INTERLAMINAR FATIGUE OF FIBER-REINFORCED LAMINATES

by

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Abstract

The possible role of interlaminar residual stresses induced during the molding processes as a governing factor in delamination was investigated. Consideration of the differences of thermal contraction coefficients resulted in a thermal induced edge normal load between the bonded lamina.

In an effort to test the hypothesis, three configurations of 0-90 angle-ply specimens with the same magnitude of singularity were statically and dynamically loaded by bending moments. It was found that tensile fatigue around surface imperfections was the dominant fatigue process although interlaminar fatigue was present. The hypothesis was not validated from these experiments.

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I. Introduction

Since the uses of fiber-reinforced laminates have been expanding in the aerospace industry and the possible uses increasing in the automotive industry, there is a growing need for an accurate analytical method for designing structures of these laminates to resist fatigue. Interlaminar fatigue or delamination of the layers is a critical mode of failure. Pipes and Pagano[5][6] have solved for interlaminar stresses in purely elastic laminate structures. Their theory connects interlaminar fatigue with an interlaminar normal force resulting from interlaminar shear. Grossman[3] obtained results for 30 and 60 angle-ply configurations that appeared to substantiate the qualitative aspects of their theory.

While a laminate is being fabricated with unidirectional plies, it is subjected to a temperature of several hundred degrees in order to properly cure the epoxy in the matrix. Upon cooling, residual shear stresses are induced between the laminae, because the thermal coefficients parallel and perpendicular to the fiber are different. The resins needed to bond these laminae exhibit plastic deformation in shear, and one would suppose that interlaminar deformation of laminates must entail substantial plastic, as well as elastic deformation of the resins. Pipes and Pagano dealt with purely elastic deformations. The purpose of this investigation is to determine whether a simple plastic model of induced interlaminar residual stresses might be used as a governing factor in the fatigue life of laminates. A mathematical model has been proposed for the singularity arising from the residual stresses in a 0-90 angle-ply composite. Also specimens have been constructed and tested.

II. Mathematical Model

In order to determine if the residual stresses induced between laminae during the molding processes are a governing factor in interlaminar fatigue, a mathematical model of the resulting stresses needed to be developed. The 0-90 angle-ply composite was used as the basic configuration. The mathematical model was restricted to unbalanced two-lamina construction and balanced three-lamina construction. Rigid-plasticity was assumed in the interlaminar matrix. This meant that if any shear strain was present, the shear stress would have a constant value.

Unbalanced Composite

The unbalanced configuration consists of two laminae in which one is rotated by 90 degrees with respect to the other. Considering the free-body diagram of the cross-section of the configuration (see Fig. 1), the 0 angle-ply lamina, horizontal fiber, is represented by layer two, and the 90 angle-ply lamina, vertical fiber, is represented by layer one. During the cooling phase of the molding processes, the horizontal lamina contracts more than the vertical lamina in the x direction. The difference in thermal coefficients results in a shear stress, τ and a distributed force "f" between the laminae. Summation of the moments about points A and B give

$$M_1 - \frac{t_1}{2} \int_o^x \tau(\ell)d\ell - \int_o^x f(\ell)(x - \ell)d\ell = 0 \qquad (1)$$

