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Compression Fatigue Analysis of Fiber Composites

M.M. Ratwani* and H.P. Kan†
Northrop Corporation, Hawthorne, Calif.

A macromechanics model, based on the delamination propagation between the plies of a composite laminate, has been developed for compression fatigue analysis of fiber composites. The model is based on the assumption that initial defects exist in composites between plies. These defects propagate due to the interlaminar stresses produced by applied fatigue loads. Existing compression fatigue data have been analyzed, using the model, and analytically predicted fatigue life compared with experimental data. Test data have been generated under constant amplitude loading on composite laminates with four different stacking sequences. Good correlations between experimentally observed fatigue data and analytical predictions have been found.

Introduction

THE fatigue characteristics of composite laminates under constant amplitude tension-tension, and tension-dominated spectrum loading have generally been extremely good, and they far exceed the comparable behavior of metals. A substantial amount of experimental work done in the past few years to study the fatigue behavior of composites under tension-compression and compression-compression loading[1-8] indicates that (contrary to the usual experience in metals) there is significant life reduction for composites when compared to tension-tension loading. The mechanism of failure has been observed to be local progressive fatigue failure of the matrix near a stress raiser resulting in fiber split and progressive delaminations leading to fiber buckling, which causes eventual laminate failure. These data indicate the compressive loading is an important design consideration and a detailed study of laminate behavior under such a loading is needed.

Tests[1] under compressive fatigue loads, both constant amplitude and spectrum, were conducted on NARMCO 5209 and Hercules AS/3501-6 prepreg systems using 1- and 2-in.-wide coupons with ¼-in.-diam holes. It was observed that the stacking sequence of the laminate has considerable effect on fatigue behavior of composites under compression loads. The use of angle plies on the surface of the laminate delayed the delamination, but the overall fatigue life of the laminate was unaffected. An extensive amount of fatigue testing[2,3] under tension-compression loading has been conducted on two different composite laminates. This work was mainly experimental using notched and unnotched specimens. Roderick and Whitcomb[4] have studied fatigue damage in (0, ± 45) and (0, ± 45, 90) boron/epoxy laminates with x-ray radiography and scanning electron microscopy. The major findings in their study were that interlaminar cracks occurred around the edge of the hole and propagated in the ± 45 deg plies. Eventually ± 45 deg fibers broke where the interlaminar cracks occurred. Axial fatigue of $[0/\pm 30]_{6s}$ laminate[5] under tension-tension, tension-compression, and compression-compression loading have shown that the fatigue effect is stronger (steeper S-N curve) under tension-compression loading than the tension-tension or compression-compression loadings.

In those studies, no attempts were made to develop analytical techniques for predicting compression fatigue behavior of composites. In view of the complex modes of failure observed in composites, and the almost unlimited geometries possible (fiber orientations and stacking order), there is a need for methodology to predict fatigue life and residual strength properties. The results abstracted from a broader study,[9] designed to develop and verify methodology for predicting compression fatigue life of composites, are discussed in this paper.

Delamination Propagation Model

A macromechanics model, based on the delamination between plies, is developed for fatigue life prediction. The model is based on the assumption that initial defects of macroscopic levels exist in the composite between plies. During fatigue loads, the defects along the free edge or around a stress raiser are opened by induced interlaminar shear and/or normal stresses. During fatigue cycling, these defects propagate causing progressive delamination of the composite. When the delamination reaches a critical size, buckling of a composite lamina or group of lamina separated from the total laminate will take place, causing the collapse of the structure.

Consider the composite laminate with a hole subjected to compressive fatigue loads, as shown in Fig. 1. If the load level is sufficiently large, delaminations will initiate between several plies. These delaminations generally initiate between plies in which the interlaminar shear stresses are maximum and interlaminar normal stresses are tensile. However, one of the delaminations will grow dominantly and govern the final failure of the laminate. This dominant delamination will generally be nearest to the surface of the laminate. The expected location of dominant delamination can be obtained by the analysis of interlaminar stresses. In Fig. 1, the dominant delamination is assumed to propagate between the first and second plies. It is assumed that there is a threshold stress below which delamination will not initiate or propagate. This phenomenon of threshold stress at which delamination does not propagate has been observed by Ryder and Walker.[2]

