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# Prediction of Thermal Cycling Induced Cracking In Polymer Matrix Composites NASA Langley Grant NAG-1-1493

Semi-Annual Report  
for the period February 1993–July 1993

N94-17295

Unclas

63/24 0193696

(NASA-CR-1994-077) PREDICTION OF  
THERMAL CYCLING INDUCED CRACKING IN  
POLYMER MATRIX COMPOSITES  
Semiannual Report, Feb. - Jul. 1993  
(MIT) 36 p

Submitted to Steve Tompkins  
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November, 1993

by

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## **Introduction**

This report summarizes the work done in the period February 1993 through July 1993 on the "Prediction of Thermal Cycling Induced Cracking In Polymer Matrix Composites" program. An oral presentation of this work was given to Langley personnel in September of 1993. As per an oral agreement with Steve Tompkins, this presentation was accepted in lieu of the usual semi-annual report. This document was prepared for archival purposes.

## **Progress**

Studies have been performed on the effects of spatial variations in material strength. Qualitative agreement was found with observed patterns of crack distribution. These results were presented to NASA Langley personnel in November 1992.

The analytical methodology developed by Prof. McManus in the summer of 1992 (under an ASEE fellowship) has been generalized. A method for predicting matrix cracking due to decreasing temperatures and/or thermal cycling in all plies of an arbitrary laminate has been implemented as a computer code. The code also predicts changes in properties due to the cracking.

Experimental progressive cracking studies on a variety of laminates were carried out by graduate student Cecelia Park at Langley Research Center. Results were correlated to predictions using the new methods. Results were initially mixed. This motivated an exploration of the configuration of cracks within laminates.

A crack configuration study was carried out by cutting and/or sanding specimens in order to examine the distribution of cracks within the specimens. These investigations were supplemented by dye-penetrant enhanced X-ray photographs. The behavior of thin plies was found to be different from the behavior of thicker plies (or ply groups) on which existing theories are based. Significant edge effects were also noted, which caused the traditional metric of microcracking (count of cracks on a polished edge) to be very inaccurate in some cases. With edge and configuration taken into account, rough agreement with predictions was achieved.

All results to date were reviewed with NASA Langley personnel in September 1993. View graphs from this presentation are attached. A paper on the previous work was also completed during this period and is attached.<sup>1</sup>




## **Current Status**

Cecelia Park is preparing her Master's thesis, which will fully document all of the work reported here. Work on a small thermal cycling and aging chamber is also in progress. The modified computer code is being documented and prepared for distribution. Work is commencing on the development of a modified theory to account for crack configuration, strength variation, and edge effects. The paper<sup>1</sup>, recently presented at the American Society for Composites 8th Technical Conference on Composite Materials, is being prepared for submission to an archival journal.

## **Bibliography**

1. McManus, H. L., Bowles, D. E., and Tompkins, S. S., "Prediction of Thermal Cycling Induced Matrix Cracking", American Society for Composites 8th Technical Conference on Composite Materials, Cleveland, OH, October 1993.

Submitted:

  
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# PREDICTION OF THERMAL CYCLING INDUCED MATRIX CRACKING

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Accepted for presentation at the American Society for Composites  
8th Technical Conference on Composite Materials  
Cleveland, Ohio, October 19-21, 1993

## ABSTRACT

Thermal fatigue has been observed to cause matrix cracking in laminated composite materials. A method is presented to predict transverse matrix cracks in a composite laminate subjected to cyclic thermal load. Shear lag stress approximations and a simple energy-based fracture criteria are used to predict crack density as a function of temperature. Prediction of crack density as a function of thermal cycling is accomplished by assuming that fatigue degrades the material's inherent resistance to cracking. The method is implemented as a computer program. Simple experiments provide data on progressive cracking of a laminate with decreasing temperature, and on cracking induced by thermal cycling. Correlations of the analytical predictions to the data is very good. A parametric study using the analytical method is presented which provides insight into material behavior under cyclical thermal loads.

## BACKGROUND

The orbital environment is characterized by wide swings in temperature as the vehicle moves in and out of the shadow of the earth. Composites subjected to this thermal cycling suffer from transverse matrix cracks [1]. These cracks cause degradation of material properties, dimensional changes, and changes in the coefficients of thermal expansion (CTE) of the composites. In extreme cases, this can cause the material to simply come apart. More relevant is the changes in properties that can be caused by even modest levels of cracking. Composite laminates are often used in space structures due to their extreme dimensional stability. Such laminates are, however, very sensitive to imperfections, and their CTE's can be changed significantly by matrix cracking levels that are not otherwise a threat to the structure. This problem is under continuing experimental study.

The problem of transverse matrix cracking in laminated composites due to mechanical loading has been extensively studied. It has been observed that Classical Laminated Plate Theory (CLPT) cannot predict transverse cracking; the strain at which transverse cracks appear is dependent on the thickness of the cracking ply-group [2], a dependency which is not predicted by CLPT. In practice, the transverse strength of a group of plies with the same orientation is often treated as a function of the ply group thickness. To be used in design, however, this approach requires data from many different layups for each material under consideration. A predictive methodology based on a shear lag model of the stress distribution around a transverse crack, combined with an energy-based criteria for crack growth, has been studied by many authors. A good review, which will not be repeated here, is given by Nairn [3], and extensive work continues to appear in the literature [4,5]. The method of Laws and Dvorak [6] is particularly well suited to investigations of progressive damage due to monotonically increasing loads.

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Transverse cracking has been recognized as the first stage of damage due to cyclic loading in many laminates [7-9]. The literature concerned with the characterization and prediction of fatigue damage has extensive descriptions and measurements of this phenomena, but is less concerned with predicting it than using it as a metric of damage. In particular, reference is made to a "characteristic state" of damage at which the laminate is allegedly saturated with cracks. The crack distribution at this state is taken to be a laminate property, independent of loading type or history.

A limited literature exists on attempts to use shear lag and fracture mechanics concepts to predict transverse cracking as a function of fatigue loading. Petitpas et al. [10] use a shear lag stress solution and a maximum stress criteria for crack formation, coupled with measurements showing that the effective transverse failure stress is a function of number of cycles, to predict the crack density as a function of cyclical loading. Lafarie-Frenot and Henaff-Gardin [11] propose an empirical relation between crack density and number of cycles in the form of a Paris-type law.

Most of these studies include the important effects of residual curing stresses, which are thermal stresses. None, however, explicitly address the issue of transverse crack formation under thermal loading.

## STATEMENT OF PROBLEM

Given appropriate material properties, the layup and geometry of a composite laminate, and a thermal loading history, we predict the resulting distribution of matrix cracks, and the resulting degraded laminate properties. An analytical model is implemented as a computer program to provide the predictions. It is verified by correlation with experiments, some performed as part of this effort and some garnered from the literature.

## ANALYTICAL MODEL

Thermal or mechanical loading stores strain energy in the laminate. Some of this energy is released by the formation of a transverse crack. Typically, these cracks appear suddenly across the full height and width of a ply or ply-group (group of adjacent plies all with the same alignment angle) parallel to the fibers, Figure 1. The amount of energy released can be approximated by a shear lag solution of the stresses in the vicinity of the crack. The appearance of a crack can then be predicted by a simple energy method- if the energy released  $\Delta G$  is greater than the energy necessary to form the crack  $G_c$ , the crack will form.

A simple shear lag model was derived. It was found to be identical in form to that of Laws and Dvorak. Readers are therefore referred to Reference 6 for details. We will use the notation of Laws and Dvorak with minor variations. We consider a laminate consisting of a ply group with fibers aligned  $90^\circ$  to the  $x$  axis embedded in an arbitrary laminate. Figure 2 shows an edge-on view of such a laminate. A unit depth in the  $y$  direction is assumed. We assume the cracks labeled A and B exist, and wish to find the strain energy released by the appearance of the additional crack C. The thickness of the cracking ply group is  $a_2$ ;  $a_1$  is the combined thickness of the remainder of the laminate, and  $a_0$  is the laminate thickness. Similarly,  $E_2$  is the uncracked axial stiffness of the cracking ply group,  $E_1$  is the axial stiffness of the remainder of the laminate, and  $E_0$  is the stiffness of the entire laminate. These effective stiffnesses can be calculated from the ply properties and laminate geometries using the relations contained in the Appendix. This is, with minor differences in notation, the problem studied by Laws and Dvorak. They derived

$$\Delta G = \frac{a_2(a_1 + a_2)E_0}{2\xi a_1 E_1 E_2} \sigma^2 \left( \tanh \frac{\xi h_1}{a_2} + \tanh \frac{\xi h_2}{a_2} - \tanh \frac{2\xi h}{a_2} \right) \quad (1)$$

where  $\Delta G$  is the change in strain energy due to the appearance of crack C,  $\xi$  is a geometric shear lag parameter, and  $\sigma$  is the axial stress in the cracking ply group prior to cracking. Laws and Dvorak expressed this stress as a function of applied load and residual thermal stresses. We wish to express it in terms of applied thermal load  $\Delta T$ . In the absence of applied