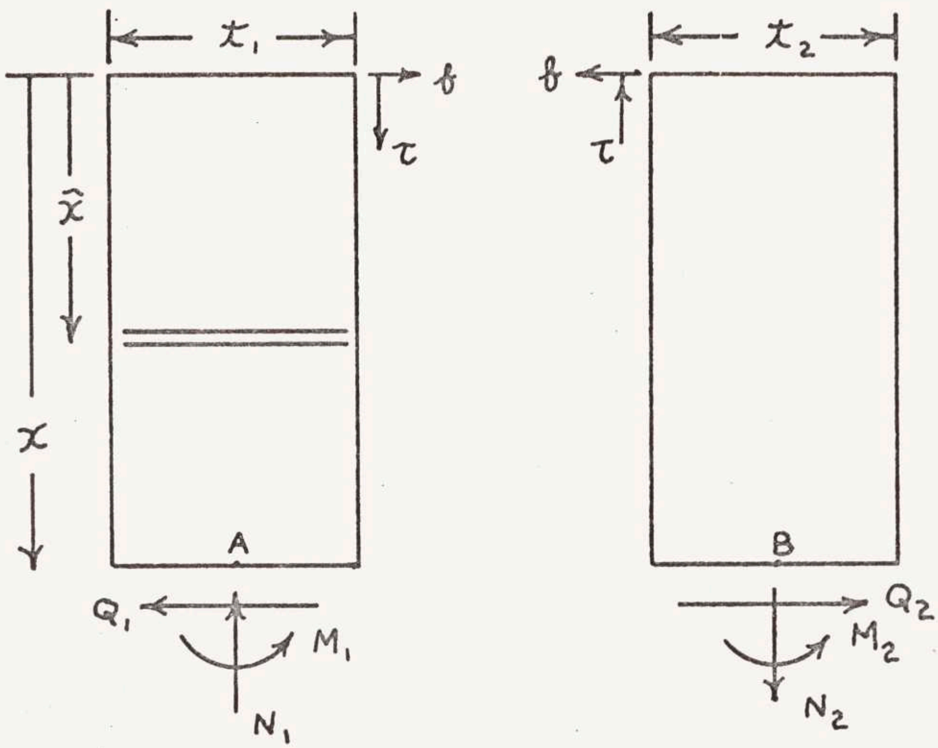

Fig. 1. Unbalanced Composite

$$M_2 - \frac{t_2}{2}\int_o^x \tau(\hat{x})d\hat{x} + \int_o^x f(\hat{x})(x - \hat{x})d\hat{x} = 0 \qquad (2)$$

Since a rigid-plastic model is being used, the shear stress, $\tau(\hat{x})$, will be henceforth represented as "k".

In the unbalanced configuration, the lamina thickness is small compared to the radius of curvature of the composite; therefore, the radius of curvature of each lamina is assumed to be equal.

$$\frac{M_1}{E_1I_1} = \frac{M_2}{E_2I_2} \qquad (3)$$

Substituting equations 1 and 2 into equation 3 gives

$$\frac{1}{E_1I_1}\int_o^x [\frac{t_1}{2}k + f(\hat{x})(x - \hat{x})]d\hat{x} = \frac{1}{E_2I_2}\int_o^x [\frac{t_2}{2}k - f(\hat{x})(x - \hat{x})]d\hat{x} \qquad (4)$$

Further simplification gives

$$\int_o^x \{k\left(\frac{\frac{t_1}{2E_1I_1} - \frac{t_2}{2E_2I_2}}{\frac{1}{E_1I_1} + \frac{1}{E_2I_2}}\right) + f(\hat{x})(x - \hat{x})\}d\hat{x} = 0 \quad . \qquad (5)$$

The partial derivative with respect to x results in

$$\frac{k}{2}\left(\frac{\frac{t_1}{E_1I_1} - \frac{t_2}{E_2I_2}}{\frac{1}{E_1I_1} + \frac{1}{E_2I_2}}\right) + \int_o^x \frac{\partial}{\partial x}(f(\hat{x})(x - \hat{x})d\hat{x}) = 0 \qquad (6)$$

If the coefficient of k/2 is labeled Λ, equation becomes

$$\frac{k}{2}\Lambda + \int_o^x f(\hat{x})d\hat{x} = 0 \qquad (7)$$

or

$$\int_0^x f(\hat{\ell})\,d\hat{\ell} = -\frac{k}{2}\Lambda \tag{8}$$

Based on equation 8, the thermal induced stresses result in a singular normal load at the edge of the lamina.

Balanced Composite

The balanced configuration consists of three laminae. The ply angle on the outside laminae are the same. In the free-body diagram of the cross-section (see Figure 2), the outer laminae have horizontal fibers and the middle lamina has vertical fibers. The difference in thermal coefficients again result in a shear stress between the laminae. The non-curvature of the balanced configuration means that

$$\tau_3 = \tau_4 = \tau_1 = \tau_2 + f_2 = f_3 = f_1 = f_4 \quad . \tag{9}$$

Summation about point A gives

$$\frac{t_1}{2}\int_0^x \tau_1(\hat{\ell})\,d\hat{\ell} - \int_0^x f_1(\hat{\ell})(x - \hat{\ell})\,d\hat{\ell} = 0 \quad , \tag{10}$$

but $\tau_1(\hat{\ell}) = k$ as before. The partial derivative with respect to x results in