During fatigue loading the delamination propagates. When the length of delamination b, as shown in Fig. 1, reaches a critical value b_c, the outer lamina (first-ply in Fig. 1) will buckle during compression excursion of the load at minimum applied stress σ_{min} in the fatigue loading, and will result in the collapse of the panel. The final failure is due to buckling, and hence the size of the critical delamination will depend on the maximum compressive stress. The value of critical delamination b_c will be smaller for a larger compressive load. This is substantiated by the experimental results of Ryder and Walker[2] where it was observed that the size of delamination at failure was small for large compressive loads, and vice versa.

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*Senior Technical Specialist, Aircraft Division. Member AIAA.
†Engineering Specialist, Aircraft Division. Member AIAA.

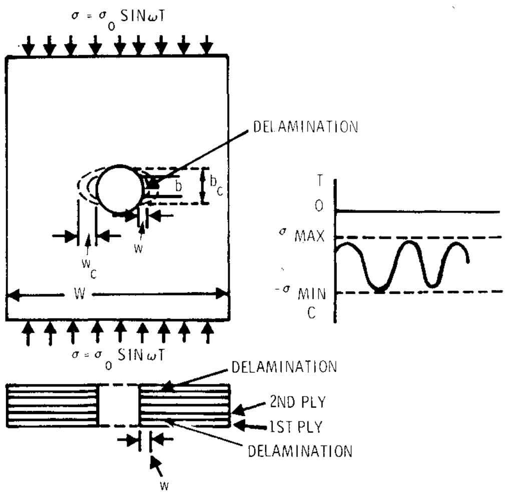

Fig. 1 Composite panel subjected to fatigue loads.

Delamination Propagation Model Under Interlaminar Shear Stresses

The delamination propagation model under interlaminar shear stresses may be expressed in a form similar to a crack propagation model in metals and is given by the equation

$$\frac{db}{dN}=c_I(\tau_{zma}-\tau_{zmi}-\tau_{th})^{n_I}b^{m_I} \tag{1}$$

where b is the length of the delamination, N the number of fatigue cycles, τ_{zmi} the minimum interlaminar shear stress, τ_{zma} the maximum interlaminar shear stress, τ_{th} the interlaminar threshold shear stress range below which delamination does not propagate, and c_I, n_I, and m_I are constants and will depend on the resin system and environmental conditions.

Let the initial defect be characterized by a delamination of size b_0, and the critical size at failure be b_c. Denoting the fatigue cycles to failure by N_f, integration of Eq. (1) yields

$$N_f=\frac{b_c^{1-m_I}-b_0^{1-m_I}}{c_I(\tau_{zma}-\tau_{zmi}-\tau_{th})^{n_I}(1-m_I)} \tag{2}$$

The value of b_c in Eq. (2) will depend on the minimum stress in fatigue loading.

Assuming the delamination in the composite to behave like a crack, the power m_I in Eq. (1) may be taken as $0.5\,n_I$. Thus, the fatigue life Eq. (2) reduces to

$$N_f=\frac{b_c^{1-0.5n_I}-b_0^{1-0.5n_I}}{c_I(\tau_{zma}-\tau_{zmi}-\tau_{th})^{n_I}(1-0.5n_I)} \tag{3}$$

The values of c_I, n_I, and τ_{th} in the preceding equations are to be obtained from experimental results. A simple buckling formula such as the Euler equation is used to determine the value of critical delamination size b_c.

Let n be the number of delaminated plies, and h be the thickness of each ply in the composite laminate. Using the Euler formula, the critical delamination size is obtained as

$$b_c=\pi nh\left(\frac{kE}{12\sigma_{\min}}\right)^{1/2} \tag{4}$$

where E is Young's modulus of delaminated plies in the loading direction, $\sigma_{\min}$ the minimum stress (compressive) in fatigue loading, and k is the constant which describes the end condition of the column.

The value of k depends on n the number of delaminated plies and is taken as 4 if outermost plies delaminate. It is taken as 2 if the plies somewhat away from the surface tend to delaminate, and is taken as unity if the delamination is at midthickness.