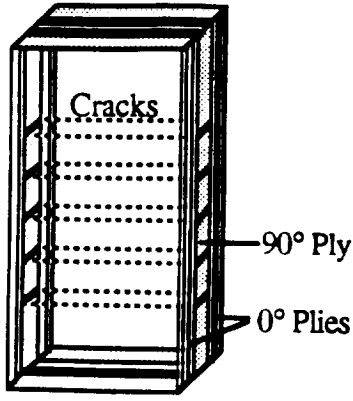


Figure 1. Cracked cross-ply laminate

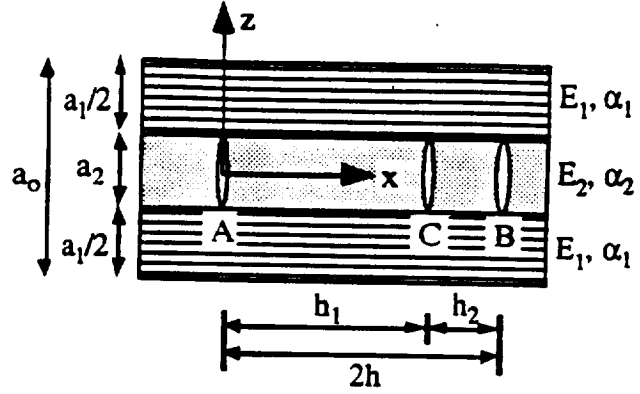


Figure 2. Cracking geometry

mechanical load, the stress due to a change in temperature can be shown to be

$$\sigma = \frac{a_1 E_1 E_2 (\alpha_2 - \alpha_1) \Delta T}{E_1 a_1 + E_2 a_2} \quad (2)$$

where  $\Delta T = T - T_f$  and  $T_f$  is the stress-free temperature of the laminate, often assumed to be the cure temperature. In materials of interest,  $\alpha_2 > \alpha_1$ , so a negative  $\Delta T$  is required for a positive stress. Combining Eqs. 1 and 2, and setting  $\Delta G$  to  $G_{lc}$ , the temperature change necessary to form a crack at location C is found to be

$$\Delta T = - \sqrt{\frac{2 \xi G_{lc} (E_1 a_1 + E_2 a_2)}{a_1 a_2 E_1 E_2 (\alpha_2 - \alpha_1)^2}} \left/ \left( \tanh \frac{\xi h_1}{a_2} + \tanh \frac{\xi h_2}{a_2} - \tanh \frac{2 \xi h}{a_2} \right) \right. \quad (3)$$

The negative value of the square root is selected on the physical grounds that this value of  $\Delta T$  produces positive stress in the cracking ply group.

Multiple cracks are handled by predicting a crack density  $\rho$ , which is the inverse of the average crack spacing. Predictions based on a single parameter  $\rho$  assume an unrealistically even crack spacing, but can be used to calculate either the maximum possible crack density, or the minimum density. For example, if we assume a uniform crack spacing, and assume new cracks will form half way between the existing cracks ( $h_1 = h_2 = h$  in Figure 2), Eq. 3 becomes

$$\Delta T^* = - \sqrt{\frac{2 \xi G_{lc} (E_1 a_1 + E_2 a_2)}{a_1 a_2 E_1 E_2 (\alpha_2 - \alpha_1)^2}} \left/ \left( 2 \tanh \frac{\xi h}{a_2} - \tanh \frac{2 \xi h}{a_2} \right) \right. \quad (4)$$

An existing crack spacing of exactly  $2h$  will allow many more cracks to form at  $\Delta T^*$ , resulting in crack spacing  $h$ ; while an existing crack spacing of just under  $2h$  will not allow any more cracks. Therefore, the minimum possible crack spacing due to temperature change  $\Delta T^*$  is  $h$ , while the maximum possible spacing is just under  $2h$ .

The above formulation does not account for the observed effects of thermal cycling. To account for these effects, we modify the approach of Petitpas et al. [10]. They report a decrease in the first ply failure stress of cross-ply laminates due to mechanical loading cycles. By modifying Eq. 1, with  $h = h_1 = h_2 = \infty$ , and setting  $\Delta G$  to  $G_{lc}$ , we can relate the stress in the cracking ply group at the time of the first crack  $\sigma_c$  to the critical strain energy release rate.

$$G_{lc} = \frac{a_2 (a_1 + a_2) E_o}{2 \xi a_1 E_1 E_2} \sigma_c^2 \quad (5)$$

Given effective failure stress as a function of loading cycles  $N$ , and the original critical strain energy release rate, we can reasonably estimate the effect of cyclical loading on the

critical strain energy release rate by dividing Eq. 5 by the zero-cycle values and rearranging.

$$G_{lc}(N) = G_{lc}(0) \left[ \sigma_{fc}(N) / \sigma_{fc}(0) \right]^2 \quad (6)$$

Given  $G_{lc}$  as a function of number of loading cycles, we can predict the limits of crack density as a function of constant thermal cycles by solving Eq. 4 for  $h$ , with  $\Delta T^*$  the absolute maximum  $\Delta T$  during the cycle, and  $G_{lc}$  being  $G_{lc}(N)$ . This procedure ignores stress ratio effects. They could be included without change in the procedure by using  $\sigma_{fc}(N)$  data collected using the correct stress ratio. The procedure also presumes that all material properties are temperature independent.

The degraded properties of the cracked laminates are now calculated as functions of crack density. Laws and Dvorak [6] give a general derivation of this problem. We will start with their formula for reduced laminate axial stiffness,  $E_o^c$ . Using our notation, and noting  $\rho = l/h$

$$E_o^c = E_o \left/ \left( 1 + \frac{\rho a_2^2 E_2}{2 \xi a_1 E_1} \tanh \frac{2 \xi}{\rho a_2} \right) \right. \quad (7)$$

Clearly, the reduction in laminate stiffness is due to a reduction in the stiffness of the cracked ply group. If  $E_2^c$  is this reduced stiffness

$$E_2^c = \kappa E_2 \quad (8)$$

where  $\kappa$  is a knock-down factor due to the cracks. It can be calculated by combining Eqs. 7, 8 and A4

$$\kappa = E_1 a_1 \left( 1 - \frac{\rho a_2}{2 \xi} \tanh \frac{2 \xi}{\rho a_2} \right) \left/ \left( E_1 a_1 + E_2 a_2 \frac{\rho a_2}{2 \xi} \tanh \frac{2 \xi}{\rho a_2} \right) \right. \quad (9)$$

All of the properties of the cracked laminate can now be calculated by CLPT [12]. The properties of the cracked plies are modified by multiplying the non-axial reduced stiffnesses by the factor  $\kappa$ . The details of these calculations are found in the Appendix.

## COMPUTER PROGRAM

The models of laminate behavior outlined in the previous section were implemented as a computer code, CRACK-O-MATIC. The code takes material properties, laminate geometries, and thermal loading histories, and predicts minimum or maximum crack densities, and the corresponding degraded laminate properties. The code currently performs three types of calculations-

- 1) prediction of the crack density due to monotonic application of a given  $\Delta T$
- 2) tabulation of crack densities and degraded material properties as a function of monotonically decreasing temperature
- 3) tabulation of crack densities and degraded material properties as a function of number of constant thermal cycles

The code is available in user-friendly form, including instructions, from the authors.

## EXPERIMENTS

Two types of experiments were carried out as part of this effort. A group of specimens were exposed to steadily decreasing temperatures, and several groups of specimens were subjected to constant thermal cycles. The decreasing temperature specimens were made from P75/934 material. The thermal cycling laminates were made from prepreg containing continuous graphite fibers of different types and the ERL 1962 epoxy resin from Amoco Performance Products, Inc. All of the composite laminates were fabricated at the NASA Langley Research Center using vendor recommended procedures. Specimens were cut from 12 inch square laminated panels of each material. One side of each specimen was polished so damage could be observed. All specimens were dried to a constant weight in a warm vacuum

oven and stored in a vacuum at room temperature to eliminate moisture absorption effects. The layups and materials used are summarized in Table 1. All specimens measured 3.0 inches by 0.5 inches. Three replicants were used in each test.

The first test measured crack density as a function of decreasing temperature. The specimens were cooled to a target temperature at no more than 10°F/min, held at that temperature for 15 minutes, and returned to room temperature at no more than 10°F/min. A small chamber equipped with liquid nitrogen cooling and electrical resistance heating was used. The specimens were shielded from contact with the heating and cooling elements, and were heated and cooled by circulating air only. After each excursion, crack densities were measured at room temperature by counting cracks under optical magnifications from 100x to 400x. The number of cracks in the middle one inch of each specimens was counted, and the results averaged. Only cracks that were observed to go more than half way through the central 90° ply group were counted. Measurements were taken of the specimens as-received, and after excursions to 0°, -50°, -100°, -150°, -200°, and -250°F. The results are shown in the next section.