$$t_1 \frac{k}{2} - \int_0^x f_1(\hat{\ell})\,d\hat{\ell} = 0$$

or

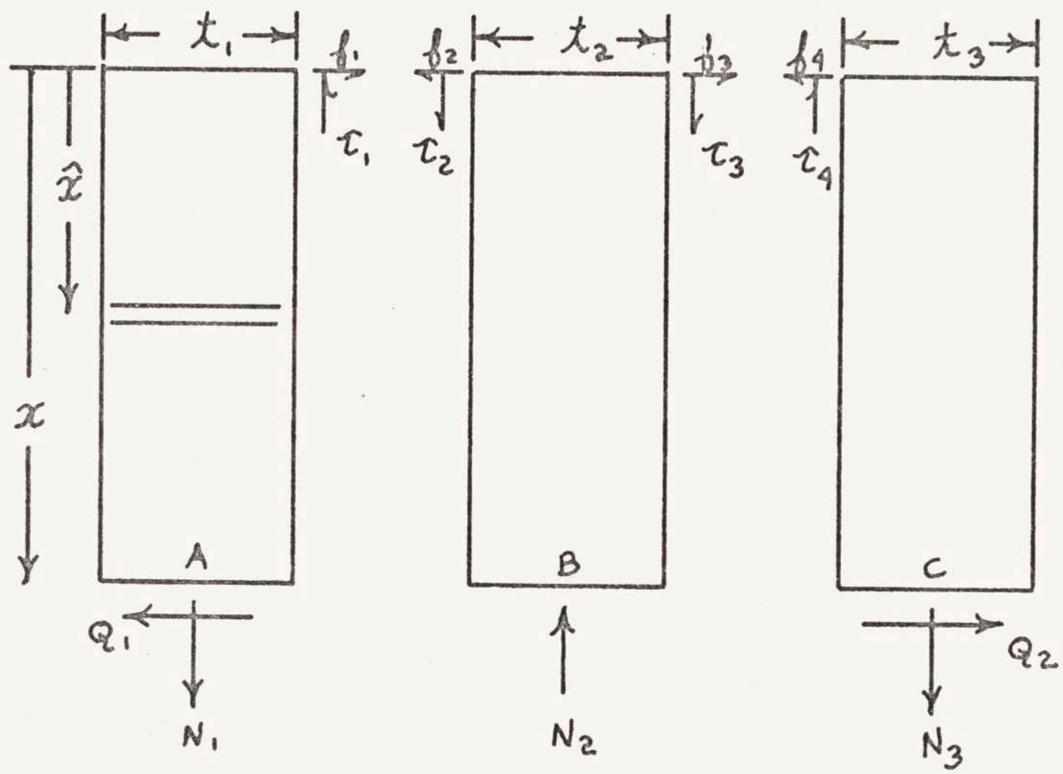

Fig. 2. Balanced Composite

$$\int_{o}^{x} f_1(\ell)d\ell = t_1 \frac{k}{2} \qquad (11)$$

Summation about point c gives the same equation, because the thicknesses of the outer laminae are equal. If the ply angle is changed in the three laminae so that the outer laminae have vertical fibers and the center lamina has horizontal fibers, the resulting singularity will be in compression and not tension.

Both of the mathematical models result in a normal edge load on the interface between laminae that is some multiple of k/2. The following points also pertain to the edge singularity:

1. It is independent of applied load.
2. It depends on the change in temperature during molding processes.
3. It depends on the configuration of the composite (e.g., thickness).
4. It is effected by modulus in the unbalanced composite.
5. It can be either plus or minus depending on lamina angle sequence.

The hypothesis that the normal edge load between laminae resulting from the plastic model was a governing factor in the fatigue life of laminates could be verified by testing specimens of different configurations having the same value of edge singularity.

III. Procedure

To test the hypothesis that interlaminar plastic shearing produces an edge singularity of normal load which govern interlaminar fatigue, the configurations of the laminates fabricated and tested had to have the same value for the edge singularity. If the specimens then delaminated in roughly the same number of stress cycles under the same stress load, the thermal induced singularity would be validated as a governing factor.

It was decided to use three difference configurations, one unbalanced and two balanced that differ only in the angle of the fiber. The fiber angle in each lamina differs by 90 degrees between the two balanced configurations. Each configuration was to have a total composite thickness of 0.066 inches. The unbalanced specimens were to have three unidirectional tape layers per lamina. The balanced specimens were to have seven layers total with two each for the outer laminae and three for the center lamina. The singularity calculated for the configurations were within 93.7% of each other (see calculation 1, appendix).

