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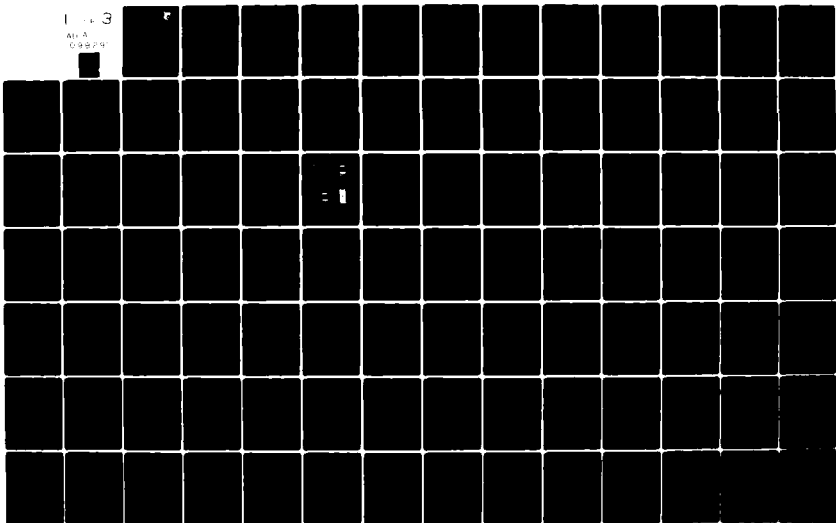
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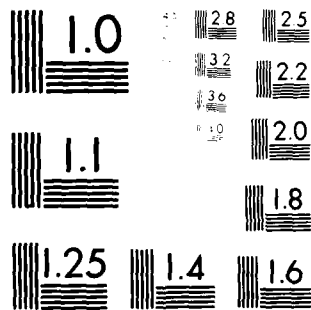
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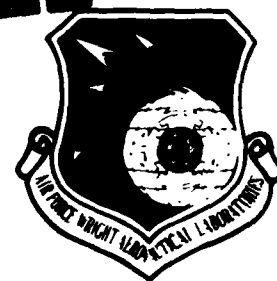




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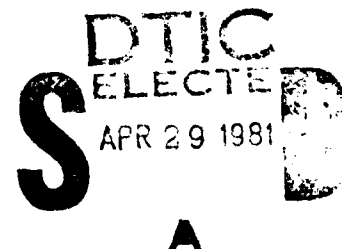
Marvin Knight
Mechanics & Surface Interactions Branch
Nonmetallic Materials Division

February 1981

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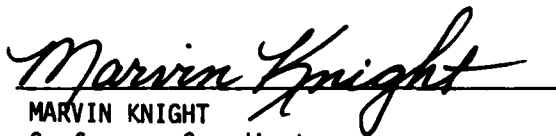
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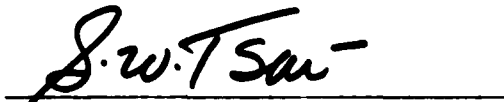
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This technical report has been reviewed and is approved for publication.



MARVIN KNIGHT
Conference Coordinator
Sixth Annual Mechanics of Composites
Review



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FOR THE COMMANDER



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20. ABSTRACT (Continue on reverse side if necessary and identify by block number) This report contains summaries of the presentations at the Sixth Annual Mechanics of Composites Review sponsored by the Air Force Materials Laboratory. Each paper was prepared by its presenter and is published here unedited. In addition to the presenter's summaries, a listing of both the inhouse and contractual activities of each participating organization is included.			

FOREWORD

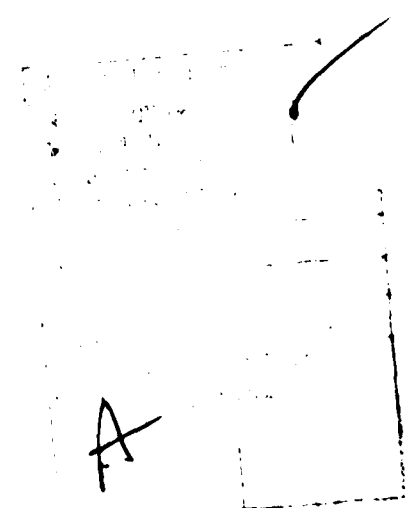
This report contains summaries of the presentations at the Sixth Mechanics of Composites Review sponsored by the Materials Laboratory. Each summary was prepared by its presenter and is published here unedited. In addition to the presenters' summaries, a listing of both the in-house and contractual activities of each participating organization is included.