Thermal cycling test were then performed. The damage state of each specimen was determined prior to thermal cycling by examining the polished edges with an optical microscope at a magnification of 400x. Specimens were thermally cycled between +/-250°F in a chamber similar to the environmental chamber described in Reference 13. For these tests, the thermal cycle had a period of about 25 minutes. After specimens had been exposed to predetermined number of cycles, the specimen edges were examined for induced damage and then subjected to additional thermal cycles. Cumulative damage was tracked for each specimen. Again, the results are in the next section.

## RESULTS AND CORRELATIONS

The results of the monotonic cooling experiments are shown in Figure 3. The measured crack density is shown as a function of temperature. The triangles represent the means of three specimens, while the bars represent the limits of the individual measurements. It can be seen that the data is very consistent. The experiment was not a true monotonic cooling test, as the specimens were returned to room temperature for crack counting, but the effect of these few thermal cycles were assumed to be negligible compared to the damage done by exposure to successively lower temperatures. The fact that cracks were observed in as-received specimens indicates that the cooling from the cure temperature of 350°F (which was assumed to be the stress free temperature) to room temperature was sufficient to initiate cracking.

The code was used to generate predictions of laminate cracking. The ply property data used is in Table 2. The material stiffness properties are NASA Langley data. The strain energy release rate  $G_{Ic}$  and the shear lag parameter  $\xi$  for the P75/934 material was not known, so the values were adjusted to fit the data. The correlation is shown in Figure 3. The match between the analysis and data is excellent. This is, of course, not surprising given the

Table 1. Tests Performed

Test Type	Material	Layup
Thermal Cycling (+/- 250°F)	T50/ERL1962	[0/+45/90] <sub>s</sub>
	T50/ERL1962	[0/90/0/90] <sub>s</sub>
	P55/ERL1962	[0/+45/90] <sub>s</sub>
	P55/ERL1962	[0/90/0/90] <sub>s</sub>
	P75/ERL1962	[0/+45/90] <sub>s</sub>
	P75/ERL1962	[0/90/0/90] <sub>s</sub>
	P100/ERL1962	[0/+45/90] <sub>s</sub>
	P100/ERL1962	[0/90/0/90] <sub>s</sub>
	P120/ERL1962	[0/+45/90] <sub>s</sub>
	P120/ERL1962	[0/90/0/90] <sub>s</sub>
	Monotonic cooling	P75/934

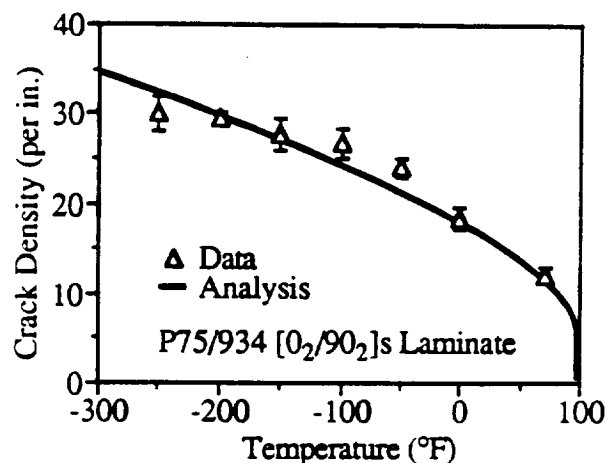


Figure 3. Monotonic cooling data and analysis

Table 2. Ply Properties

Material	$E_t$ (Msi)	$E_r$ (Msi)	$\nu$	$G$ (Msi)	$Y_t$ (ksi)	$G_{lc}$ (J/m <sup>2</sup> )	$\alpha_t$ ( $\mu\epsilon/^\circ\text{F}$ )	$\alpha_r$ ( $\mu\epsilon/^\circ\text{F}$ )	$\xi$
P75/934	34.3	0.9	0.29	0.7	-	40	-0.68	16.0	0.65
T300/934	-	-	-	-	6.00	250	-	-	0.90
T50/ERL1962	28.8	1.04	0.27	0.63	4.36	132	-0.30	18.0	0.90
P55/ERL1962	25.0	1.00	0.34	0.70	3.25	73	-0.38	16.0	0.65
P75/ERL1962	34.3	0.96	0.29	0.70	3.88	104	-0.53	22.0	0.65
P100/ERL1962	51.7	0.87	0.30	0.70	2.67	50	-0.53	21.0	0.65
P120/ERL1962	58.5	0.87	0.28	0.70	2.89	58	-0.68	16.0	0.65

adjustments of the constants, but the constants cannot change the overall trend predicted by the analysis, so the fact a fit was achievable is at least encouraging.

More definite verification of the utility of the analysis can be seen in the fatigue study correlation. In this study, two different layups and five different materials were studied. The analysis used material properties measured at NASA Langley and shown in Table 2.  $G_{lc}$  values were not available, so they were scaled from available transverse ply strength data, and the available  $G_{lc}$  of T300/934 material. Assuming transverse strength  $Y_t$  is a fracture dominated property, it is reasonable to assume that for materials  $r$  and  $s$ ,

$$G_{lc}(r) = G_{lc}(s) [Y_t(r)/Y_t(s)]^2 \quad (10)$$

It was found that this method produced values within the range of measured  $G_{lc}$  values available for some materials. The value of the shear lag parameter was held constant for all fibers of the same type, although it was found by comparison to available data on T300 materials that a value of 0.9 was appropriate for PAN based fibers, while the above correlation suggests 0.65 for pitch based fibers. Given the geometric differences between the fiber types, and the fact that the shear lag parameter is a geometric parameter, this is reasonable.

The dependencies of  $G_{lc}$  on the number of loading cycles for all materials considered were scaled using Eq. 6 and the data of Petitpas [10]. The result for all materials considered was

$$G_{lc}(N) = G_{lc}(0) [1 - 0.0336]^{2 \log_{10} N} \quad (11)$$

The results are shown in Figure 4. Crack density is plotted vs. number of cycles, with the mean of the data shown as symbols and the predictions as lines. The correlations are

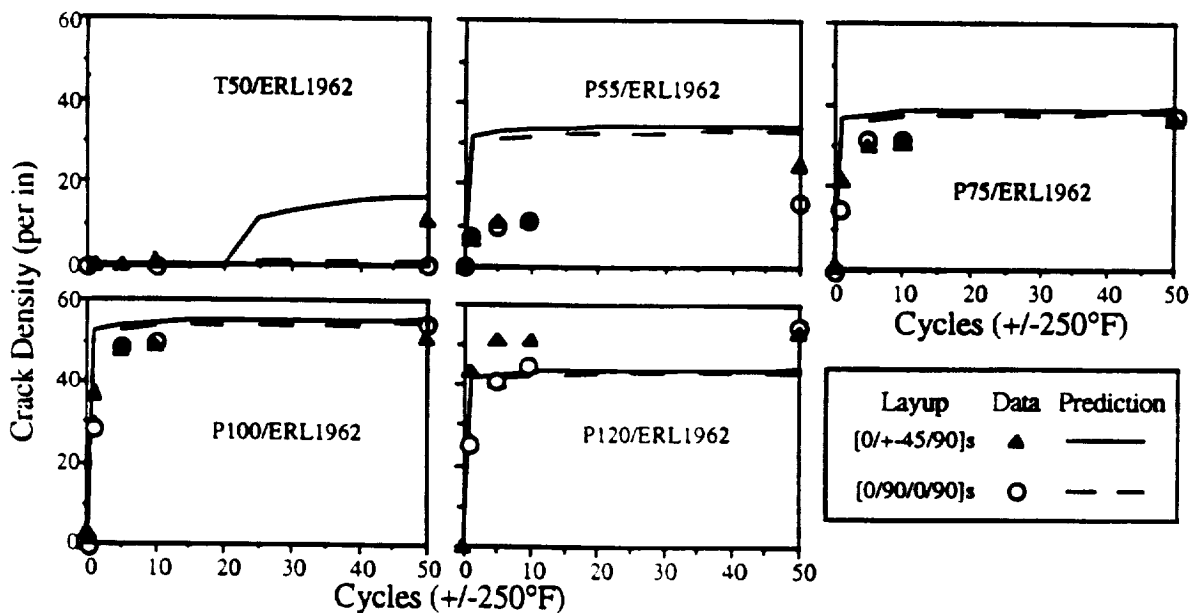


Figure 4. Thermal fatigue data and analysis for several materials and layups

startlingly good. No modifications of any of the analytical inputs were made to achieve these correlations. It can be seen that the effects of variations in material types and layups are successfully predicted, both qualitatively and, with the sole exception of the P55 material, quantitatively as well. Most notable is the difference between the two different layups of T50 material- this subtle difference causes one laminate to crack after a few cycles, while the other remains uncracked, and the effect is successfully predicted by the code.

