental data from fiber/resin-matrix laminates shows that the lamination residual stresses may be of sufficiently high magnitude to strain the plies transverse to the fiber direction near or beyond the corresponding strength in certain ply orientation configurations. Determination of residual stresses and their effects on the laminate structural integrity has been the subject of considerable in-house and sponsored research at NASA Lewis Research Center (LeRC) over the past 8 years. The important results generated during this period and the significant conclusions derived therefrom are reviewed and summarized herein.

Origin of Lamination Residual Stresses

Lamination residual stresses in the plies of angleplied fiber composite laminates (fig. 1) originate from the four following factors:

1. The differences in the ply thermal expansion coefficients along ($\alpha_{\ell 11}$) and transverse ($\alpha_{\ell 22}$) to the fiber direction (fig. 2). For fiber/resin composites the ratio of these coefficients ($\alpha_{\ell 22}/\alpha_{\ell 11}$) is about 20 while for boron/aluminum composites it is about 3.

2. The difference in the ply orientation angle (θ_{ℓ}) between the ply material axis (along the fiber direction (1)) and the laminate structural axis (X), Figure 1. The larger this angle, the greater the lamination residual stress in the ply. There are no residual stresses in unidirectional composites because all plies have the same orientation angle. However, they are present in unidirectional hybrids.

3. The difference (ΔT) between the temperature at which the laminate is fabricated and the temperature at which the laminate is used at the instant the lamination residual stresses are calculated. For fiber/epoxy composites the ΔT is approximately 167 K (300° F); for fiber/polyimide composites, about 295 K (530° F); and for boron aluminum composites, about 500 K (900° F).

4. The ability of the ply to support stress along its material axis. That is, if a ply cannot support stress along the fiber direction ($\sigma_{\ell 11}$), there will be no lamination residual stress along this direction, and similarly for the stress transverse to the fiber direction ($\sigma_{\ell 22}$) or the intralaminar shear stress ($\sigma_{\ell 12}$).

Evidence of Lamination Residual Stresses

Visual evidence of lamination residual stress in angle ply laminates exists in the form of transply cracks and warpage of unsymmetric angleplied laminates. Both of these effects occur after the laminate has been fabricated and before it is subjected to any mechanical loading.

Transply cracks occur when the lamination residual stress in the transverse direction exceeds the ply transverse tensile strength ($S_{\ell 22T}$) or fracture strain. For typical fiber/resin composites $S_{\ell 22T}$ has a range of 28 to 69 MPa (4 to 10 ksi) while for boron/aluminum composites its range is 82.7 to 165.5 KPa (12 to 24 ksi). The photomicrograph in Figure 3 taken from Ref. 1 shows transply cracks in a $[(0/90)_2]_S$ (0/90/0/90/90/0/90/0 - stacking sequence) high-modulus-graphite-fiber/epoxy angleplied laminate. The transply cracks shown occur in both the 0^o and the 90^o-plies and, as can be seen, they occur at regular intervals.

Laminates warp after they are removed from the mold if their ply configuration is unsymmetric. This is illustrated in Figure 4, which shows the warpage of [0/45/-45/0] unsymmetric laminate.

Measurement of Lamination Residual Stresses

The magnitude of lamination residual stress in angleplied laminates may be determined directly by embedding strain gages as described in Refs. 2 and 3, or indirectly by measuring the corner displacement (ref. 4) or the curvature of unsymmetric warped laminates.

Strain gages are embedded during the fabrication of the laminate. The gages are usually placed as is shown in Figure 5. The strain variation in the plies during the fabrication process is determined by monitoring the embedded strain gages. Typical strain versus temperature records during fabrication are shown in Figure 6 taken from Ref. 3. The stresses corresponding to these strains are determined by using appropriate stress-strain-temperature relationships as will be described later. It is worthy of note that practically all the lamination residual strain is thermal and builds up as the laminate is cooled from the cure temperature 440 K (350^o F) to room temperature.

The lamination residual strains and stresses of warped laminates can be determined from the measured curvatures as will be described later. Note that it is possible to determine the variation of curvatures versus temperatures by heating a warped laminate.

Equations for Predicting Lamination Residual Stresses

Equations for predicting approximately ply lamination residual stresses ($\sigma_{\ell i}$), or ply thermal stresses in general, are derived using linear laminate theory as described in Ref. 5. These equations, in matrix form, are given by

$$\{\sigma_{\ell i}\} = [E_{\ell i}]^{-1} \left\langle [R_{\ell i}]\{\varepsilon_{cox}\} - z_{\ell i}[R_{\ell i}]\{\kappa_{cx}\} - \Delta T_{\ell i}\{\alpha_{\ell i}\} \right\rangle \tag{1}$$

where the reference plane strains $\{\varepsilon_{cox}\}$ and the curvature changes $\{\kappa_{cx}\}$ for a free composite (free of external loads and boundary constraint) are computed from

$$\begin{Bmatrix} \{\varepsilon_{cox}\} \\ \{\kappa_{cx}\} \end{Bmatrix} = \begin{bmatrix} [A_{cx}][C_{cx}] \\ [C_{cx}][D_{cx}] \end{bmatrix}^{-1} \begin{Bmatrix} \{N_{c\Delta Tx}\} \\ \{M_{c\Delta Tx}\} \end{Bmatrix} \tag{2}$$

Equations (1) and (2) show that the ply residual stress is a function of the following factors: (1) the composite or laminate structural stiffnesses,

$[A_{cx}]$, $[C_{cx}]$ and $[D_{cx}]$, and the thermal forces $\{N_{c\Delta Tx}\}$ and $\{M_{c\Delta Tx}\}$, (2) the ply spatial location in the composite $Z_{\ell i}$ and $[R_{\ell i}]$, (3) the ply stress-strain relation $[E_{\ell i}]$, (4) the ply thermal coefficients of expansion $\{\alpha_{\ell i}\}$ and, (5) the temperature difference between the ply and the reference value $\Delta T_{\ell i}$. This difference equals the ply temperature minus the cure temperature in computing residual and thermal stresses. The above discussion leads to the identification of the independent variables which influence the ply residual stress. These are: (1) Constituent materials' elastic properties, (2) Constituent materials' thermal properties, (3) Fiber volume ratio, (4) Void content, (5) Ply distance from reference plane, (6) Ply orientation relative to composite structural axes and (7) Difference between ply temperature and cure temperature. In subsequent sections the effects of some of these variables on ply lamination residual stresses will be illustrated and discussed.

The matrix equations (1) can also be used to predict ply lamination residual stress in warped laminates from measured curvatures. For this case the reference plane strains $\{\varepsilon_{cox}\}$ are also needed. These strains may be determined directly by measurement or they may be approximated using the following equation:

$$\{\varepsilon_{cox}\} = \Delta T \{\alpha_{cx}\} \tag{3}$$

where ΔT is the difference between cure and room temperatures and $\{\alpha_{cx}\}$ are the thermal expansion coefficients of the laminate along the structural axes. The thermal expansion coefficients are determined using linear laminate theory (ref. 6).

The ply thermal stresses versus temperature can be determined using measured curvatures and either measured or predicted via equation (3) reference plane strains. For this case both the curvatures and the reference plane strain need to be known at the temperature of interest.

Lamination Residual Stresses in Typical Angleplied Laminates

Lamination residual stresses in the plies of typical angleplied laminates used for structural components were computed using equations (1) and (2) as described in Ref. 5. Some typical results obtained from these computations are illustrated in Figure 7 for ply transverse residual stresses.