The Mechanics of Composites Review is designed to present programs covering activities throughout DOD and NASA. Programs not covered in the present review are candidates for presentation at future mechanics of composites reviews. The presentations cover both in-house and contract programs under the sponsorship of the participating organizations.

Since this is a review of on-going programs, much of the information in this report has not been published as yet and is subject to change; but timely dissemination of the rapidly expanding technology of advanced composites is deemed highly desirable. Works in the area of mechanics of composites have long been typified by disciplined approaches. It is hoped that such a high standard of rigor is reflected in the majority, if not all, of the presentations in this report.

Feedback and open critique of the presentations and the review itself are most welcome as suggestions and recommendations from all participants will be considered in the planning of future reviews.

We express our appreciation to the authors for the contribution of their summaries and to the points of contact within the organizations for their effort in supplying the program listings.



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28-30 OCTOBER 1980

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INFLUENCE OF FREQUENCY AND ENVIRONMENTAL CONDITIONS
ON DYNAMIC BEHAVIOR OF GRAPHITE/EPOXY COMPOSITES

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This research has three objectives. The first is to establish a data base to facilitate confident use of graphite/epoxy composites in dynamic applications. The second is to determine the extent to which viscoelastic factors influence dynamic behavior. This is reflected in the frequency dependence of response to time dependent excitation. The third objective is to determine the effects of moisture absorption and elevated temperature on dynamic behavior over a wide range of exciting frequencies.

These objectives have been accomplished by performing flexural vibration tests on graphite/epoxy beams. Dynamic behavior in the dry, room temperature state 25°C (77°F) is contrasted with the following four elevated temperature states:

- a. 82°C (180°F), dry
- b. 60°C (140°F), moisture saturated
- c. 82°C (180°F), moisture saturated
- d. 93°C (200°F), moisture saturated

The properties determined are an effective flexural modulus and damping for frequencies from 10-1000 Hertz.

The specimens were manufactured by McDonnell Douglas Astronautics Company - St. Louis from Narmco 5208/T300 unidirectional tape. Four distinct layup configurations have been tested as beams: (0), (+45), (0, +45, 90, -45, 0, +45), and (90). They are each 12 plies thick and symmetric. The layup configurations are denoted A, B, C and D, respectively.

Considerable attention must be given to test conditions and testing technique. Since environmental effects are to be determined, vibration testing in a vacuum chamber at room temperature --- the usual means of determining damping --- could be used only for the reference state. All other tests have been performed in an environmental chamber at the proper temperature. Moisture saturation is achieved by immersion in a constant-temperature water bath for an extended period. Moisture absorption is monitored by periodically weighing the specimens.

The beams have been tested in cantilever fashion using electro-magnetic noncontacting transducers and exciters. The tests are resonance tests. The natural frequency is varied by means of three approaches: mass addition, excitation of higher modes, and variation of unsupported beam length. This

permits data in the 10-1000 Hertz range to be obtained. Small amplitude excitation is used exclusively in the elevated temperature tests to reduce aerodynamic damping effects to a minimum. Reliable aerodynamic damping estimates suggest that this influence is ignorable for the test conditions adopted. Damping is determined by suspending excitation and observing the decay of the response.

A succinct presentation of the major findings appears in Tables 1 and 2. In Table 1, values of an effective dynamic modulus in simple flexure that have been averaged over the frequency range of 10-1000 Hz are presented for each environmental condition and specimen type. This data describes the overall tendency of the different environmental conditions to affect stiffness. Below each of these entries in the table are values in parentheses which are percent variations over the 10-1000 Hz frequency range. These are direct indicators of frequency dependence. All values in Table 1 have been determined by establishing the best least squares fit to the data and then using the properties of the fitted curve to establish table entry values.

A and C specimens respond in a fiber controlled mode of behavior. Both environment and frequency have little effect on their stiffness. B and D specimens, however, exhibit matrix controlled behavior. Naturally, the response of these specimens is more sensitive to both environment and frequency. This is reflected in the data in Table 1.

Damping information is presented in Table 2. Average damping coefficient values for low frequency vibration are given. These values are obtained from tests in the fundamental mode of vibration, which always correspond to frequencies less than 150 Hz. This is the range of greatest practical interest. Within it, frequency effects are small in all cases. Again, A and C specimen behavior is not greatly influenced by environmental conditioning. B specimen damping is affected by moisture, but very little by temperature. The damping of saturated D specimens increases sharply from 82C to 93C.

Overall, the damping results indicate that frequency effects are quite small in all cases. They are a bit greater for matrix controlled modes of response exhibited by the B and D specimens at the higher frequencies. At the same temperature, damping increases with moisture saturation. For dry specimens, however, by contrast, damping decreases with temperature.