## PARAMETRIC STUDIES

Once a reasonable level of confidence in the analysis was achieved, an attempt was made to obtain greater physical understanding of the cracking problem through parametric exercises with the code. Simulations of monotonic cooling and thermal cycling were performed, varying material critical energy release rate, cracking layer thickness, and surrounding laminate layup. The effect of crack densities on the laminate properties were also examined as a function of cracking layer thickness.

Figure 5 shows predicted cracking due to cooling for a family of layups with different assumed  $G_{Ic}$ . A  $[0_2/90_2]$  layup and P75/934 elastic properties used for the calculations. Increasing  $G_{Ic}$  decreases both the temperature at which cracking initiates and the level of cracking at any given temperature. Note that in no case does the level of cracking reach a steady state- at least in theory there is not "characteristic damage state" for this laminate suffering damage due to monotonic cooling.

Figure 6 shows the effect of decreasing cracking layer thickness. The baseline  $[0_2/90_2]$  laminate, with four  $90^\circ$  plies in the cracking group, is compared to a  $[0_2/90]$  (with 2) and a  $[0_2/\overline{90}]$ , where the bar indicates that the laminate is symmetric about a single  $90^\circ$  ply. Thinner ply groups delay the onset of cracking, but once cracking starts it continues to higher densities in the thinner ply groups. Figure 7 shows the same study applied to a thermal cycling case. Due to the severe  $\pm 250^\circ\text{F}$  thermal cycles, the thicker ply groups are severely cracked on the first cycle. The laminate with a single (5 mil) center ply shows somewhat different behavior; if the single ply is reduced only slightly in thickness, to 4 mils, the laminate does not initially crack, but does crack after thermal cycling. This is in qualitative agreement with observed behavior [14].

Figure 8 shows the effect of different surrounding laminates on the cracking of a two-ply central  $90^\circ$  ply group in thermal cycling. The effect is noticeable, and the trends non-intuitive. The number of zero degree plies in the  $[0_n/90]$  family of laminates (where  $n$  from 1 to 3 was checked) makes little difference. The more compliant  $[0/+45/90]$  laminate has worse cracking, but the presence of other  $90^\circ$  ply groups in the  $[0/90/0/90]$  laminate seems to help. Again, these results are partially confirmed by experiment- note the T50 results in Figure 4.

The effects of cracking on laminate stiffness and CTE are seen in Figures 9 and 10. Figure 9 shows normalized stiffness loss. The stiffness loss is always fairly small, and is negligible in the case of thin cracking ply groups. Figure 10 shows the laminate CTE; normalization would be confusing in this case as the uncracked CTE's of each laminate are noticeably different. Cracking causes the CTE to change more than 100% of the nominal value, and even change signs, in some cases. Obviously this is an important effect, especially in structures requiring extreme dimensional stability. The change in CTE due to cracking is larger than the expected variation due to manufacturing and material variations [15], and hence may cause the structure to drift outside of design limits for thermal deformations. This effect is much less severe for thin cracking layers.

## CONCLUSIONS AND RECOMMENDATIONS

The method of shear lag analysis combined with an energy method for crack prediction can be used to predict thermal cycling induced matrix cracking in composite laminates. The method has been verified by comparisons with several different kinds of experimental results. A parametric study using the analytical method gives good insight into the physical problem. One important result is that laminates with thinner ply groups crack at lower temperatures and/or larger number of thermal cycles, but when they do crack they will eventually reach higher crack densities. On the other hand, the effect on laminate properties when thin ply

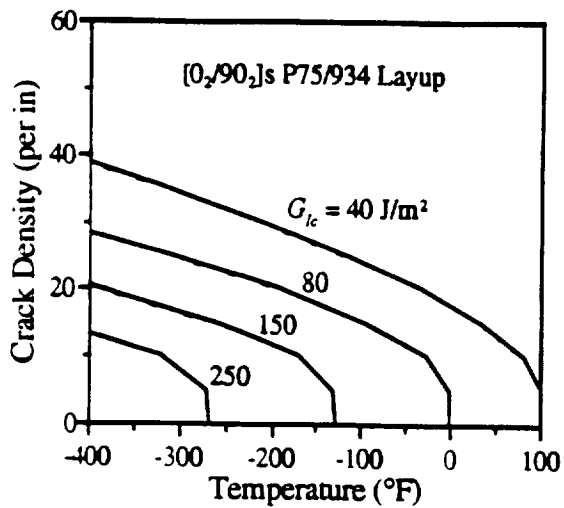


Figure 5. Sensitivity of crack density to  $G_{ic}$  in monotonic cooling

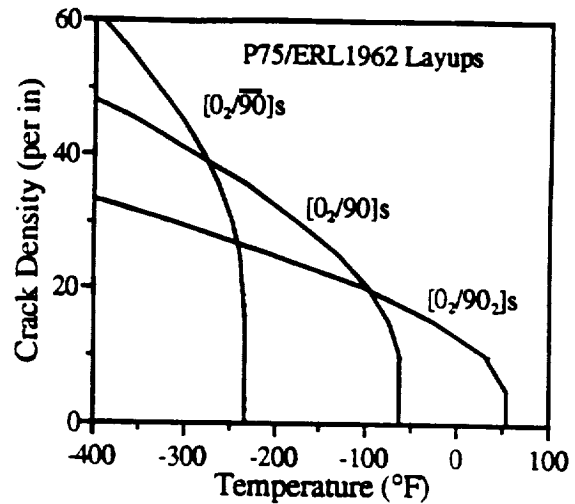


Figure 6. Sensitivity of crack density to cracking layer thickness in cooling

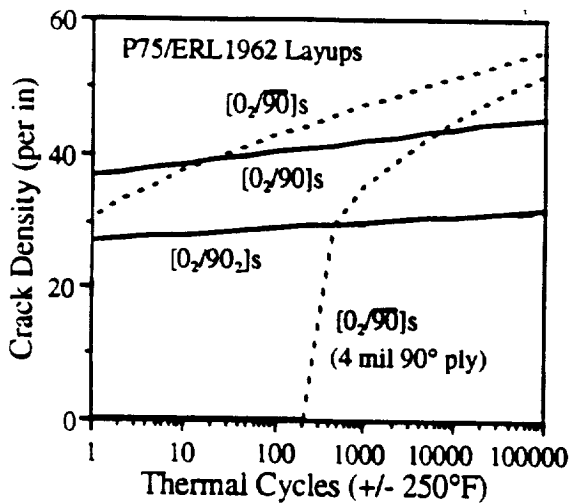


Figure 7. Sensitivity of crack density to cracking layer thickness in thermal cycling

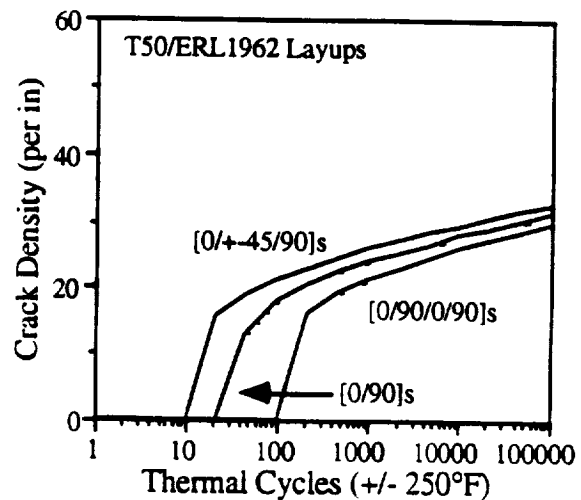


Figure 8. Sensitivity of crack density to surrounding layup in thermal cycling

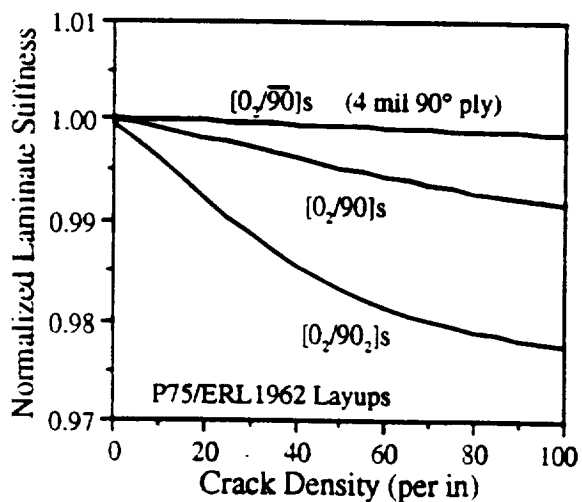


Figure 9. Stiffness loss due to cracking

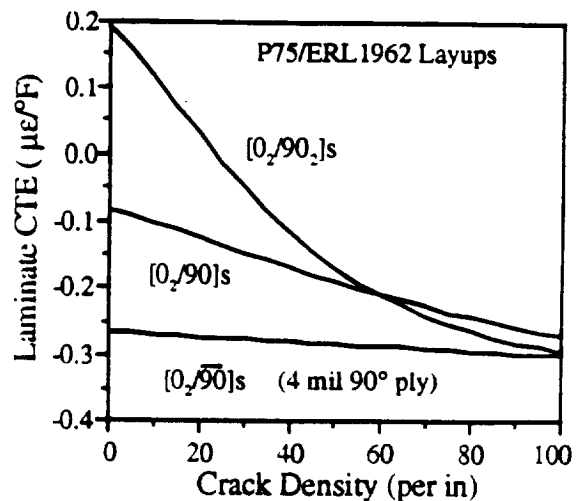


Figure 10. CTE change due to cracking

groups crack is much smaller than the effect caused by cracking of thick ply groups. It was also found that the material's transverse strain energy release rate is important to the onset and eventual density of cracking, and that higher values are desirable. Less intuitive was the finding that the layup has an important effect on the level of cracking, but that no one factor dominates this effect.