The specimens were to be fabricated from Scotchply, which is a 3M glass-epoxy lamina (see table 1, appendix). Scotchply used was to be unidirectional pre-impregnated glass tapes. It was chosen because it is easy to work with and it is inexpensive. The fabrication of specimen will be discussed later in this section.

In order to have shearing action working on the singularity calculated, it was decided to fatigue cycle the specimens in the bending mode. A static bending test was first to be performed. Then oscillating dynamic bending with zero mean load was to be performed on the specimen. The testing will be discussed later.

Fabrication of Specimens

The glass-epoxy used in this investigation was Scotchply Type 1001 pre-impregnated glass tape. To keep the tape from deteriorating during storage, the foot wide rolls were refrigerated. The fabrication of a sheet of composite was as follows:

1. The refrigerated tape was cut into the desired lengths for the composite sheet.
2. The hydraulic press which was used for the curing processes was preheated to 325°F. During the heat up, the mold used to shape the sheet was covered with Frekote releasing agent three times. The releasing agent allowed for easy removal of the cured composite sheet.
3. The strips of tape were carefully hand laid into the mold. At room temperature, the uncured tapes had a tendency to stick together.
4. The composite sheet was then cured in the hydraulic press at 325°F and 120 psi for thirty-five minutes.
5. The cured sheet was allowed to cool to room temperature in the press at zero psi. This minimized effects of thermal shock in the sheet.

After the cured sheet reached room temperature, the specimens were cut from the laminated plate by the use of a band saw. The specimens were next roughed to finished size by a hand file. The final dimensions were obtained by the use of emery paper. Machine sanding could not be used because it easily created localized heating which damages the epoxy material. The specimens were then nine and a half inches long and one inch wide with smooth straight edges. The free length of specimen after the grip pads were installed was three inches.

The grip pads were made from eleven layers of unidirectional tape. The grip edges towards the free length were tapered to reduce strain concentrations. Bonding of the grip pads to the specimens was accomplished with Eastman 910 strain gage adhesive (see Figure 3).

Testing

The testing was performed on a Baldwin SF-1 fatigue machine with a bending fixture of three inch moment arm. The machine has the capabilities of providing a static load. With bending fixture, the maximum static bending moment is 1500 inch pounds. The machine also can provide an oscillating load by means of a rotating mass. The oscillating load can be about a given static mean load and has a maximum bending moment of 1500 inch pounds. The testing fixture with specimen appears in Figure 4.