TABLE 1. SUMMARY OF DYNAMIC MODULUS INFORMATION

Specimen Type	Average Dynamic Modulus (GN/m ²) (Percent Variation Over the Range 10-1000 HZ)				
	25C (77F) Dry	82C (180F) Dry	60C (140F) Wet	82C (180F) Wet	93C (200F) Wet
A	101.5 (6.03)	99.7 (1.65)	98.1 (0.30)	98.9 (0.54)	100.0 (1.22)
B	20.3 (6.75)	18.4 (15.10)	18.2 (20.72)	19.0 (6.13)	16.9 (21.50)
C	64.1 (2.95)	62.8 (3.05)	63.5 (0.68)	63.5 (0.17)	62.2 (1.48)
D	7.8 (18.56)	7.2 (12.92)	7.0 (1.61)	6.3 (2.82)	5.0 (18.86)

TABLE 2. AVERAGE DAMPING COEFFICIENT FOR LOW FREQUENCIES

Specimen Type	Damping Coefficient (Percent of Critical Value)				
	25C (77F) Dry	82C (180F) Dry	60C (140F) Wet	82C (180F) Wet	93C (200F) Wet
A	0.054	0.052	0.074	0.076	0.070
B	0.478	0.478	0.567	0.559	0.563
C	0.109	0.098	0.108	0.1200	0.117
D	0.595	0.502	0.600	0.605	0.764

COMPOSITE MATERIALS FOR STRUCTURAL DESIGN

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Activities at Texas A&M University related to a research project on composite materials sponsored by the Air Force Office of Scientific Research (Contract No. F49620-78-C-0034) are summarized in Table 1. Shown are the elements of the research project as well as the involvement of graduate students and industry; some undergraduates participate by assisting in the laboratory and/or taking related coursework.

The graduate students serve as research assistants on the project, with their work leading to an M. S. Thesis or Ph. D. Dissertation. Thirteen M. S. students participated during the first two years of the project, and their thesis titles are listed in Table 2. Beginning September 1980, there are seven new M. S. students and one Ph. D. student.

The thesis titles provide an indication of the range of topics under study. Faculty direct the thesis research and are also involved in other theoretical and experimental research efforts, where this latter effort often provides a basis for the individual student projects.

Faculty participating in the research project are: Walter L. Bradley (Mechanical Engineering); Walter E. Haisler (Aerospace Engineering); Joe S. Ham (Physics); Kenneth L. Jerina (Civil Engineering); Richard A. Schapery- P. I. (Civil and Aerospace Engineering); Jack Weitsman (Civil Engineering).

Results from several of the studies will be reviewed at the meeting. In this report, because of space limitations, we shall review results from only one area - viscoelastic fracture mechanics. Figures 1 and 2 illustrate the relation between energy release rate and delamination crack speed in unidirectional graphite/epoxy and glass/epoxy composites; the specimen and loading is shown schematically in Fig. 1. The speed dependence is believed to be due primarily to dissipative processes in the material close to the crack tip; over most of the length of the split laminate the response is elastic as the fibers are parallel to the beam axis (direction of crack propagation).

The energy release rate increases with crack speed for glass/epoxy, whereas the opposite behavior is found for graphite/epoxy in many cases (cf. Fig. 1). The energy release rate increases with increasing moisture and temperature at high levels (cf. Fig. 2), which is theoretically consistent with the negative slope of $G(\dot{a})$. The negative slope is predicted to cause unstable (stop-start) crack growth on the micro-scale, and fracture surface features seem to confirm this. The theory correctly predicts a region of negative slope for the graphite composite (cf. Fig. 3). More detailed studies are underway to gain a better understanding of fracture and possible toughening mechanisms. As part of this effort a non-linear viscoelastic fracture mechanics theory based on a generalized J-integral has been developed for both initiation and growth of cracks.

TABLE 1

COMPOSITE MATERIALS PROGRAM

AT

TEXAS A&M UNIVERSITY

INTEGRATED RESEARCH PROGRAM ON ADVANCED COMPOSITES AND RESINS

- Processing
Effect of cure cycle parameters on resin and laminate properties
- Testing
Deformation and fracture characteristics at different temperatures and humidities versus processing parameters and time; fractography
- Theoretical Modelling and Analysis
Micro- and macro-mechanisms of deformation and fracture; cure cycle optimization; lamina and laminate response versus damage state and time
- Applications (Research-Design Interaction)
Projects for some M. S. Theses based on structural design problems proposed by industry

GRADUATE STUDENT ACTIVITIES

Twelve-Month Academic and Research Program in Composite Materials Specialty Leading to a Master of Science Degree in Engineering.