These results are significant in light of recent efforts to reduce matrix cracking in composite space structures through the use of thinner prepreg plies [14]. The hope has been to eliminate cracking entirely. Our results indicate that even if no cracks are evident after a few thermal cycles, cracks may form after further thermal cycling, and ultimately may reach a higher density that would be reached with thick plies (see Figures 6 and 7). However, these cracks have little effect on the laminate properties, as seen in Figures 9 and 10. Thin plies may be useful for enhancing the stability of space structures, but for different reasons than those originally proposed. It is also evident that, when comparing laminates with different ply group thicknesses, crack density is alone is not a useful damage metric.

Further work is needed to make this method useful for quantitative, predictive analyses. Accurate critical strain energy release rate data, preferably over a wide range of temperatures and collected using a variety of specimen types and materials, is required. The method also needs to be generalized so that cracking in more than one layer, and in layers with ply angles of other than  $90^\circ$ , can be predicted. Effects such as temperature dependence of material properties, stress ratio effects, and random load cycles also need to be included.

## ACKNOWLEDGMENTS

The work reported here was sponsored by an American Society for Engineering Education/NASA summer faculty fellowship. The preparation of this paper was supported by a National Science Foundation Young Investigator Grant. Graduate student Cecelia Park assisted in the preparation of the manuscript.

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## APPENDIX

We assume that the cracked laminate is much wider than it is thick. Therefore, the 2-dimensional problem presented in Figure 2 is a plane strain problem. The appropriate equivalent stiffnesses  $E_o$ ,  $E_1$ , and  $E_2$  are calculated by CLPT [12]. For each ply  $i$ , we have measured ply properties  $E_{ii}$ , (fiber direction stiffness),  $E_{\bar{ii}}$  (transverse stiffness),  $\nu_i$  (major Poisson's ratio),  $G_i$  (shear stiffness),  $\alpha_i$  (fiber direction CTE), and  $\alpha_{\bar{i}}$  (transverse CTE). The ply has thickness  $t_i$ , and the fibers are aligned at an angle  $\theta_i$  to the  $x$  axis. The cracking ply group is treated as a single ply  $k$ , where  $\theta_k = 90^\circ$ . The equivalent stiffnesses are then calculated as follows. For each ply  $i$

$$Q_i = \begin{bmatrix} Q_{11(i)} & Q_{12(i)} & 0 \\ Q_{12(i)} & Q_{22(i)} & 0 \\ 0 & 0 & Q_{66(i)} \end{bmatrix} \quad (A1)$$

$$Q_{11(i)} = E_{ii}/D_i \quad Q_{12(i)} = \nu_i E_{\bar{ii}}/D_i \quad Q_{22(i)} = E_{\bar{ii}}/D_i \quad Q_{66(i)} = G_i/D_i \quad D_i = 1 - \nu_i^2 E_{ii}/E_{\bar{ii}}$$

$$\bar{Q}_i = T_i^{-1} Q_i T_i^{-T} \quad (A2)$$

$$T_i = \begin{bmatrix} \cos^2 \theta_i & \sin^2 \theta_i & 2 \sin \theta_i \cos \theta_i \\ \sin^2 \theta_i & \cos^2 \theta_i & -2 \sin \theta_i \cos \theta_i \\ -\sin \theta_i \cos \theta_i & \sin \theta_i \cos \theta_i & \cos^2 \theta_i - \sin^2 \theta_i \end{bmatrix}$$

For the laminate

$$A = \sum_{i=1}^n \bar{Q}_i t_i \quad (A3)$$

$$E_o = A_{11} / a_o \quad E_2 = Q_{22(k)} \quad E_1 = \frac{E_o a_o - E_2 a_2}{a_1} \quad a_o = a_1 + a_2 \quad (A4)$$

With damage in layer  $k$

$$Q_{11}^c = Q_{11(k)} \quad Q_{22}^c = \kappa Q_{22(k)} \quad Q_{12}^c = \kappa Q_{12(k)} \quad Q_{66}^c = \kappa Q_{66(k)} \quad (A5)$$

and we recalculate Eqs. A1-A3 using these values in Eq. A1 when assembling  $Q_k$ . The effective stiffness  $E^{eff}$  and effective CTE  $\alpha^{eff}$  of a laminate are defined as

$$E^{eff} = \frac{1}{A'_{11} a_o} \quad A' = A^{-1} \quad (A6)$$

$$\alpha^{eff} = \alpha'_{11} \quad \alpha' = A' \sum_{i=1}^n \bar{Q}_i \bar{\alpha}_i t_i \quad \bar{\alpha}_i = T_i^T \alpha_i \quad \alpha_i = \begin{Bmatrix} \alpha_{ii} \\ \alpha_{\bar{ii}} \\ 0 \end{Bmatrix} \quad (A7)$$



# THERMALLY INDUCED MICROCRACKING IN COMPOSITE STRUCTURES

Cecelia H. Park



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Thesis Advisor: Prof. Hugh McManus

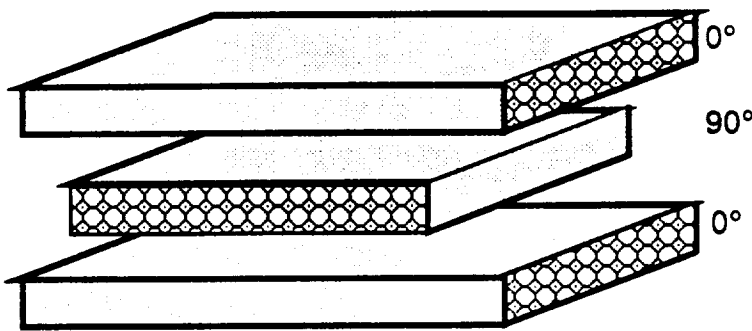
August, 1993

## **PROBLEM**

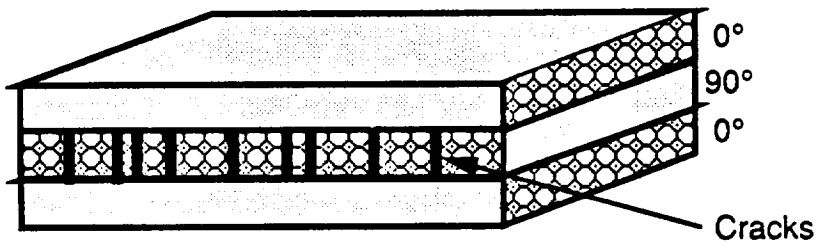
- Structures are subjected to extreme temperatures/thermal cycles in space (+/- 150°F in LEO and +/- 250°F in GEO)
- Thermal cycling causes transverse microcracks in plies of composite structures
- Matrix cracking has been shown to change laminate properties
- In dimensionally critical applications, cracking may degrade performance
- Greater understanding of this problem is necessary. Want to be able to predict matrix cracking and its affect on laminate properties