The fatigue damage was observed by a microscope that viewed the

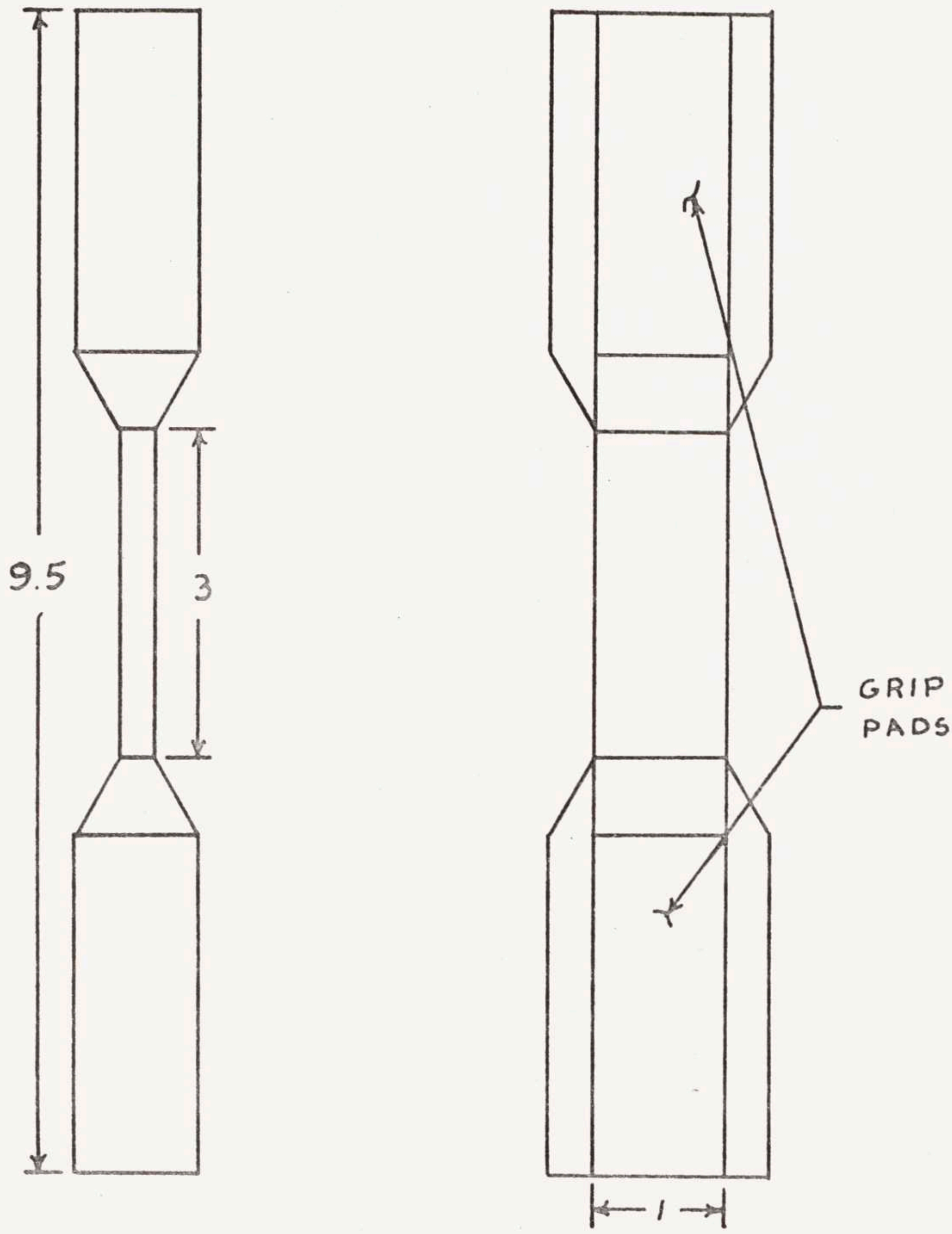

Fig. 3. Specimen Dimensions

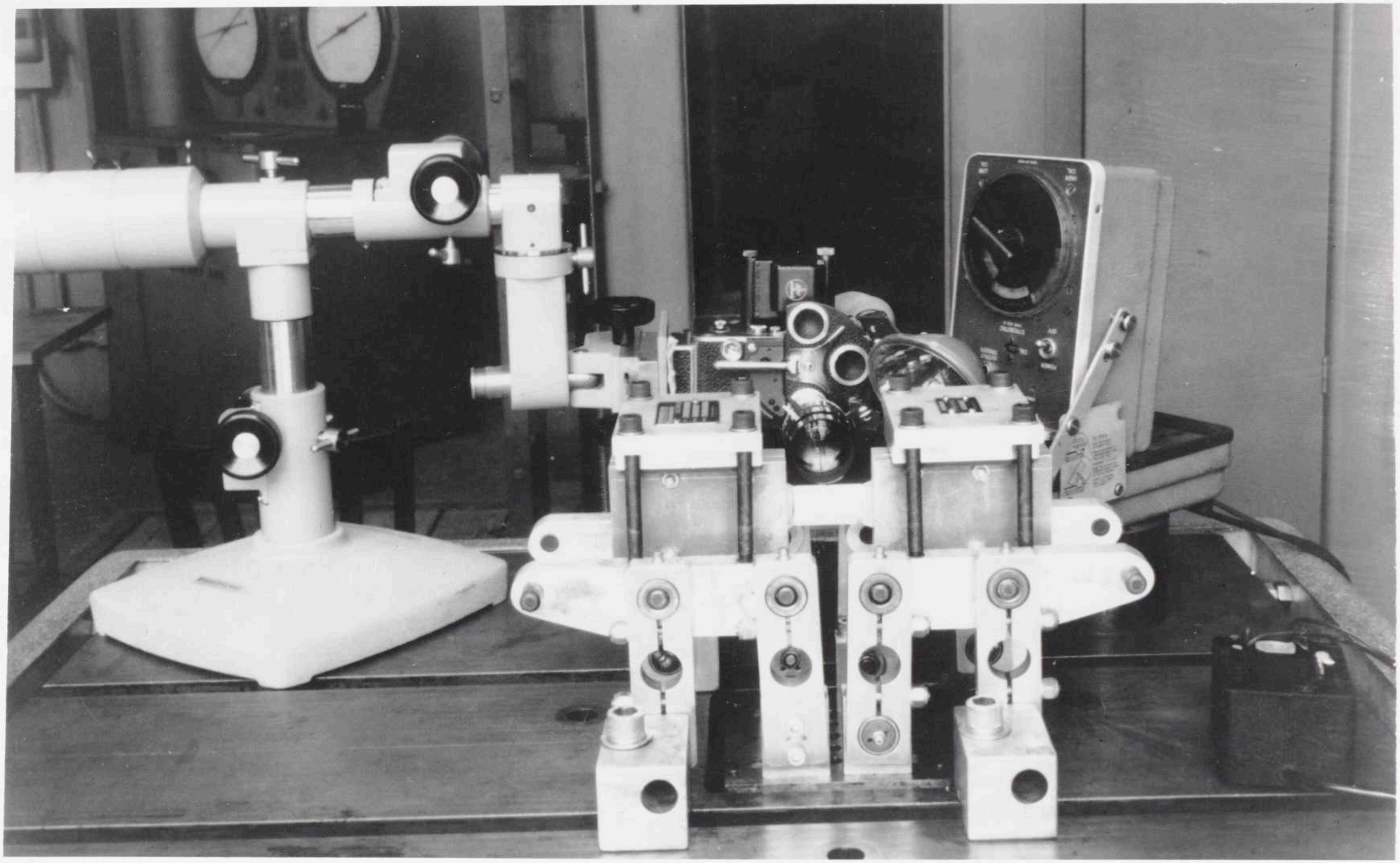

Fig. 4. Test Fixture

top edge of the specimen. The specimen was "stopped" during testing with the use of a stroboscope.

IV. Results

The test data for the static and dynamic experiments appear in Table 2. The static bending of the unbalanced specimens and balanced specimens with vertical outer fiber (VOF) showed that buckling occurred at stresses between 6.24×10^3 psi and 7.90×10^3 psi. A close examination of the edges of the composite revealed that cracks ran across the vertical laminae in both configurations. The crack density was lower when fewer surface imperfections were present. The balanced specimen with horizontal outer fiber (HOF) buckled at a much higher stress (5.73×10^4 psi), but the free length had been reduced to two inches to eliminate possible buckling problems. In spite of this, cracks that ran across the layer were found in the vertical lamina on the edge under tensile stress.

In the dynamic tests even though the stress levels were varied, all of the specimens had cracks across the vertical laminae. With the unbalanced and VOF specimens, the testing was terminated because cracks were not visible under the microscope during testing or wild vibrations or buckling occurred. Further observation revealed cracks across the vertical laminae and "L" type cracks that ran across the vertical lamina but in addition ran along the interface between the vertical and horizontal laminae (see Figure 5).