The points to be noted from Figure 7 are:

1. The ply residual transverse stress ($\sigma_{\ell 22}$) in $[0_2/\pm\theta]_S$ angleplied laminates increases rapidly with θ up to about 45° and levels off thereafter (fig. 7(a)). This figure also shows that $\sigma_{\ell 22}$ decreases with increasing fiber volume ratio in the range shown (0.35 to 0.75).

2. The ply residual stress $\sigma_{\ell 22}$ is not sensitive to θ in $[0/\pm\theta/90]_S$ angleplied laminates (fig. 7(b)).

3. The ply residual transverse stress is approximately the same in both symmetric $[(\pm\theta)_2]_S$ and unsymmetric interspersed $[(\pm\theta)_4]$ angleplied laminates (fig. 7(c)). The corresponding stress in unsymmetric non-interspersed $[\theta_4/-\theta_4]$ angleplied laminates is about 30 percent higher than that in either the symmetric or the unsymmetric interspersed laminates.

The magnitude reached by the ply residual transverse stress in the symmetric laminates with 0.55 fiber volume ratio is about 41 MPa (6 ksi) which is well within the range of the corresponding strength 28 to 70 MPa (4 to 10 ksi) mentioned in a previous section. Based on these calculations the transply cracks shown in Figure 3 would be expected.

Lamination Residual Stresses in Special Angleplied Laminates

As is well known, the laminate configuration is selected to meet specific design requirements. For example, the two angleplied laminates $[(\pm 45)_4/(\pm 22.5)_2/0_6]_S$ and $[(\pm 30)_4/(\pm 15)_2/0_6]_S$ are usually selected for structural components where cantilever flexural and torsional stiffness control the design. The first laminate offers greater torsional stiffness while the second offers greater flexural stiffness and somewhat greater flexural strength. Typical applications for such laminates may be skins on airplane wings and/or compressor fan blades for aircraft turbine engines.

The ply residual transverse stresses in the above angleplied laminates are shown in Figure 8 taken from Ref. 5 as a function of matrix modulus. As can be observed the ply residual transverse stress in the $[(\pm 45)_4/(22.5)/0_6]_S$ angleplied laminate is about twice as high as that in the $[(\pm 30)_4/(\pm 15)/0_6]_S$ angleplied laminates. Also, the ply residual transverse stress increases with increasing matrix modulus (E_m). This increase is more rapid where E_m is smaller than $4.8\text{x}10^6$ kPa ($0.7\text{x}10^6$ psi) and appears to level off where E_m is greater than this value. As can also be observed, the residual transverse stress in the first laminate is about 62.0 MPa (9 ksi) for an E_m of about $3.4\text{x}10^6$ kPa ($0.5\text{x}10^6$ psi). This is greater by about 50 percent than the corresponding ply transverse strength of about 41 MPa (6 ksi) (ref. 5) for this composite.

The following points may be concluded from the results shown in Figure 8 and the above discussion:

1. The lamination residual stress is an important design variable and needs to be considered in selecting laminate configurations for specific designs.

2. The ply lamination residual stresses in special angleplied laminates may reach magnitudes which exceed the strength of the ply and therefore produce transply cracks in these laminates.

3. The lamination residual stress depends strongly on the matrix modulus (E_m) where E_m is less than about $6.9\text{x}10^6$ kPa ($0.7\text{x}10^6$ psi). This includes almost all resin matrices. Composites made from a matrix with low modulus will have relatively low lamination residual stresses and thermal stresses in general. However, the strengths and moduli of these composites tend to be low also (ref. 5).

4. The lamination residual stress in the composite decreases as the matrix modulus decreases. The modulus can decrease either due to mechanical load, thermal or any other environmental exposure, or combinations thereof. Composite strengths and moduli decrease when this reduction in matrix occurs. The decay in residual stress with loss in modulus is consistent with the statement made earlier that residual stress exists only if the material can support it.

Though data are not presented here, it was shown in Ref. 5 that the lamination residual stress does not depend on void content in the composite and decreases with decreasing matrix thermal expansion coefficient. These two effects, predicted in Ref. 5, were confirmed experimentally in Ref. 3. The residual stress decreases with increasing fiber volume ratio (fig. 7(a)).

The residual stress increases with increasing ply orientation angle (fig. 7(a)) and it increases with increasing matrix modulus (fig. 8). These observations were also confirmed by the data of Ref. 3. Lamination residual stresses in various pseudo-isotropic (quasi-isotropic) laminates where investigated in Ref. 7. The ply residual transverse stress in these laminates has a magnitude comparable to that shown in Figure 7(b). Lamination residual stresses in high modulus graphite/polyimide angleplied laminates suitable for aircraft turbine engine fan blade use were investigated in Ref. 8. The results showed that the ply transverse residual stress can be minimized at the expense of torsional stiffness. Both predictions developed at LeRC and experimental data shown in Ref. 9, neither of which are reported herein, showed that the lamination residual stresses in the plies of interply hybrids (ply-by-ply staking) are approximately equal to those in the non-hybrids provided that they both have the same laminate configuration.

Lamination Residual Stresses in Superhybrid Composites

Superhybrid composites consist of adhesively bonded and strategically located titanium, boron/aluminum and graphite/epoxy plies. The fibers in both the boron/aluminum and the graphite/epoxy plies are oriented in the same direction. Superhybrids were developed recently (refs. 10 through 12) to meet diverse design requirements which require balanced mechanical properties such as: strength, stiffness, impact and erosion resistance, lightweight, reduced residual stresses, and resistance to hygrothermal environments.

Superhybrid composites are fabricated using the same procedures as for other advanced fiber composites. These procedures require an elevated temperature cure and pressure (ref. 10). When the superhybrid is cooled to room temperature lamination residual stresses buildup in all the plies. Those in the adhesive plies are of special interest because it is through these plies that the superhybrid developes the composite action.

Lamination residual stress in the plies of three different superhybrids are shown in Figure 9 taken from the data of Ref. 10. The laminate configuration in each superhybrid is shown in the photograph of its cross-section in the upper part of the figure. The lamination residual stresses are shown in the lower part. As can be observed in Figure 9 the margins of safety (MOS) values, based on the combined stress failure criterion (ref. 6), for the adhesive are greater than 0.66. This means that the lamination residual stresses in the adhesive are relatively low compared to its tensile strength. Another observation that can be made is that the transverse residual stress in the graphite/epoxy composite is about 21 MPa (3 ksi) which is about 50 percent of that in Figure 7(a) for 0.55 fiber volume ratio angleplied laminate.

The conclusion from the above discussion is that superhybrids can be configured which will have relatively low lamination residual stresses in the adhesive.

Lamination Residual Stresses in Complex Composite Components

Composite structural components may have complex shapes with curved surfaces and variable thickness. One such a component is a fan blade for aircraft turbine engines. Composite fan blades for possible high-tip-speed application were investigated at LeRC (ref. 13). A photograph of this blade is shown in Figure 10. The blade is made from high-tensile strength graphite fiber/PMR - polyimide (HTS/PMR) composite and consists of [±45/±20/0] generic symmetric laminate configuration. The airfoil shape of the blade has camber twist and tapered thickness from leading edge to trailing edge and from root to tip. A detailed description of the blade is given in Ref. 13.

Fiber reinforced polyimide composites are usually cured at 589 K (600° F). The temperature difference between this and room temperature is 294 K (530° F). This temperature difference produces high ply transverse residual stress which may produce transply cracks in the blade and may also produce interply delaminations. Transply cracks in graphite fiber/polyimide laminates having ply orientation angles greater than 30° are a common occurrence.