# BACKGROUND

## Crack Formation In Thermally-Loaded Composites



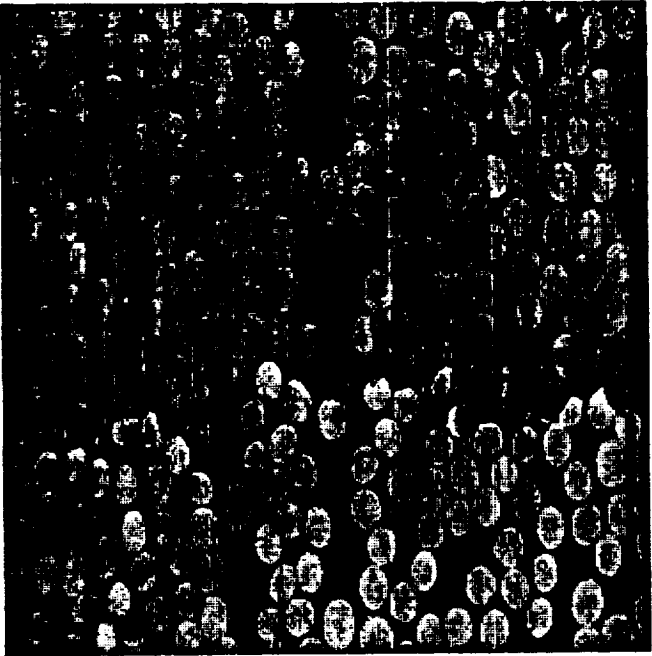
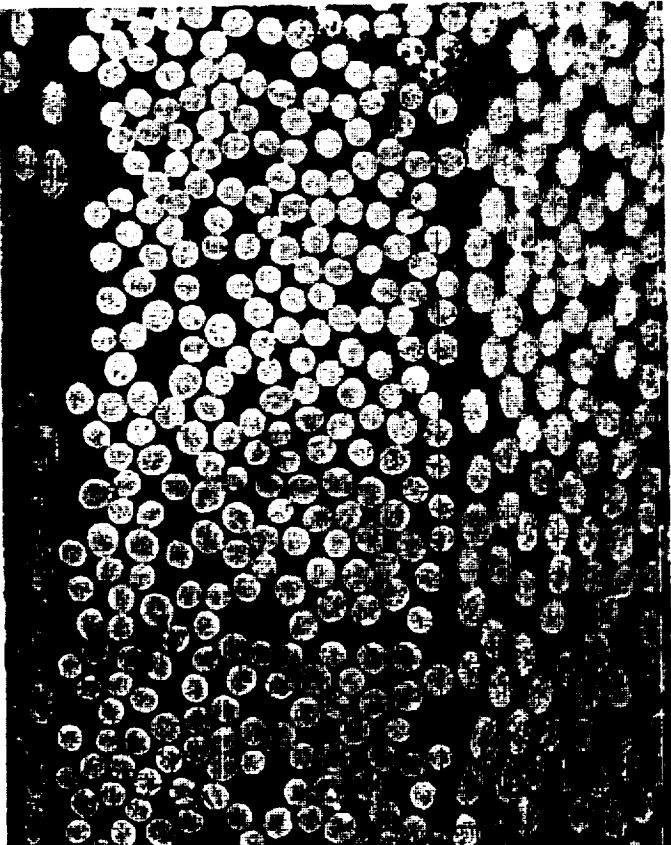
Unconstrained Plies



Constrained Laminate

Cracks develop due to thermal (CTE) mismatch

**CLOSEUP VIEW OF MICROCRACKING BEHAVIOR**



## **PREVIOUS ANALYTICAL WORK**

- **Modified CLPT**
- **Microstructural**
- **Statistical Strength Distributions**
- **Fracture Mechanics Approach**

**Most of the previous work involves:**

**Mechanical progressive damage  
Crossply laminates**

**Some work on:**

**Mechanical fatigue  
Thermal initiation**

**Our problem requires thermal, progressive and fatigue  
damage study for all types of laminates**

## **SUMMARY OF WORK BY H.M.**

- Thermal progressive and fatigue damage study for crossply laminates
- Analytical model uses shear lag approximation of stress state and energy-based fracture criteria to predict crack density as a function of temperature and thermal cycles.
- Crack density predictions correlate well with experimental data
- Cracking has significant effect on CTE
- Less effect on stiffness
- Overall, good results for analysis of crossplies

## **RESEARCH OBJECTIVE**

Develop analytical model so that given the following:

Material Properties  
Layout and Geometry  
Thermal Loading History

Find:

Crack Density  
Degraded Laminate Properties

**For all plies and arbitrary layout**

## **GENERAL TASKS:**

- Develop analysis/computer program to predict crack density and laminate properties for any general laminate
- **Experiments to obtain progressive cracking and fatigue damage data**
- **Crack configuration study**
- Crack distribution analysis
- Put together thermal environmental chamber to be used for thermal progressive and cycling experiments



# EXPERIMENTAL PROCEDURE

## Progressive Thermal Loading

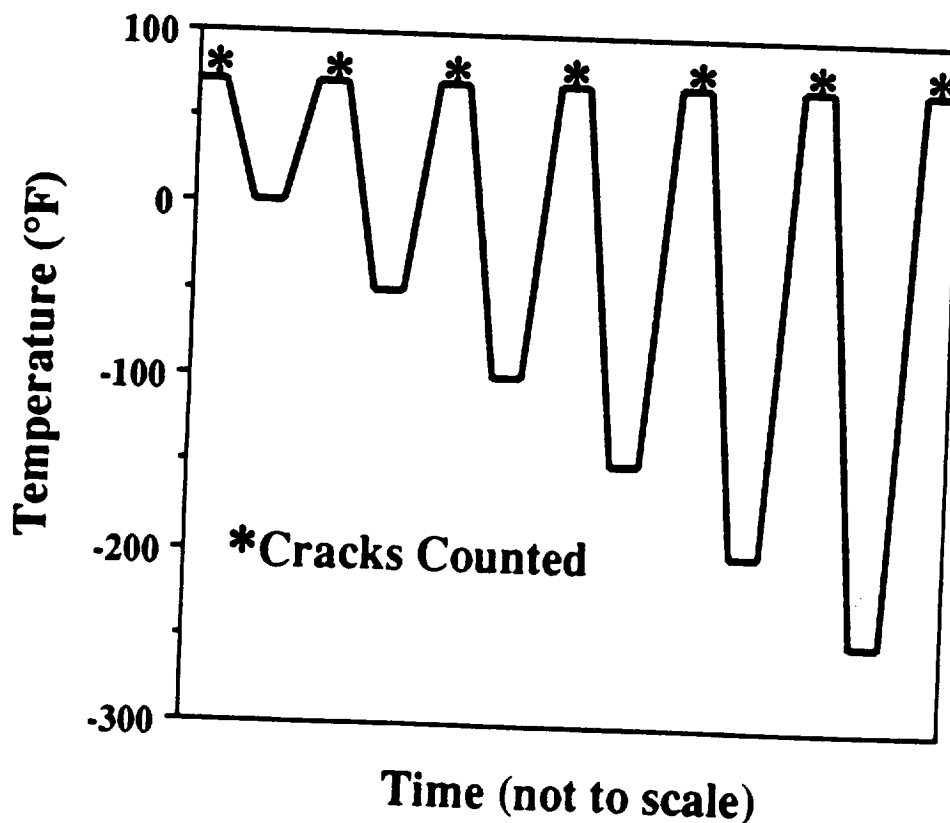
Material: P75/934 Graphite Epoxy

Size: 0.5 X 2.5 inches

Layups: [0<sub>2</sub>/90<sub>2</sub>]<sub>s</sub>                      Quantity: 3 each  
          [0/+45/90/-45]<sub>s</sub>                    9 total  
          [0<sub>2</sub>/±30]<sub>s</sub>

Specimens cooled to progressively lower temps. (70° to -200°F) in thermal environmental chamber

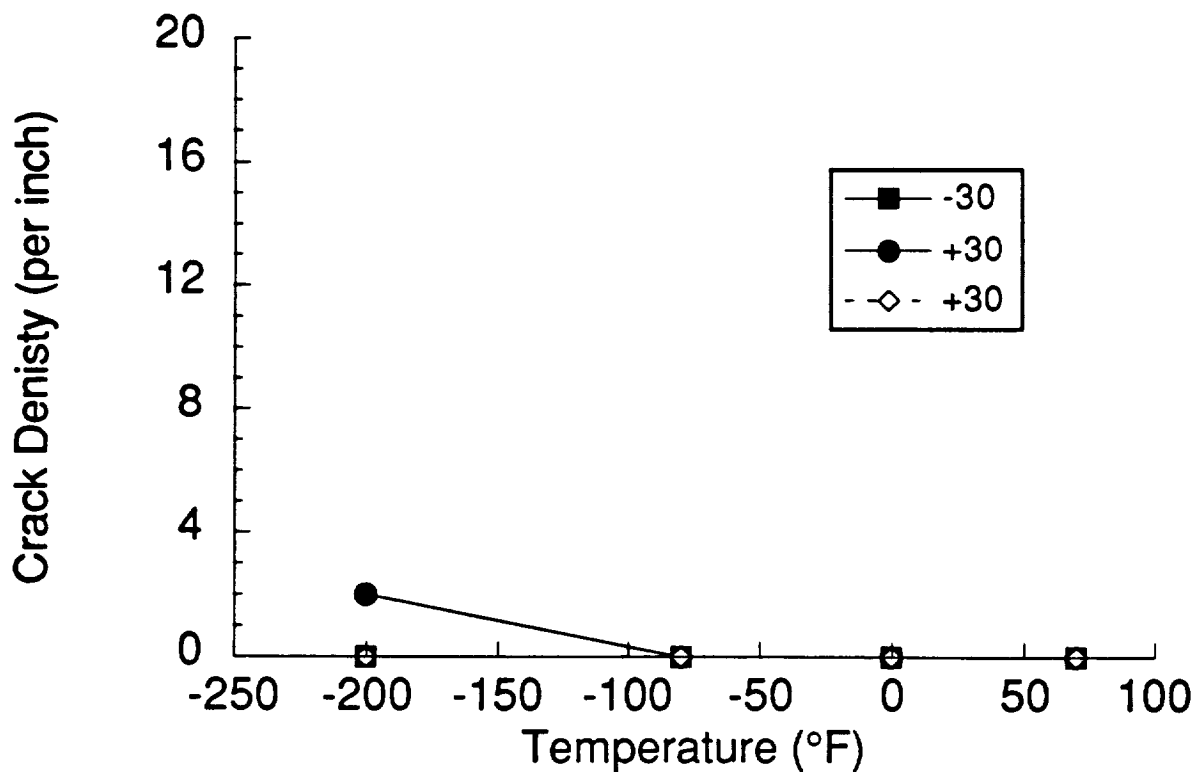
Crack density (cracks/inch) and distribution taken on polished edge (Side A) using optical microscope.