Doctoral Level Program also available.

- Research on AFOSR Contract
- Course work in mechanics, materials science, and mathematics, and in the testing, analysis and design of composites. Seminar

INDUSTRY PARTICIPATION

- Aircraft company employees participate as graduate students
- Provide lecturers for composite materials seminar
- Provide instructors for course on designing with composites
- Discuss research and design projects with students and faculty

TABLE 2

MASTER OF SCIENCE THESES

First Year (Completed August 1979)

1. Delamination Fracture Toughness of a Unidirectional Composite
2. Stresses Due to Environmental Conditioning of Cross-Ply Graphite/Epoxy Laminates
3. Experimental Investigation of Moisture and Temperature Conditioning of T300/5208 Graphite/Epoxy Composite Material
4. Determination of the Relationship of Free Volume to Mechanical Behavior for an Epoxy System Subjected to Various Aging Histories
5. Aerolastic Tailoring of Composite Materials

Second Year

1. Residual Thermal Stress in an Unsymmetrical Cross-Ply Graphite/Epoxy Laminate
2. Delamination Fracture Toughness of a Unidirectional Graphite/Epoxy Composite
3. Nonlinear Viscoelastic Characterization of AS-3502 Graphite/Epoxy Composite Material
4. An Investigation of Intra-ply Microcrack Density Development in a Cross-Ply Laminate
5. Moisture and Temperature Effects on Curvature of Anti-Symmetric Cross-Ply Graphite/Epoxy Laminates
6. Stress and Energy Analysis of Cross-Ply Laminates with Microcracks
7. Experimental Investigation of Free Volume Concepts in Relationship to Mechanical Behavior of an Epoxy System Subjected to Various Aging Histories
8. Application of Energy Release Rate Principles to Compression Debonding