Currently, there are no data available on the initial delamination size b_0 in composites. In metallic structures, the initial flaw is generally assumed to be 0.13 mm (0.005 in.). A flaw size of 0.13 mm (0.005 in.) may be a reasonable assumption along straight machined edges in composites. Around holes and cutouts this initial flaw size is likely to be larger than 0.13 mm (0.005 in.). It may be noted that no generality is lost in the development of the fatigue model by the arbitrary assumption of initial flaw size, as long as some finite value is used.

Determination of Constants in Delamination Propagation Model Under Shear Stresses

The constants c_I, n_I, and τ_{th} in fatigue Eq. (3) depend on the resin system and the environment. These constants were obtained for Fiberite 934 and AS/3501-6 resins under room temperature dry conditions.

Fiberite 934 Resin

Compression fatigue data obtained by Ryder and Walker[2] on $(0/45/90/-45_2/90/45/0)_s$ laminate, using Fiberite 934 resin, was used to obtain the constants in the fatigue life prediction model. In this reference, all the fatigue tests were conducted at constant minimum (compressive) stress of 0, −68.9, and −110 MPa (0, −10, and −16 ksi). The maximum tensile stress was varied in the tests.

The interlaminar stresses of the specific laminate indicate that the interlaminar normal tensile stresses are small compared to the shear stresses and occur at locations different from those where shear stresses are maximum. Hence, the delamination will mainly propagate under the shear stresses and the delamination propagation model given by Eq. (1) can be used for these data. For a constant minimum stress, the value of critical delamination size b_c at failure is constant and fatigue life Eq. (3) reduces to

$$N_f=\frac{C_2}{(\tau_{zma}-\tau_{zmi}-\tau_{th})^{n_I}} \tag{5}$$

where C_2 is a constant.

The experimental results of Ryder showed no delaminations in the specimen tested at a maximum stress of 231 MPa (33.5 ksi) [interlaminar shear stress 51 MPa (7.4 ksi)] and a minimum stress of 0. Hence, for this resin system, the threshold value of shear stress is taken as 51.7 MPa (7.5 ksi). Using this value of threshold, the values of n_I for two data sets at 68.95 and 110.31 MPa (−10 and −16 ksi) were found to be 5.45 and 4.69, respectively, in Eq. (5). The average value of n_I for the two cases is 5.07. The value of $n_I=5$ is assumed for this resin system.

Assuming a threshold interlaminar shear stress of 51.7 MPa (7.5 ksi) and $n_I=5$, the value of $c_I=3.17\times10^{-5}$ was obtained in Eq. (3). Using these values of c_I and n_I in Eq. (3), the fatigue life of the laminate was predicted for the fatigue tests conducted at minimum stress of −68.9 and −110 MPa (−10 and −16 ksi). A comparison of test results and analytical predictions is shown in Fig. 2. It is seen that Eq. (3) describes the fatigue behavior well.

AS/3501-6 Resin

The constants for this resin system were obtained from the experimental data, obtained by Grimes,[10] on $(\pm45)_{6s}$ laminate. These data, obtained at $R(\sigma_{\min}/\sigma_{\max})=10.0$ are plotted in Fig. 3. The interlaminar stresses produced in this laminate are predominantly shear stresses, and the interlaminar normal stresses are very small. The delamination propagation in such a laminate will be essentially under interlaminar shear stresses. The stress analysis of this laminate

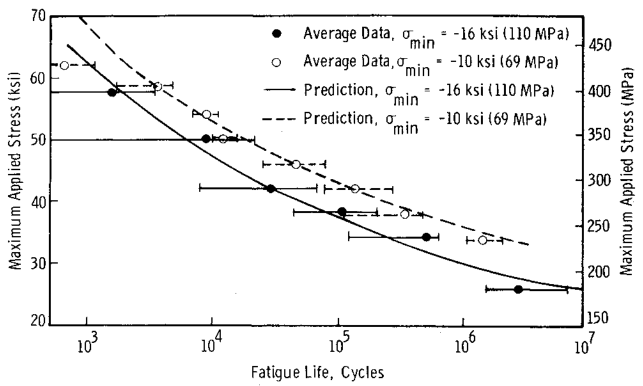

Fig. 2 Comparison of observed and predicted fatigue life of $(0/45/90/-45_2/90/45/0)_s$ laminate (Fiberite 934 resin).