The lamination residual stresses in the plies of the high-tip-speed composite blade just described were computed in Ref. 14 using a temperature difference of 294 K (530° F). Maximum-stressed-ply residual stresses from Ref. 14 are summarized in Table I for 9-chord sections and 3-span stations. The corresponding ply (unidirectional composite) fracture stresses are given at the end of the table for comparison purposes. The residual stresses near the trailing edge are not given because the finite element analysis model was not sufficiently fine to represent the blade in this region (ref. 14).

It can be seen in Table I that the ply residual transverse stresses in the maximum stressed ply are about 41 MPa (6 ksi) and appear to be about uniform throughout the blade except at the trailing edge. A ply transverse stress of 41 MPa (6 ksi) is about 67 percent of the corresponding transverse fracture stress 62 MPa (9 ksi). The conclusion from the above discussion is that ply transverse residual stresses in complex components reach relatively high magnitudes compared to corresponding fracture stresses and may, therefore, cause transply cracks prior to actual application of, or at relatively low values of mechanical load. This conclusion is similar to those reached for the flat laminates.

Laminate Warpage Due to Lamination Residual Stresses

Flat composite laminates have been observed to warp upon removal from their fabrication mold. The warpage may be of a magnitude sufficient to render the laminate useless for its intended purpose. The warpage is the result of ply stacking sequences which are not symmetric and the lamination residual stresses which are induced by the temperature difference between the cure and room temperatures. Ply stacking sequences which are not symmetric with respect to bending result in laminates which exhibit coupled bending-stretching response when subjected to either thermal or mechanical loads. Warpage, then, is the result of bending which is coupled to the laminate thermal contraction (negative-stretching) induced by the temperature difference between cure and room temperatures. This temperature difference for typical composite systems is as follows: fiber/epoxy about 167 K (300° F), fiber/polyimide about 294 K (530° F), and for boron/aluminum about 500 K (900° F).

A theory for predicting laminate warpage due to lamination residual stresses is described in Ref. 4. This theory consists of approximate relationships to predict the warpage corner deflection, depicted in Figure 11, using the curvatures $\{\kappa_{cx}\}$ from equations (2). Predicted corner deflections of a warped laminate, assuming that one of the plies is misoriented, are shown in Figure 12 taken from Ref. 4. The results shown in Figure 12 were obtained by assuming that one of the 45-plies in the $[0_2/\pm 45]_s$ symmetric laminate was misoriented as follows:

$$[0_2/45\text{-}45/\text{-}45/(45 - \Delta\theta)/0_2]$$

where $\Delta\theta$ varied from 0 to 20 degrees. In addition, the laminate was assumed to be a 25.4 cm x 25.4 cm (10 in. x 10 in.) square plate by 0.15 cm (0.06 in.) thick and made from high-modulus graphite-fiber/epoxy (MOD-1/ERLA-4617) composite. The temperature reduction was taken to be equal to 167 K (300° F).

The key points to be noted from Figure 12 are:

1. The major part of the warpage corner deflection is due to bending.

2. A ply misorientation of as little as 0.5° produces a warpage corner deflection equal to the thickness of the laminate.

The reader is referred to Ref. 4 for comparison with experimental data as well as other factors which may contribute to laminate warpage.

Some of these are: nonuniform fiber and void content; unequal ply thickness; and nonuniform temperature and moisture distribution through the laminate thickness. The effect of nonuniform moisture on warpage is discussed in Ref. 15.

Effects of Lamination Residual Stresses on Laminate Mechanical Properties

The presence of lamination residual stresses in symmetric angleplied laminates may effect mechanical properties that is, how the laminate responds under mechanical load. The effects depend mainly on the following four variables:

1. Magnitude and sense of ply residual stress;

2. Sense of mechanical load;

3. Ply orientation; and

4. Composite system.

The total ply strain history from cure temperature to fracture under tensile load was measured in $[0_2/\pm45]_S$ angleplied laminates made from several composite systems (ref. 3). Typical 0-ply strain history in a boron/epoxy laminate is shown in Figure 13. The strain history plotted is as follows: (1) Fabrication temperature 450 K (350° F) to 297 K (75° F) with no load and (2) Tensile load stress from 0 to fracture, about 690 MPa (100 ksi), parallel to the 0-ply direction at constant temperature, 297 K (75° F). The points to be observed from Figure 13 are:

1. The ply strains vary linearly over the range of the fabrication temperature and with mechanical load.

2. The residual longitudinal strain is slightly compressive whereas the transverse strain is about 0.2 percent tensile.

3. The longitudinal strain due to mechanical load is tensile and increases linearly almost to fracture with increasing tensile stress.

4. The corresponding transverse strain is compressive and increases in magnitude linearly to fracture with increasing tensile stress.

5. The total longitudinal strain at fracture is about 0.6 percent, while the transverse strain is about -0.2 percent (0.2 percent residual and -0.4 percent mechanical).

Observation (5) leads to the following conclusions:

1. Lamination residual strains will have negligible effect on the mechanical properties of $[0_2/\pm\theta]_S$ angleplied laminates subject to either monotonic or cyclic tensile load.

2. Combined-stress failure theories based on first ply failure and using uniaxial ply fracture strains, and neglecting residual strains, will predict laminate tensile fracture stresses in $[0_2/\pm45]_S$ angleplied laminates which are lower conservative) than the experimental values.

3. The transverse strain due to mechanical compressive load will be tensile, opposite of that shown in Figure 13. This means that lamination residual strain will affect the compressive fracture stress (monotonic or cyclic) or angleplied laminates. Also, combined-stress failure theories, described in item (2) above will overestimate the compressive fracture stress in these laminates. Although results are not presented here, it was found that the residual stress may degrade the thermal fatigue resistance of some angleplied laminates (ref. 3), especially those made from S-glass fiber or low modulus matrix material. These laminates were thermally cycled between 200 and 411 K (-100° and 280° F).

The effects of lamination residual stresses on the initial tangent modulus and Poisson's ratio of $[0_2/\pm45]_S$ angleplied laminates made from boron/reinforced aluminum and tested at various angles to the 0°-ply direction were investigated in Ref. 16. These effects are summarized in Table II. Predicted initial tangent properties are shown with and with no residual strains. As can be seen, the agreement between predicted results and measured data is considerably improved for the modulus when the residual strains are considered. The degree of agreement achieved for the Poisson's ratio is inconclusive and may depend on strain gradients through the laminate thickness (ref. 16).

The effects of lamination residual stresses on the mechanical properties as measured on a thin tube made from $[(\pm45)_2]_S$ boron/epoxy angleplied laminate and subjected to combined loads were investigated in Ref. 17. The transverse ply residual stress is 72.4 MPa (10.5 ksi) which is greater than the ply transverse strength 62.0 MPa (9 ksi), Table III. This means that the plies have failed in the transverse direction and that they cannot support transverse tensile stress. The ply influence coefficients due to mechanical load stress are shown in Table IV. From these influence coefficients it can be seen that the transverse ply stress ($\sigma_{\ell22}$) due to axial load stress (σ_{cxx}) will be relatively small when compared to the longitudinal stress ($\sigma_{\ell22}$), ($\sigma_{\ell22}/\sigma_{\ell11}$ = 0.10/0.90) and similarly for the torsional load stress (σ_{cxy}), ($\sigma_{\ell22}/\sigma_{\ell11}$ = -0.14/1.86 for the +45° ply and $\sigma_{\ell22}/\sigma_{\ell11}$ = 0.14/1.86 for the -45° ply).