During testing of the HOF specimens, which had higher stress levels, cracks across the central lamina were observed with the microscope. The stress cycles at which each new crack appeared were recorded (see Table

Fig. 5. Typical "L" Type Crack

3). The number of cycles in the interval between observed cracks was within two times or one half the average number of cycles in the interval for each specimen. Later examination showed finer cracks across the vertical center lamina and "L" type cracks. The cracks in all the configurations appeared around surface imperfection of the order of 2.5×10^{-3} inches to 6.25×10^{-3} inches.

The cracks in the vertical laminae indicated that the predominant mode of failure is the tensile fatigue around the surface imperfections. The presence of interlaminar fatigue is noted by the "L" type of crack. The section of the crack that ran parallel to the interface of the laminates showed that delamination had occurred.

V. Conclusions

As a result of this investigation, the singularity modeled for the thermal induced stresses has not been validated as a governing factor in interlaminar fatigue. In the 0-90 angle-ply composite under bending, interlaminar fatigue was not the dominant fatigue process. Failure occurred under tensile mode of fatigue around the surface imperfections in the vertical laminae. The "L" type of crack indicated that some interlaminar fatigue was present.

Recommendations

The following are recommendations for further research in this field.

1. Experiments should be performed on specimens where the imperfections have been virtually eliminated.
2. The testing of the thermal induced singularity as a possible governing factor should be considered on fiber-reinforced composites of different ply angles.