# EXPERIMENTAL RESULTS

## PROGRESSIVE LOADING

### [0<sub>2</sub>/±30]<sub>s</sub>

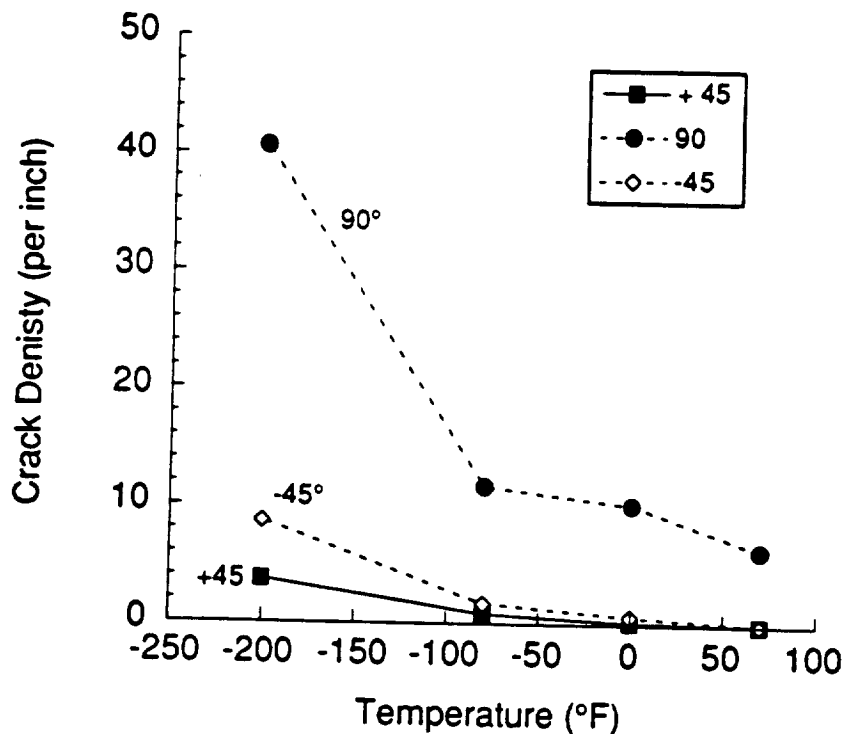


- Specimens hardly cracked, some cracking in +30 plies
- Preliminary analysis predicts zero cracks

# EXPERIMENTAL RESULTS

## PROGRESSIVE LOADING

### $[0/+45/90/45]_s$



- Trends for the middle -45 plies match the preliminary analysis
- Preliminary predictions for 90° plies start to crack too late but end up with same density
- +45° plies start at correct temperature but crack density is low
- 90° and +45° data does not seem to agree

## **CRACK CONFIGURATION STUDY**

### **MOTIVATION:**

- Preliminary analysis for general laminates does not correlate well with experimental data
- Determine if assumptions in our model are valid
- Better understanding/ interpretation of experimental data

### **EXPERIMENTS:**

- 1) Xray photography
- 2) Inspection of other edge (Side B).
- 3) Sanding edges

# XRAY PHOTOGRAPHY

## RESULTS:

**[0<sub>2</sub>/90<sub>2</sub>]<sub>s</sub>**: Cracks propagate through entire width of specimen. About one crack/inch does not go through.



**[0/+45/90/-45]<sub>s</sub>**: Thick middle -45 layer showed good behavior like the crossplies but pictures not as clear. Evidence of some cracks in other plies after cooling

**[0<sub>2</sub>/±30]<sub>s</sub>**: Inconclusive

## CONCLUSIONS:

- Cracks in [0<sub>2</sub>/90<sub>2</sub>]<sub>s</sub> specimens behave 'ideally'
- Except for middle -45 layer, cracks either too thin/small to see on photographs or do not propagate through the width
- [0/+45/90/-45]<sub>s</sub> and [0<sub>2</sub>/±30]<sub>s</sub> specimens need further examination

# EDGE COMPARISON OF SIDE A AND B

## RESULTS:

Specimen	Ply	Crack Density	Distribution
[02/90]s	90°	Matched	Matched
[0/+45/90/-45]s	-45°	Matched	Matched
	+45°	Similar	No Match
	90°	No Match!!	NA
[02/±30]s	-30°	NA	NA
	+30°	Similar	No Match

- CONCLUSIONS:**
- Thick plies behave well. Crack propagate through
  - Some cracks in thin plies do not propagate through.
  - Cracking behavior is dependent ply thickness

## **SANDING EXPERIMENT**

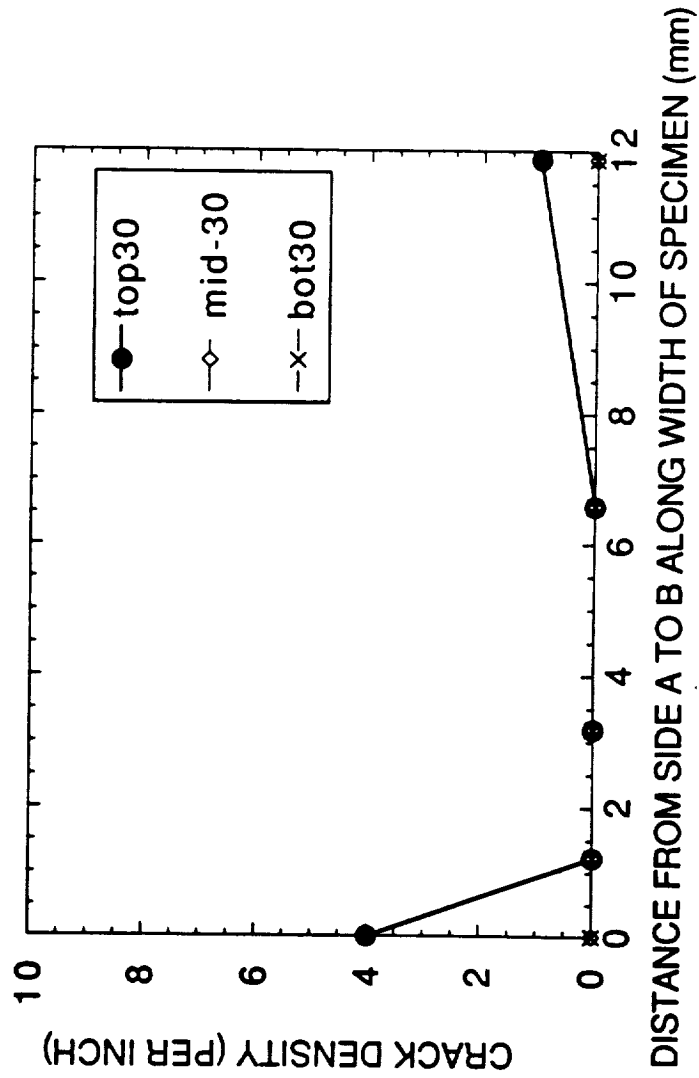
### **MOTIVATION:**

- Understand cracking along width of specimen. Investigate how far and what percentage of cracks run through specimen

### **PROCEDURE:**

- Sanded off approximately 1mm, 2mm, 3mm etc. In such intervals through width of specimen
- Used #180 'rough' sandpaper first then finished off last 10% with #600 'fine' sandpaper to avoid damage
- After each sanding, polished/cleaned specimen
- Crack density and distribution taken for  $[0_2/\pm 30]_s$  and  $[0/+45/90/-45]_s$  specimens