Therefore, the transverse ply properties contribute little to the strength of this laminate even in the absence of residual stress. Combined-load fracture-stresses predicted for the tube (ref. 17) by neglecting the transverse contribution are compared with measured data in Table V. Two bounds are predicted. The lower bound is based on the failure of the first ply while the upper bound is based on the failure of the remaining ply. As can be seen even the predicted upper bound stresses are lower (about 4-percent) than the measured values. The major conclusion from this discussion is that the lamination residual stresses have negligible effect on the mechanical properties of $[(\pm45°)_2]_S$ angleplied fiber/resin laminates when subjected to combined mechanical loads.

Summary of Results and Conclusions

NASA Lewis Research Center research in the area of lamination residual stresses in composites has been extensive. Some of the more significant findings from this research over the last 8 years are summarized as follows:

1. Evidence of the presence of lamination residual stresses in angle-plied laminates are transply cracks and warpage of unsymmetric laminates which occur prior to application of any mechanical load.

2. Lamination residual strains have been measured using the embedded strain gage technique.

3. Lamination residual strains result from the temperature difference between cure and room temperature and vary linearly within this temperature range.

4. Measured ply transverse residual strains reach magnitudes comparable to ply fracture strain.

5. Lamination residual stresses can be predicted with reasonable accuracy, compared to measured data, using linear laminate theory. They depend on temperature difference, composite system, ply configuration, and fiber volume ratio. These stresses appear to be insensitive to void volume ratio and composite component geometry such as variable thickness, curvature and twist.

6. Warpage of unsymmetric laminates is very sensitive to small ply misorientations. For example, 0.5^{o} misorientation in one of the 45^{o}-plies will produce a warpage corner deflection of a $[0_2/\pm 45]_S$ angleplied laminate made from HM/epoxy equal to the laminate thickness.

7. Lamination residual stresses have negligible effects on the mechanical properties of $[0_2/\pm 45]_S$ fiber reinforced resin angleplied laminates when they are subjected to tensile monotonic or tensile cyclic loads. However, they may affect the compressive fracture properties and the thermal fatigue resistance.

8. Lamination stresses affect the initial tangent properties of boron reinforced aluminum angleplied laminates.

9. Lamination residual stresses do not affect the fracture stress of $[(\pm 45)_2]_S$ fiber reinforced resin angleplied laminates when these laminates are subjected to combined loads.

10. Lamination residual stresses (strains) are usually present in angleplied fiber composite laminates; they are also present in unidirectional hybrids and superhybrids. For specific applications, the magnitudes of lamination residual stresses should be determined and evaluated relative to the anticipated applied stresses. Particular attention should be given to cyclic thermal loadings in applications where the thermal cycling takes place over a wide temperature range.

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TABLE I. - LAMINATION RESIDUAL STRESSES IN THE MAXIMUM-STRESSED-PLY AT SELECTED SPANWISE AND CHORDWISE SECTIONS IN A FIBER COMPOSITE FAN BLADE (SEE FIG. 10)

Chord fraction	Number of plies per section	Maximum stressed ply	Ply stresses (ksi) (1 ksi = 6.9 MPa)			
			$\sigma_{\ell 11}$	$\sigma_{\ell 22}$	$\sigma_{\ell 12}$	Comb. stress MOS
Root section:						
L.E.	7	6	1.37	6.06	-2.61	0.578
1/8	30	16	-8.61	5.98	.02	.552
1/4	45	31	-4.95	6.04	.00	.568
3/8	58	44	-3.81	6.08	-.01	.568
1/2	71	15	-4.15	5.94	-.02	.586
5/8	70	15	-5.52	6.11	.11	.555
3/4	64	15	-5.75	6.08	.04	.558
7/8	37	15	-7.44	5.99	.00	.559
T.E.	8	8	-----	----	-----	-----
Mid-span section:						
L.E.	6	1	-4.14	5.66	0.00	0.623
1/8	16	1	-3.06	5.63	-.04	.633
1/4	27	27	-2.65	5.64	-.08	.634
3/8	42	15	-3.18	5.66	.00	.628
1/2	60	15	-3.04	5.67	.00	.628
5/8	61	47	-3.29	5.66	.00	.627
3/4	51	2	-3.00	5.62	.07	.634
7/8	26	2	-3.02	5.63	.25	.633
T.E.	6	6	-----	----	-----	-----
Tip section:						
L.E.	5	1	-2.60	5.63	-0.09	0.636
1/8	8	2	-2.68	5.63	.02	.635
1/4	17	1	-2.75	5.62	.03	.636
3/8	33	15	-3.41	5.63	.01	.630
1/2	51	15	-3.05	5.63	.00	.632
5/8	48	15	-2.91	5.61	.00	.635
3/4	33	1	-3.07	5.61	-.03	.635
7/8	17	1	-2.99	5.63	-.08	.634
T.E.	5	5	-----	----	-----	-----

Notes: L.E. = leading edge
T.E. = trailing edge
Unidirectional composite
Material fracture stresses 1 ksi = 6.9 MPa
Longitudinal tensile 170 ksi
Longitudinal compression 150 ksi
Transverse tensile 9 ksi
Transverse compression 20 ksi
Intralaminar shear 17 ksi

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TABLE II. - COMPARISONS OF MEASURED AND PREDICTED RESULTS

[Initial tangent elastic constants; boron/aluminum composites; 4-mil diam fiber; 6061-0 aluminum alloy; 0.50 fiber volume ratio; $\Delta T = -700^\circ$ F.]

Composite ply orientation	Loading angle	Initial tangent modulus 10^6 psi			Initial tangent Poisson's ratio		
		Predicted with		Measured*	Predicted with		Measured*
		No residual strain	Residual strain		No residual strain	Residual strain	
$[0_8]$	0	34	34	34	0.24	0.24	0.22
$[0_2\pm15]_S$	0	33	31	30	.28	.34	.24
	-80	21	15	13	.20	.20	.20
$[0_2\pm30]_S$	0	30	23	23	.33	.72	.33
	30	25	18	16	.33	.06	.09
	-22.5	27	21	22	.34	.31	.33
$[0_2\pm45]_S$	0	27	17	18	.34	.69	.30
	-37.5	25	14	16	.30	.20	.24
$[0_2\pm90]_S$	-37.5	20	14	15	.41	.11	.11

*Measured values were taken at about 10 percent of the composite fracture strain.

Note: 1° F = 0.56 K
10^6 psi = 6.89×10^6 kPa.

TABLE III. - RESIDUAL STRESS +45° OR -45° PLY FOR A TEMPERATURE REDUCTION OF 167 K (300° F)

$$\begin{Bmatrix} \sigma_{\ell 11} \\ \\ \sigma_{\ell 22} \\ \\ \sigma_{\ell 12} \end{Bmatrix} = \begin{Bmatrix} -72.4 \\ (-10.5) \\ 72.4 \\ (10.5) \\ 0 \end{Bmatrix} \begin{matrix} \text{MPa} \\ \text{(ksi)} \end{matrix}$$

TABLE IV. - INFLUENCE COEFFICIENTS FOR CALCULATING PLY STRESSES FROM COMPOSITE STRESSES

$$\begin{Bmatrix} \sigma_{\ell 11} \\ \sigma_{\ell 22} \\ \sigma_{\ell 12} \end{Bmatrix} = \begin{bmatrix} a_{11} & a_{12} & a_{13} \\ a_{21} & a_{22} & a_{23} \\ a_{31} & a_{32} & a_{33} \end{bmatrix} \begin{Bmatrix} \sigma_{cxx} \\ \sigma_{cyy} \\ \sigma_{cxy} \end{Bmatrix}$$

(ply stress) (influence coeffs) (composite stress)

$$\begin{bmatrix} 0.90 & 0.90 & 1.86 \\ .10 & .10 & -.14 \\ -.50 & .50 & 0 \end{bmatrix} \quad +45^{\circ} \text{ ply \& no residual stress}$$

$$\begin{bmatrix} 0.90 & 0.90 & -1.86 \\ .10 & .10 & .14 \\ .50 & -.50 & 0 \end{bmatrix} \quad -45^{\circ} \text{ ply \& no residual stress}$$

TABLE V. - COMPARISON OF FRACTURE STRESSES

[The specimen was loaded to fracture in combined loading-condition (1:0:1.2).]