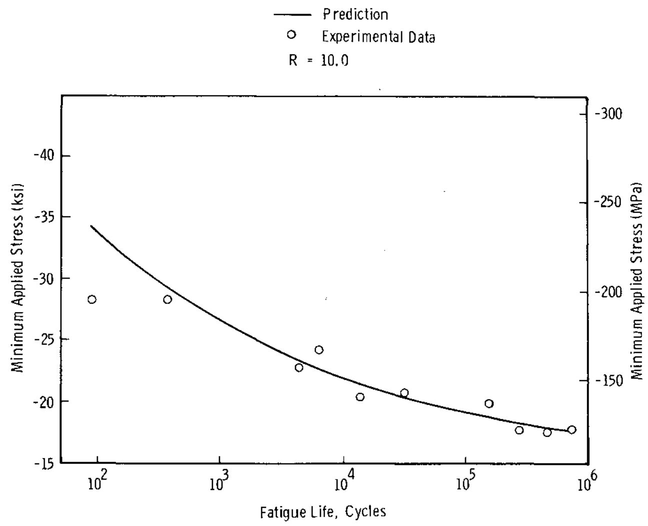

Fig. 3 Comparison of observed and predicted fatigue life of $(\pm 45)_{6s}$ laminate (AS/3501-6).

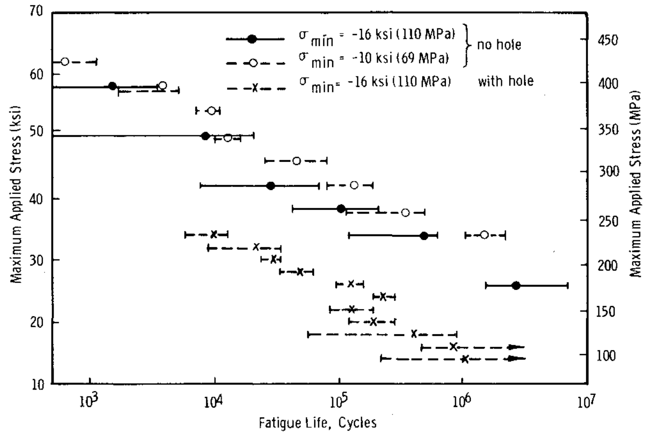

Fig. 4 Notched and unnotched fatigue data on $(0/45/90/-45_2/90/45/0)_s$ laminate (Fiberite 934 resin[2,3]).

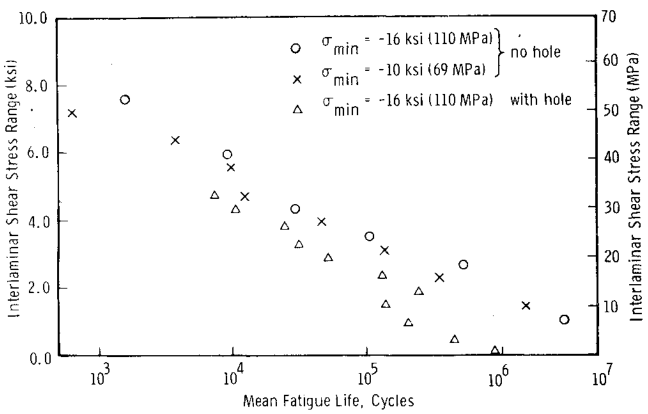

Fig. 5 Mean fatigue life vs interlaminar shear stress range $(0/45/90/-45_2/90/45/0)_s$ laminate (Fiberite 934 resin).

shows maximum interlaminar shear stress to be 0.5471 of the applied stress.