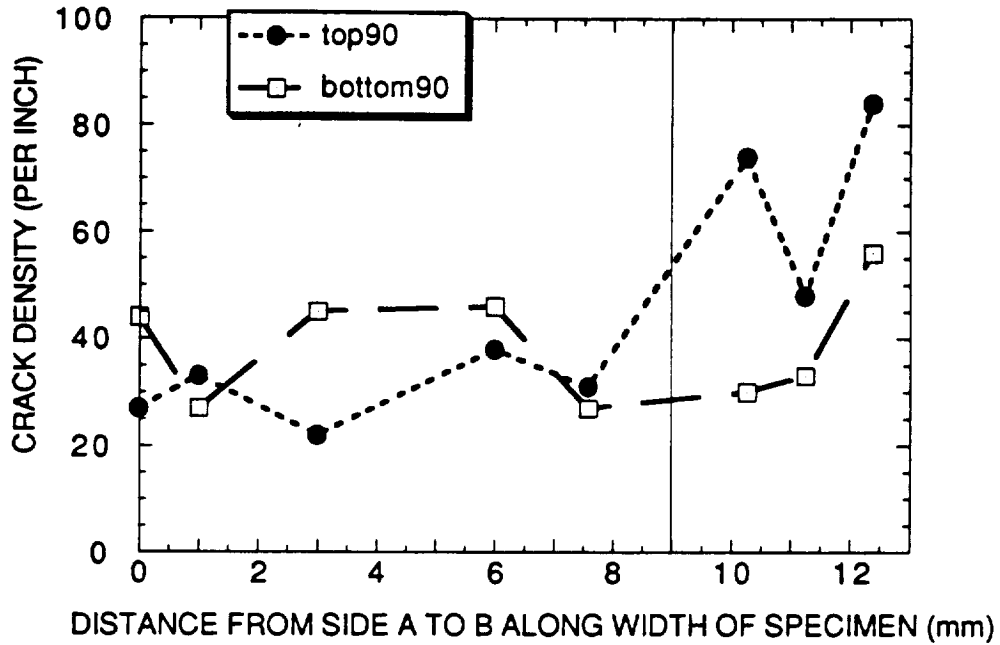
# SANDING EXPERIMENT RESULTS [0<sub>2</sub>/±30]s SPECIMENS



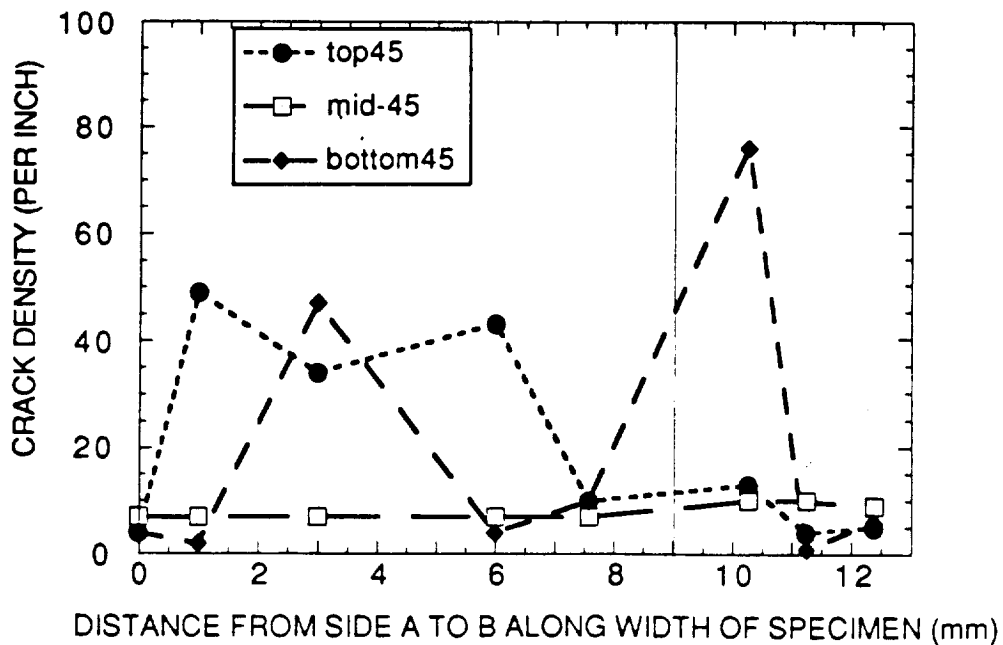


# SANDING EXPERIMENT RESULTS [0/+45/90/-45]<sub>s</sub>

## 90° PLY DATA



## + 45° AND -45° PLY DATA



## **CONTROL TESTS**

### **TESTS AND PROCEDURES:**

- 1) Revised sanding methods - sanded off only small section of width (.04mm-.15mm) using:
  - 'Rough' #180 sandpaper
  - 'Fine' #600 sandpaper

New cracks should not form within the small widths sanded.

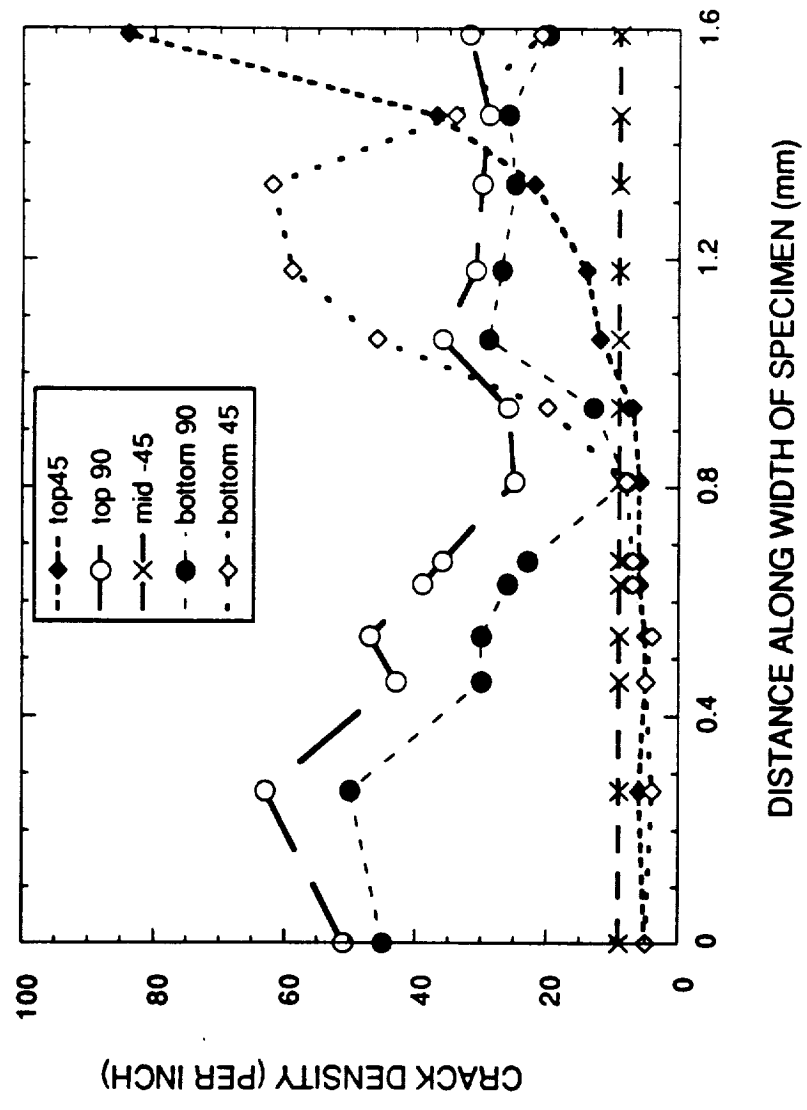
- 2) Sanding new specimens, not subjected to thermal loads.
  - Sanded new [0/+45/90/-45]<sub>s</sub> specimens 'mimicking' three procedures used so far (fine, rough and fine/rough combo)
  - Cracks will only appear in new 'undamaged' specimens if sanding causes the damage.

## **CONTROL TESTS**

### **OBSERVATIONS AND CONCLUSIONS:**

- $\pm 45^\circ$  plies did not have any cracks before sanding and never formed any more after any type of sanding test
- $90^\circ$  plies in general more prone to damage from sanding, 'tempermental' ply. Fine sanding did not damage
- Previous graph for quasi's seems reasonably valid.
- Data from fine sanding experiments seem to be indicative of real cracking behavior

**SANDING RESULTS USING FINE SANDPAPER**  
**[0/+45/90/-45]<sub>S</sub>**



## **SANDING EXPERIMENT**

### **CONCLUSIONS:**

- Thick plies behave. Density and distribution remain constant through entire width
- Thin plies all shattered within specimen.
- Appears from distribution study in thin plies that very few cracks go through width if any.
- Edge crack count not good indication of actual damage for thin plies

## **OVERALL CONCLUSIONS FROM CRACK CONFIGURATION STUDY**

- Thick plies behave 'ideally'
- Assumptions in analysis valid for thick plies
- Thin plies do not behave 'ideally'
- Previous successful work has been in application to thick plies
- Composite structures with thin plies must be treated differently, may require new type of analysis. Reevaluate original assumptions
- Must be cautious in using edge crack density count as measure of damage

## **FUTURE WORK (SEPTEMBER)**

- Complete analysis/computer program
- Experiments at Langley:
  - Progressive and Fatigue damage tests to correlate with analysis
  - Test specific specimens to support ideas from crack configuration study
  - Distribution data
  - More info on how to set up thermal chamber
- Modify analysis for thin plies???