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Appendix

Table 1

Scotchply Mechanical Properties

For Unidirectional Fibers

	Fiber Angle 0°	90°
Tensile Strength	160,000 psi	2900 psi
Flexure Modulus	5,300,000 psi	1,600,000 psi
Tensile Modulus	5.7×10^6 psi	1.4×10^6 psi
Compressive Strength (Edge)	90,000 psi	20,000 psi
Thermal Coefficient	4.8×10^{-6}/°F	12.3×10^{-6}/°F
Interlaminar Shear Strength	4300 psi	
Rockwell Hardness (M) Scale	100-108	
Thickness/ply	0.010"	
Percentage of Resin	40%	

Table 2

Summary of Experimental Data

No. of Specimen	Stress in psi	Stress Cycles	Comments
Unbalanced Specimens			
1	6240(static)	buckled	I cracks in vertical laminate*
2	7900(static)	buckled	I cracks in vertical laminate*
3	2370	160K	I and L cracks in vertical laminate
4	4100	unstable vibration during start-up	I and L cracks
5	4100	24K	I and L cracks in vertical laminate
Balanced Specimens VOF			
6	6820(static)	buckled	I cracks in vertical laminate
7	7900(static)	buckled	few cracks but fewer imperfections
8	4780	3K	wild buckling. I and L cracks in vertical laminate
9	3280	154K	I and L cracks in vertical laminates
10	4100	173K	I and L cracks in vertical laminates
11	4780 5450 6140 6820 8190	30K 70K 34K 10K buckled	I and L cracks in vertical laminates

*I cracks run across the laminate.

Table 2 (Cont.)

No. of Specimen	Stress in psi	Stress Cycles	Comments
Balanced Specimens			
HOF			
13	57,300	buckled	I cracks in vertical laminate on tensile edge of specimen
14	13,600	cracking heard during start-up I cracks in center laminate	
15	6820	235K	I cracks appeared in vertical laminate
16	9550	210K	throughout test
17	9550	478K	I and L cracks appeared in vertical laminate
18	10,900	250K	throughout test
19	6820	432K	" "

Table 3

Stress Cycles until Crack is Observed

Specimen NO.	Stress Cycles	Interval Between Cycles	NO. of Cracks Observed	
16	0		1	Average Cycles in Interval 46,000
	29,000	29,000	2	
	100,000	71,000	1	
	139,000	39,000	1	
17	0		1	Average Cycles in Interval 39,000
	17,000	17,000	1	
	36,000	19,000	1	
	92,000	56,000	1	
	136,000	44,000	2	
	180,000	44,000	1	
	237,000	57,000	1	
	274,000	37,000	1	
18	0		1	Average Cycles in Interval 55,000
	75,000	75,000	1	
	108,000	33,000	2	
	143,000	35,000	1	
	223,000	80,000	1	
19	0		1	

Table 3 (Cont.)

Specimen NO.	Stress Cycles	Interval Between Cycles	NO. of Cracks Observed	
	43,000	43,000	2	Average Cycles in Interval 44,000
	68,000	25,000	1	
	75,000	7,000	1	
	98,000	23,000	1	
	187,000	89,000	1	
	259,000	72,000	1	
	310,000	51,000	1	

Calculation 1

Calculation of the singularity for unbalanced and balanced specimens of Scotchply with a total composite thickness of 0.066 inches.

Case where $t_1 = t_2$ for unbalanced specimen and the singularity will be noted by β (6 layers).

$$\beta = -\frac{k}{2}\left(\frac{\frac{t_1}{E_1 I_1} - \frac{t_2}{E_2 I_2}}{\frac{1}{E_1 I_1} + \frac{1}{E_2 I_2}}\right)$$

if $t_1 = t_2$, $I_1 = I_2$

$$\beta = -\frac{k}{2}\left(\frac{E_2 - E_1}{E_2 + E_1}\right)t$$

$$\beta = -\frac{k}{2}\left(\frac{(1.6 \times 10^6 - 5.3 \times 10^6)\text{psi}}{(1.6 \times 10^6 + 5.3 \times 10^6)\text{psi}}\right)0.033 \text{ inch}$$

$$\beta = \frac{k}{2}(0.0177)\text{inch}$$

Case where balanced specimen has 7 layers (two each for the outer laminates and three for the center laminate).

$$\beta = t\,\frac{k}{2} = \frac{k}{2}\,0.0189$$

Ratio of the unbalanced to balanced singularity

$$\frac{0.0177\ k/2}{0.0189\ k/2} = 0.937$$

There is agreement within 93.7%.