The experimental data of Grimes[10] showed no delaminations to propagate at interlaminar shear stress range of 50.88 MPa (7.38 ksi). Hence, for AS/3501-6 resin the threshold value of cyclic interlaminar shear stress range is taken as 48.3 MPa (7.0 ksi). The constants c_I and n_I in fatigue life Eq. (3) were obtained for the experimental data of Fig. 3 by least square fit. This gave the value of n_I to be about 5 and $c_I = 2.15 \times 10^{-4}$. The analytical predictions made with Eq. (3) are shown in Fig. 3. It is seen that Eq. (3) describes the fatigue behavior well.

Fatigue Life Prediction Model for Laminates with Predominantly Interlaminar Shear Stresses

The experimental fatigue data show the power n_I in the delamination propagation equation to be 5 for AS/3501-6 resin and Fiberite 934 resin. It is tentatively concluded that the value of $n_I = 5$ may be used for commonly used resins in composites. It may be noted that the value of n_I in crack propagation equations for metals is generally between 3 and 4. Using the value of $n_I = 5$ in Eq. (3), the fatigue life equation may be written as

$$N_f = C \frac{(1/b_0^{1.5}) - (1/b_c^{1.5})}{(\tau_{zma} - \tau_{zmi} - \tau_{th})^5} \tag{6}$$

where C is a constant. The value of the constant C is 3.1×10^3 for AS/3501-6 resin and is 2.1×10^4 for Fiberite 934 resin.

Verification of Analytical Models

The analytical fatigue life prediction model correlates fatigue life with interlaminar stresses. In order to predict fatigue life, interlaminar stresses have to be determined. The NASTRAN finite-element computer program was used to determine such interlaminar stresses in various laminates.

Verification of Correlation Between Fatigue Life and Interlaminar Stresses

The compression fatigue data, obtained from various sources, were plotted as a function of interlaminar shear stress range, to verify the correlation between fatigue life and interlaminar stresses. The tension-compression fatigue data, obtained on notched and unnotched specimens[2,3] (using Thornel T300 fiber/fiberite 934 resin), are plotted as a function of maximum tensile stress in Fig. 4. It is seen that the fatigue life of laminates with holes is considerably lower than that of laminates without holes. These data have been plotted in Fig. 5 as a function of the interlaminar shear stress range. It is seen that all the data for the Fiberite 934 resin lie within the scatterband expected in composites.

The fatigue data[1,10] on the AS/3501-6 resin system have been analyzed. These data have been obtained on laminates with holes and without holes at different values of R (minimum to maximum stress) ratios and on specimens with different width W. The interlaminar stress analysis of all the laminates was carried out, and the fatigue data have been plotted as a function of interlaminar shear stress range in Fig. 6. It is seen that the fatigue data lie within the scatterband expected in composites. The scatter is large at high stress levels due to the influence of interlaminar normal stresses.

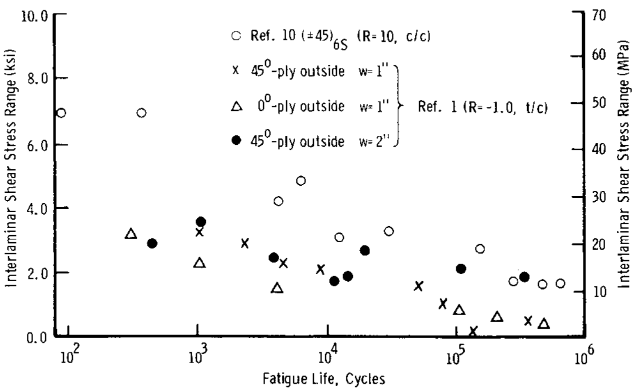

Fig. 6 Fatigue life vs interlaminar shear stress range for laminates using AS/3501-6 resin system.

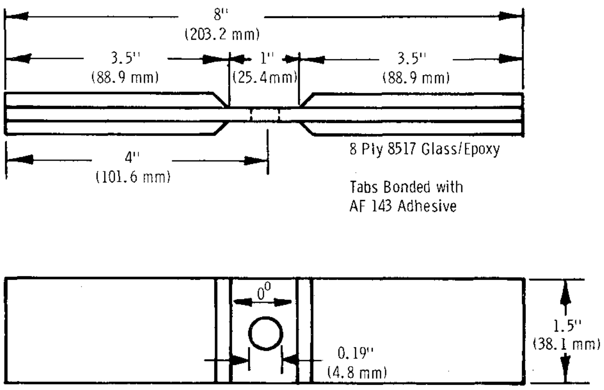

Fig. 7 Test specimen geometry.