Stress type	Stress value, ksi		
	Measured	Predicted	
		Lower bound	Upper bound
Axial	20.2	17.9	19.4
Torsional	23.1	20.6	22.3

Note: 1 ksi = 6.89 MPa.

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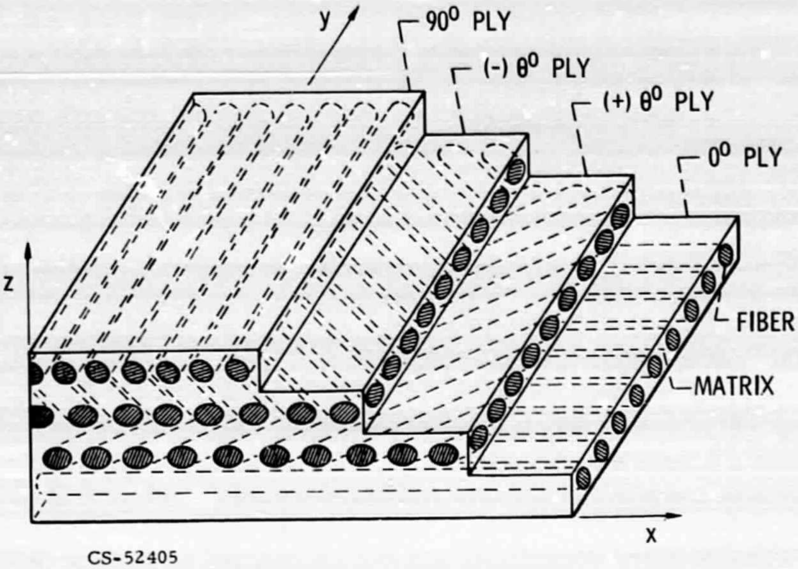

Figure 1. - Schematic of fiber composite laminate geometry (coordinates xyz denote structural axes).

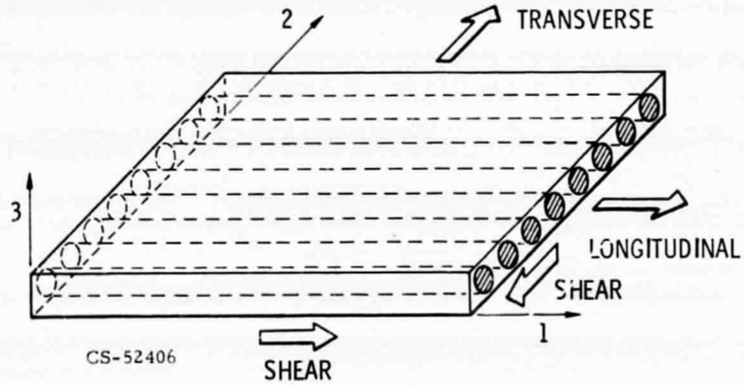

Figure 2. - Schematic of single ply geometry (coordinates 123 denote material axes).

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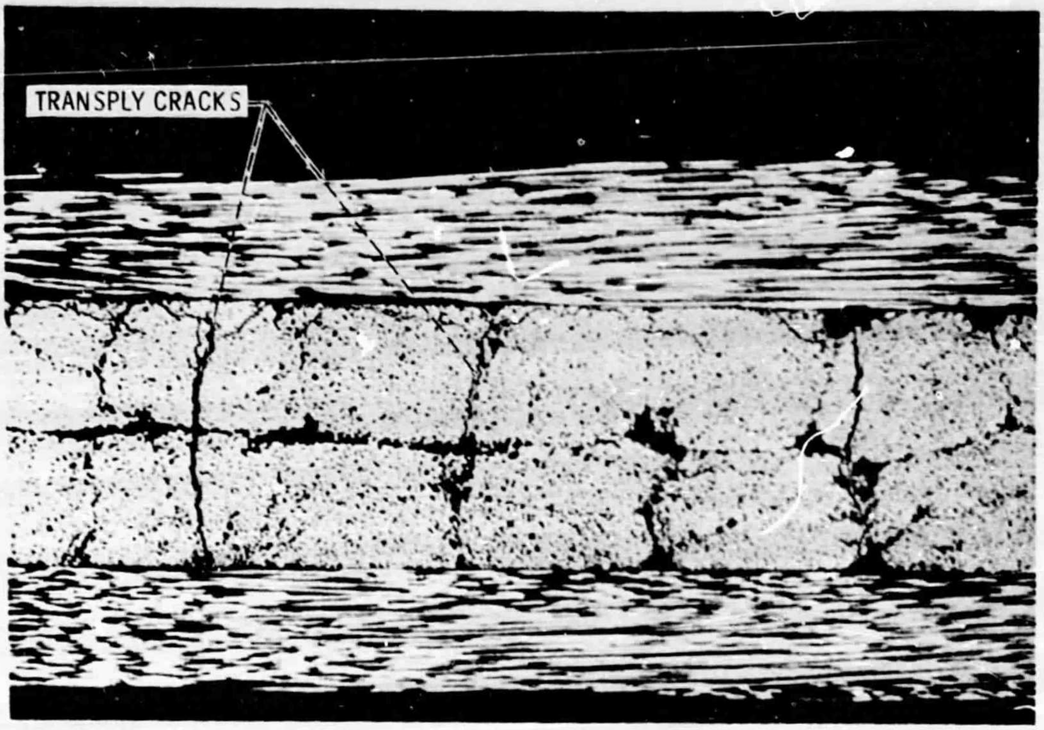


Figure 3. - Photomicrograph showing transply cracks $(0/90)_S$ high-modulus/E poxy.

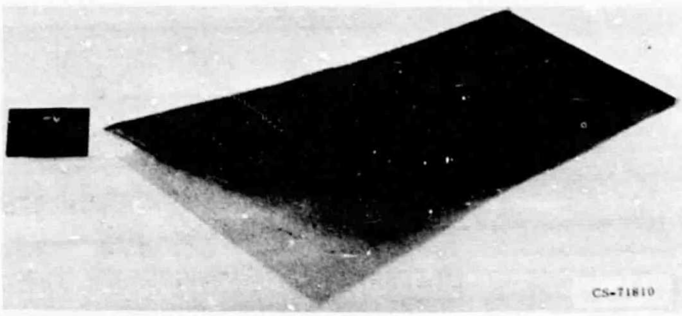


Figure 4. - warped unsymetric kevlar-fabric/E poxy laminate 0/45/-45/0 .

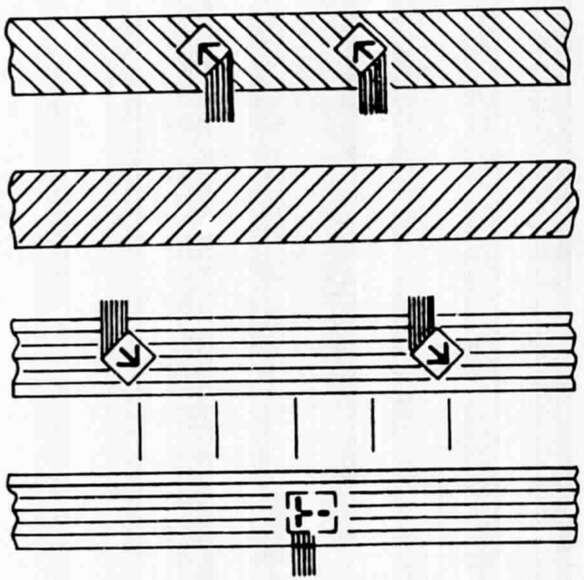


Figure 5. - Strain gage layout in $[0/\pm 45]_s$ laminate for measuring lamination residual strains.