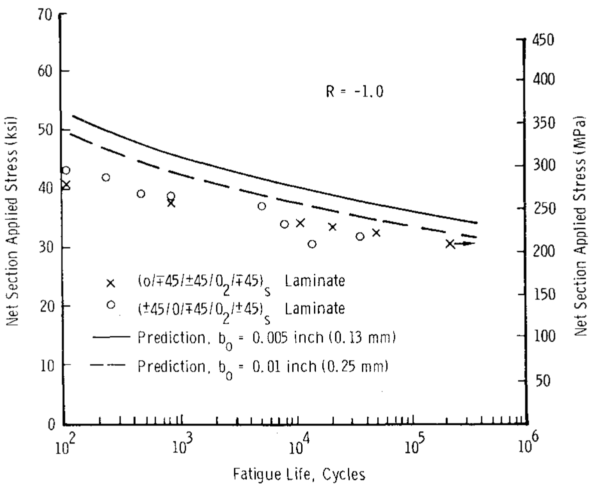

Fig. 8 Comparison of observed and predicted fatigue life of laminates tested in Ref. 1.

Verification of Life Prediction Model

The fatigue life prediction model Eq. (6) has been verified with the data available in the literature and experimental data generated in the present studies.

Verification of Model with Existing Data

The fatigue life of $(0/\mp 45/\pm 45/0_2/\mp 45)_s$ and $(\pm 45/0/\mp 45/0_2/\pm 45)_s$ laminates, tested by Rosenfeld and Huang, was predicted using Eq. (6). The experimental data had been obtained on AS/3501-6 resin using the specimen configuration shown in Fig. 7.

The interlaminar stress analysis of the $(0/\mp 45/\pm 45/0_2/\mp 45)_s$ laminate indicates the dominant interlaminar shear stress to be 0.1534 of the applied stress between 0 and -45 deg plies near the surfaces.[9] The maximum interlaminar shear stress in the $(\pm 45/0/\mp 45/0_2/\pm 45)_s$ laminate occurs between $+45$ and -45 deg plies and is 0.1582 of the applied stress. A comparison of observed and predicted fatigue life of the two laminates is shown in Fig. 8. The fatigue life is predicted assuming interlaminar shear stress 0.1534 of applied stress and initial flaw sizes of 0.13 and 0.25 mm (0.005 and 0.01 in.). The predicted life, assuming an initial flaw of 0.25 mm (0.01 in.) agrees reasonably well with experimentally observed life.

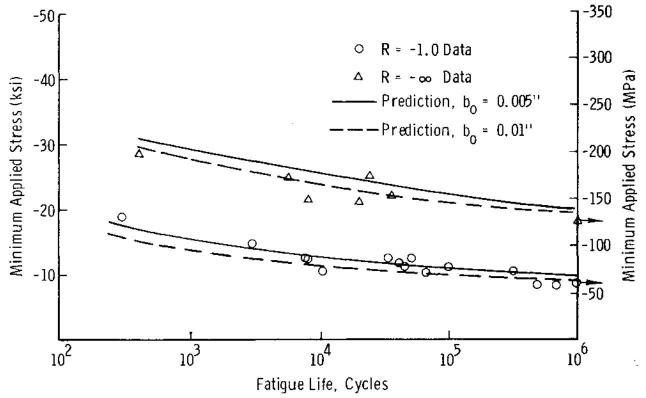

Fig. 9 Comparison of observed and predicted fatigue life of $(\pm 45/90_2/\pm 45/90_2)_s$ laminate (AS/3501-6 resin, $R = -1.0$, $-\infty$).