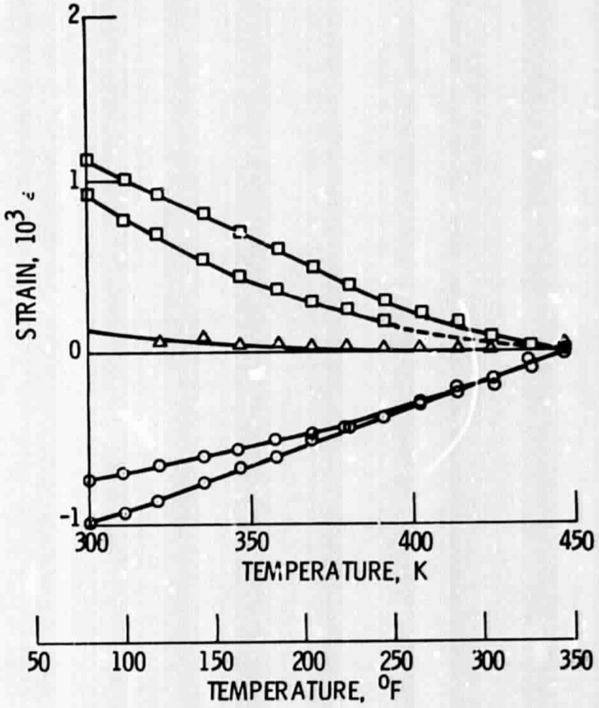


Figure 6. - Apparent strains in $[0_2/\pm 45]_s$ boron/ epoxy laminate during cure. (L, T, D denote longitudinal, transverse, and 45° directions, respectively; 1 and 2 denote top and middle surface locations, respectively.)

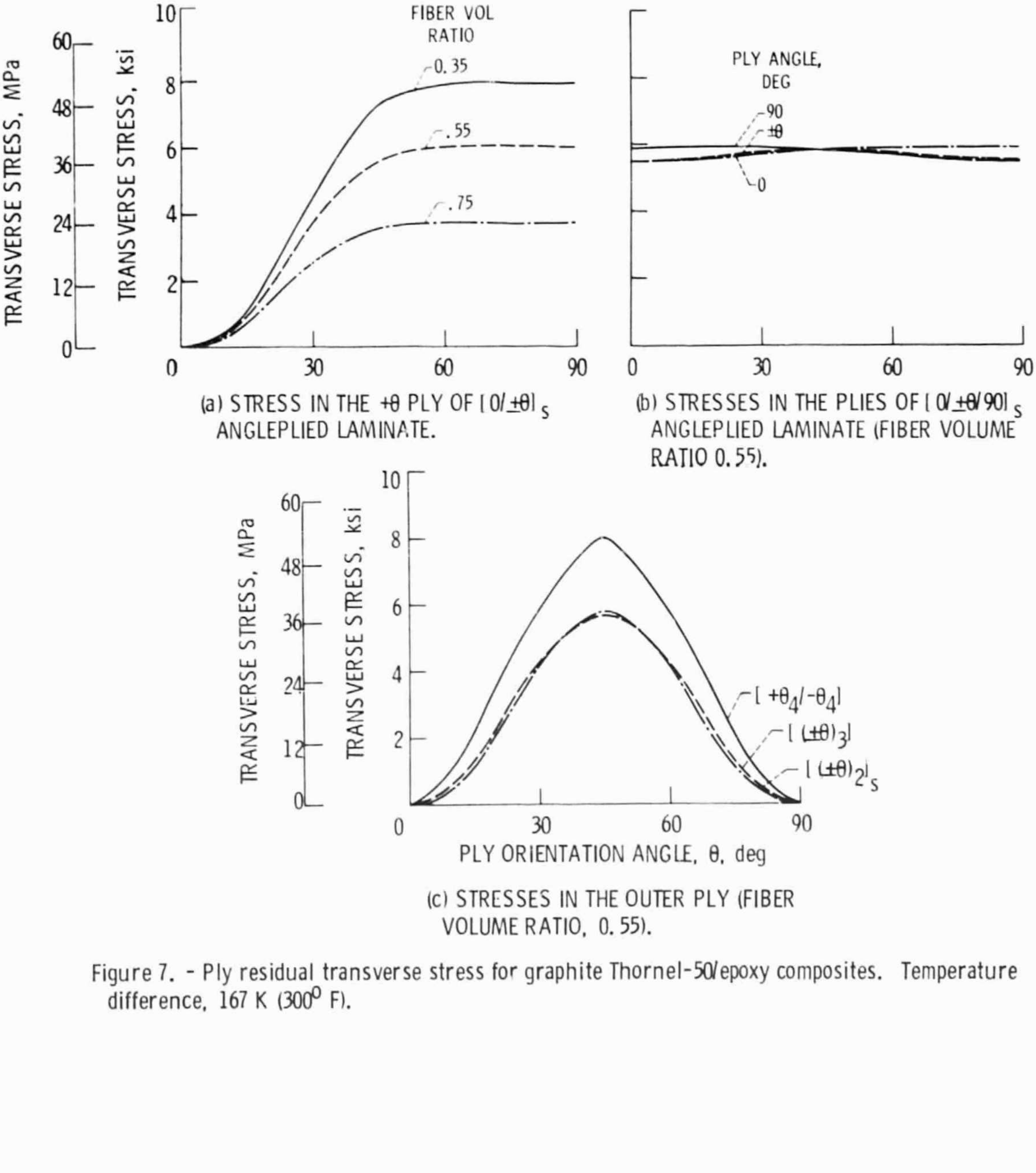


Figure 7. - Ply residual transverse stress for graphite Thornel-50/epoxy composites. Temperature difference, 167 K (300^{0} F).

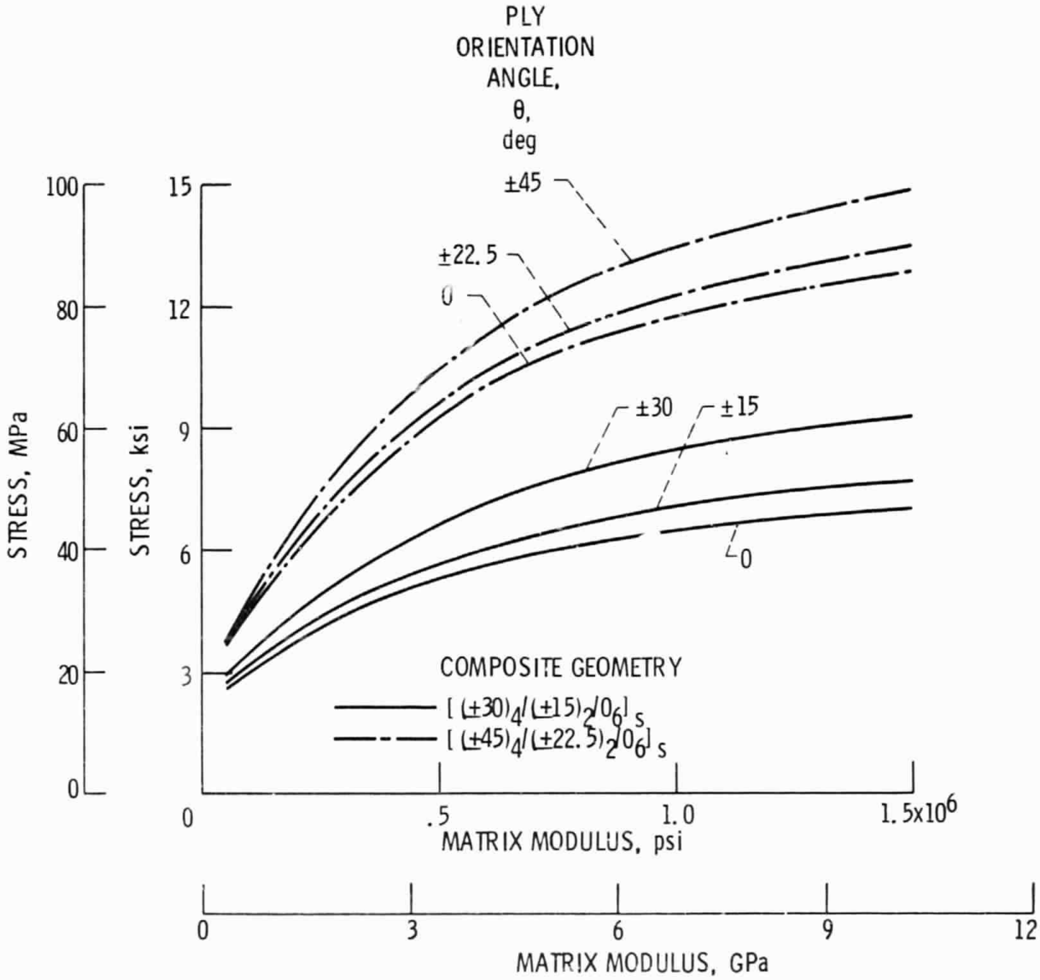
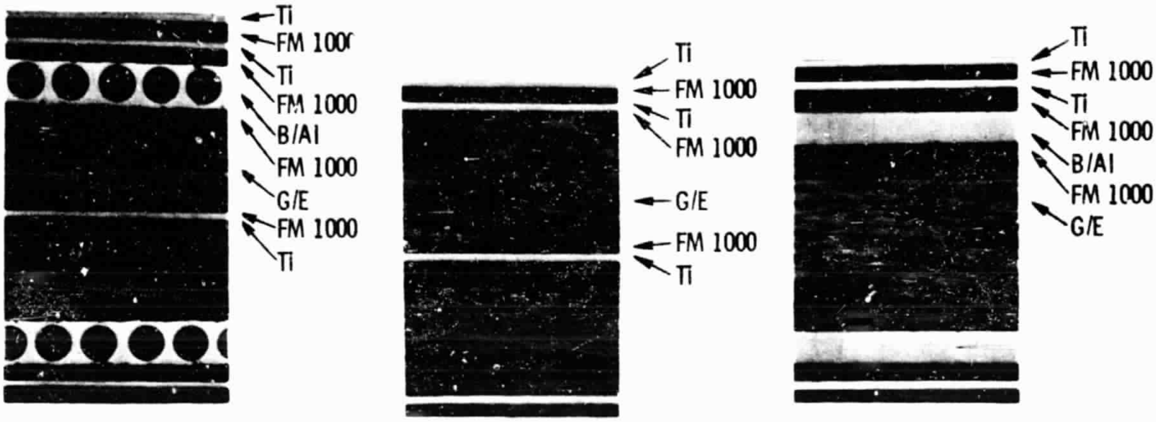


Figure 8. - Effect of matrix modulus on ply residual transverse stress. Modmor-I/polyimide composites; zero void content; fiber volume ratio, 0.50; temperature difference, 333 K (600° F).



PLY	PLY STRESS, MPa/(ksi)								
	LONGIT[a]	TRANSV[b]	S.M.[c]	LONGIT	TRANSV	S.M.	LONGIT	TRANSV	S.M.
TOP TITANIUM	82.7 (12.0)	135.1 (-19.6)	0.95	123.4 (17.9)	139.3 (-20.2)	0.92	86.8 (12.6)	138.6 (-21.1)	0.94
ADHESIVE	24.8 (3.6)	22.1 (3.2)	.67	25.5 (3.7)	21.4 (3.1)	.66	24.8 (3.6)	21.3 (3.1)	.67
B/Al	80.6 (11.7)	16.5 (2.4)	.99	------	------	----	91.0 (13.2)	2.8 (0.4)	.99
Gr/Ep	65.5 (-9.5)	21.4 (3.1)	.83	18.6 (-2.7)	20.0 (2.9)	.85	59.7 (-8.4)	20.0 (2.9)	.85

[a]PARALLEL TO FIBER DIRECTION.
[b]TRANSVERSE TO FIBER DIRECTION.
[c]SAFETY MARGINS (BASED ON COMBINED-STRESS FAILURE CRITERION).

Figure 9. - Lamination residual stresses in superhybrid composites.

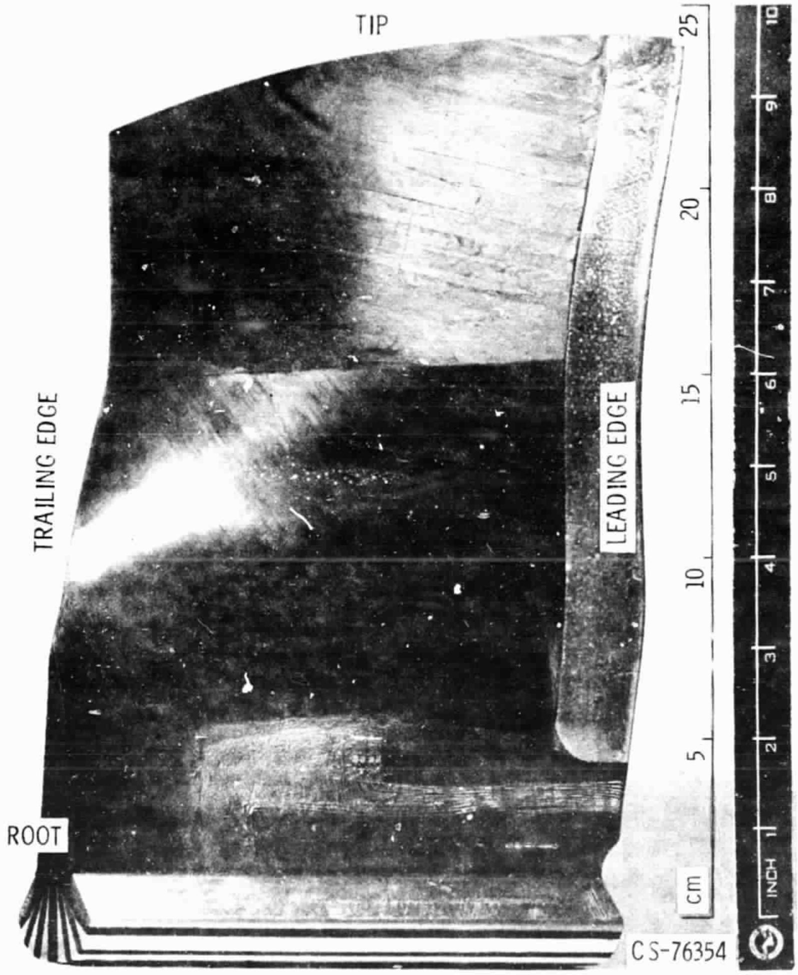


Figure 10. - High-tip-speed composite blade.