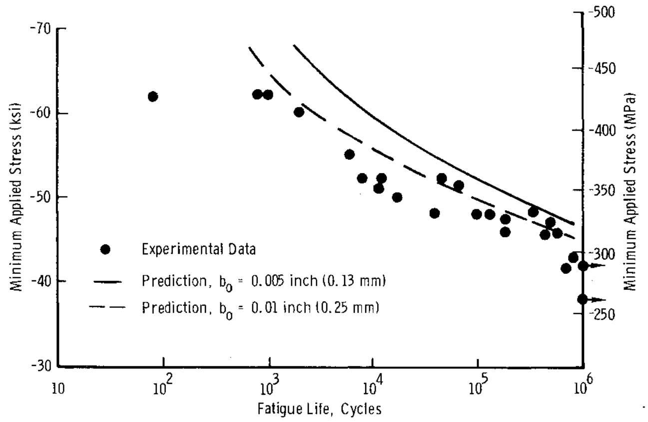

Fig. 10 Comparison of observed and predicted fatigue life of $(0/\pm 45/90/0_2/\pm 45)_s$ laminate (AS/3501-6 resin, $R = -1.0$).

The fatigue life prediction model was also verified with the experimental data on the $(\pm 45/90_2/\pm 45/90_2)_s$ laminate, obtained independently in Northrop research program.[11]

These data have been obtained on specimens shown in Fig. 7, using AS/3501-6 resin, under constant amplitude loading at $R = -\infty$ and -1.0. The interlaminar stress analysis of the laminate shows the maximum interlaminar shear stress to be 0.4239 of applied stress. A comparison of experimental and predicted fatigue life, assuming initial flaw sizes of 0.13 and 0.25 mm (0.005 and 0.01 in.) is shown in Fig. 9. The agreement between analytical predictions and experimental results is very good.

Verification of Fatigue Model with Data Obtained in Navy Contract[9]

Compression fatigue data were obtained on AS/3501-6 resin laminates with four different stacking sequences, namely $(0/\pm 45/90/0_2/\pm 45)_s$, $(90/\pm 45/0_3/\pm 45)_s$, $(90/0/\pm 45/0_2/\pm 45)_s$, and $(\pm 45/0/90/0_2/\pm 45)_s$. The specimen geometry shown in Fig. 7 was used in the test program. The fatigue life of these laminates was predicted with Eq. (6) using constants for the AS/3501-6 resin and initial flaw sizes of 0.13 and 0.25 mm (0.005 and 0.01 in.). Figure 10 shows a comparison of

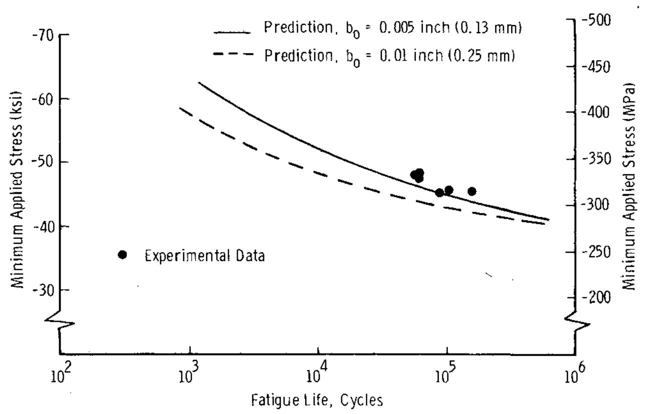

Fig. 11 Comparison of observed and predicted fatigue life of $(90/\pm45/0_3/\pm45)_s$ laminate (AS/3501-6 resin, $R = -1.0$).

observed and predicted fatigue life for $(0/\pm45/90/0_2/\pm45)_s$ laminate, assuming two initial flaw sizes. It is seen that comparison between predicted fatigue life and experimentally observed life is good.

The comparison of observed and predicted fatigue life for $(90/\pm45/0_3/\pm45)_s$ laminate is shown in Fig. 11. The predicted fatigue life, assuming initial flaw of 0.13 mm (0.005 in.) agrees well with experimental results.

A good correlation between predicted and observed fatigue life was obtained for other laminates.[9]

Conclusions

This paper presents an analytical model for compression fatigue analysis of composite laminates. Based on the preceding studies, the following conclusions are reached.

1) The compression fatigue life of a composite laminate can be related to interlaminar stresses produced in the laminates under external loading.

2) The delamination propagation model can be used to predict compression fatigue life.

Acknowledgments

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