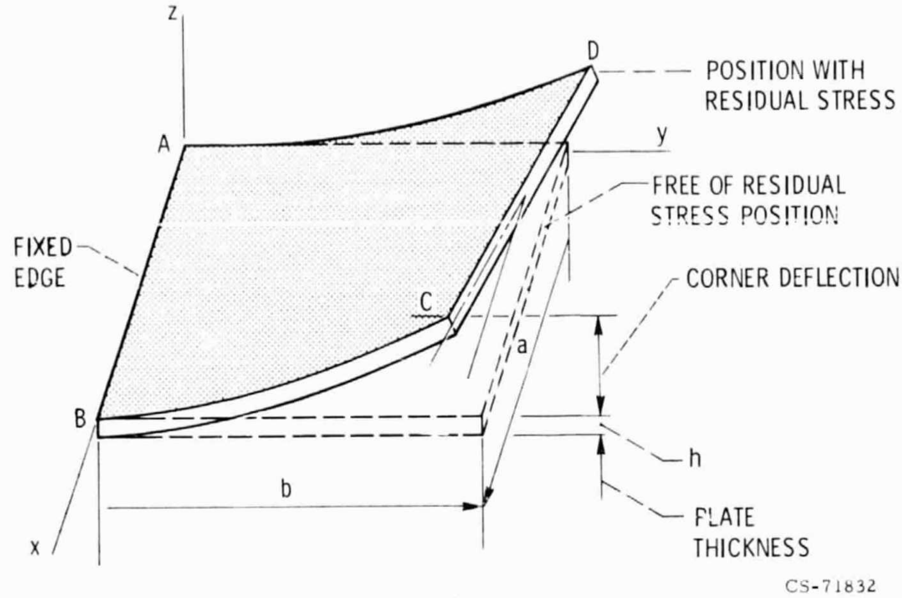


Figure 11. - Schematic depicting corner deflection due to warpage.

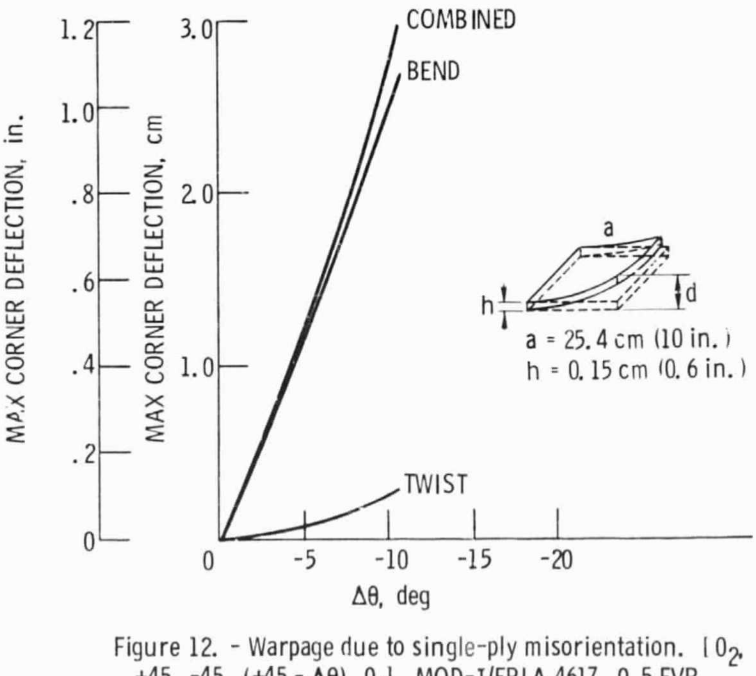


Figure 12. - Warpage due to single-ply misorientation. $[0_2, \pm 45, -45, (+45 - \Delta\theta), 0_2]$, MOD-I/ERLA 4617, 0.5 FVR, $\Delta T = -167$ K (-300° F).

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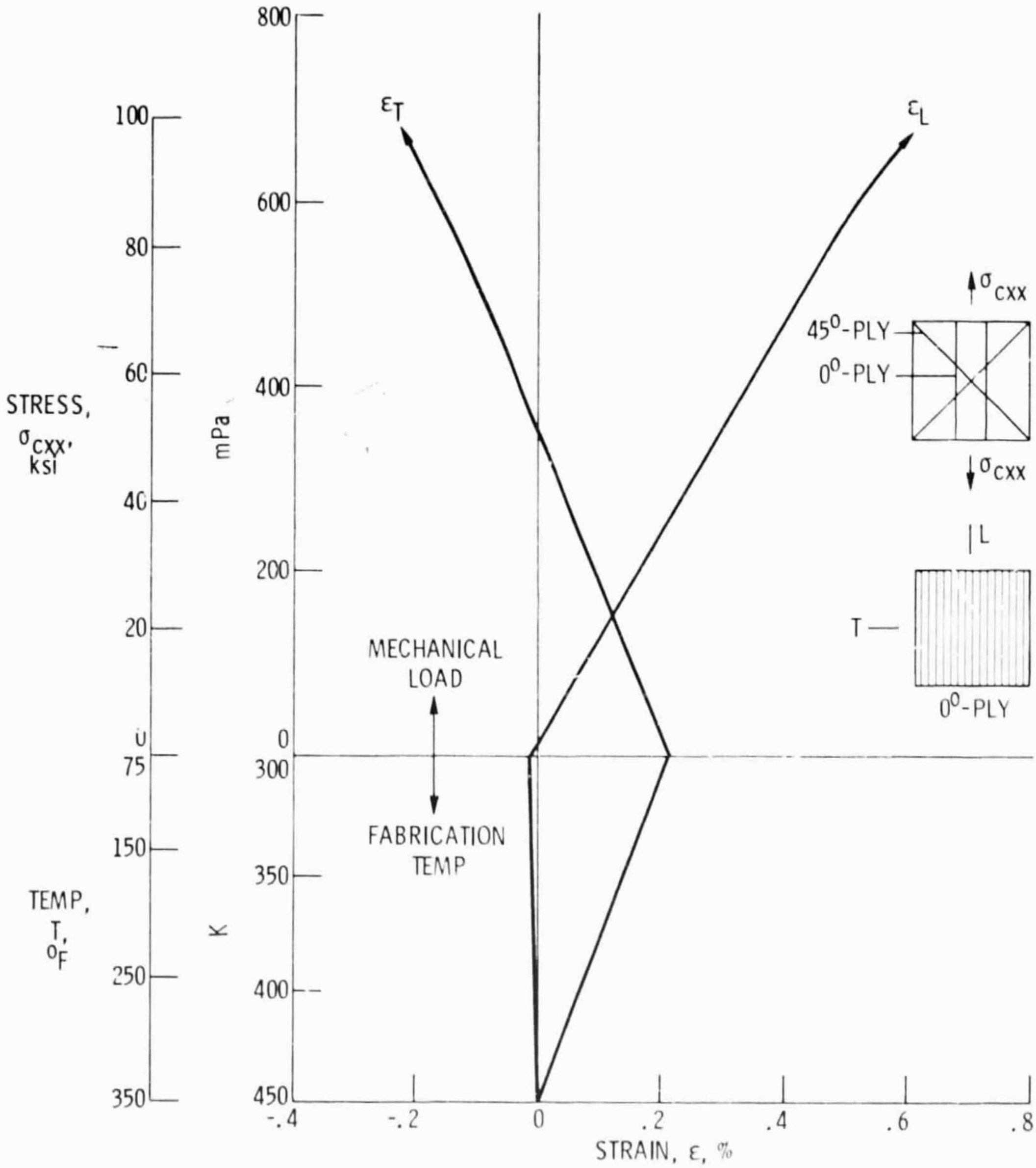

Figure 13. - Total strain history in 0-degree plies of $[0_2/\pm45]_s$ boron/epoxy laminate from curing to failure under uniaxial tension.