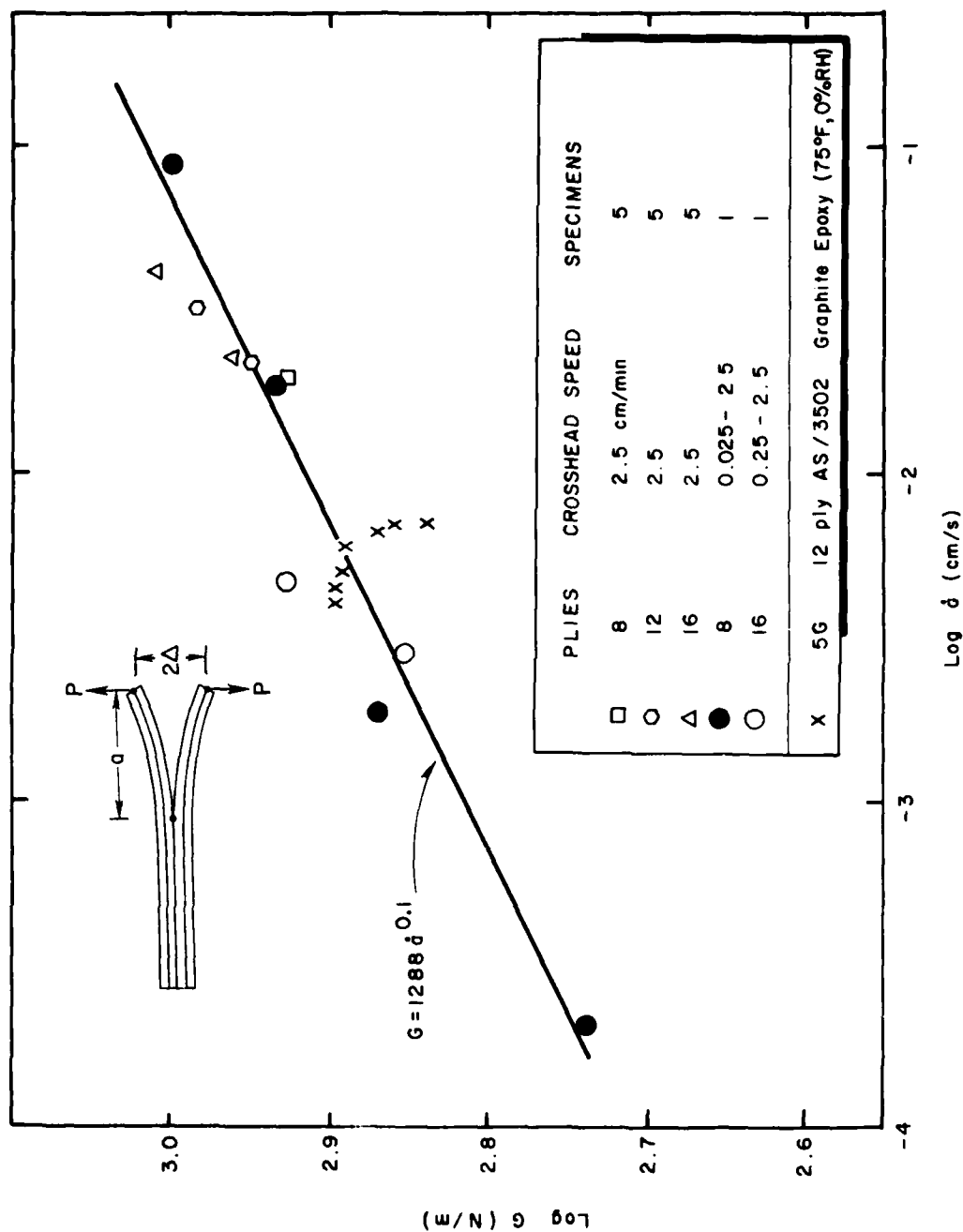


Figure 1. Energy release rate for Scotchply (1002) versus delamination crack speed (dry, 75°F). Graphite/epoxy data plotted for comparison; note that the actual G is one-fifth that shown.

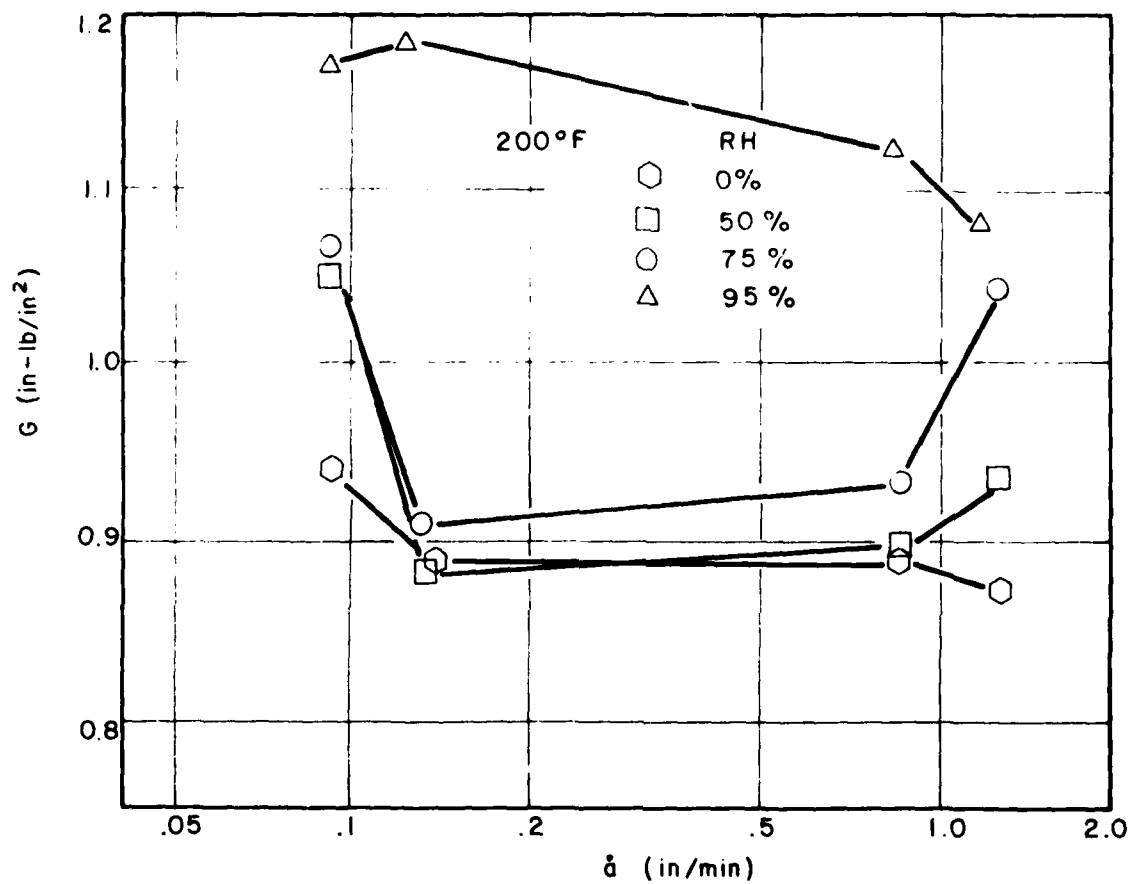


Figure 2. Energy release rate versus delamination crack speed for graphite/epoxy (AS/3502) in various environments (saturated state).

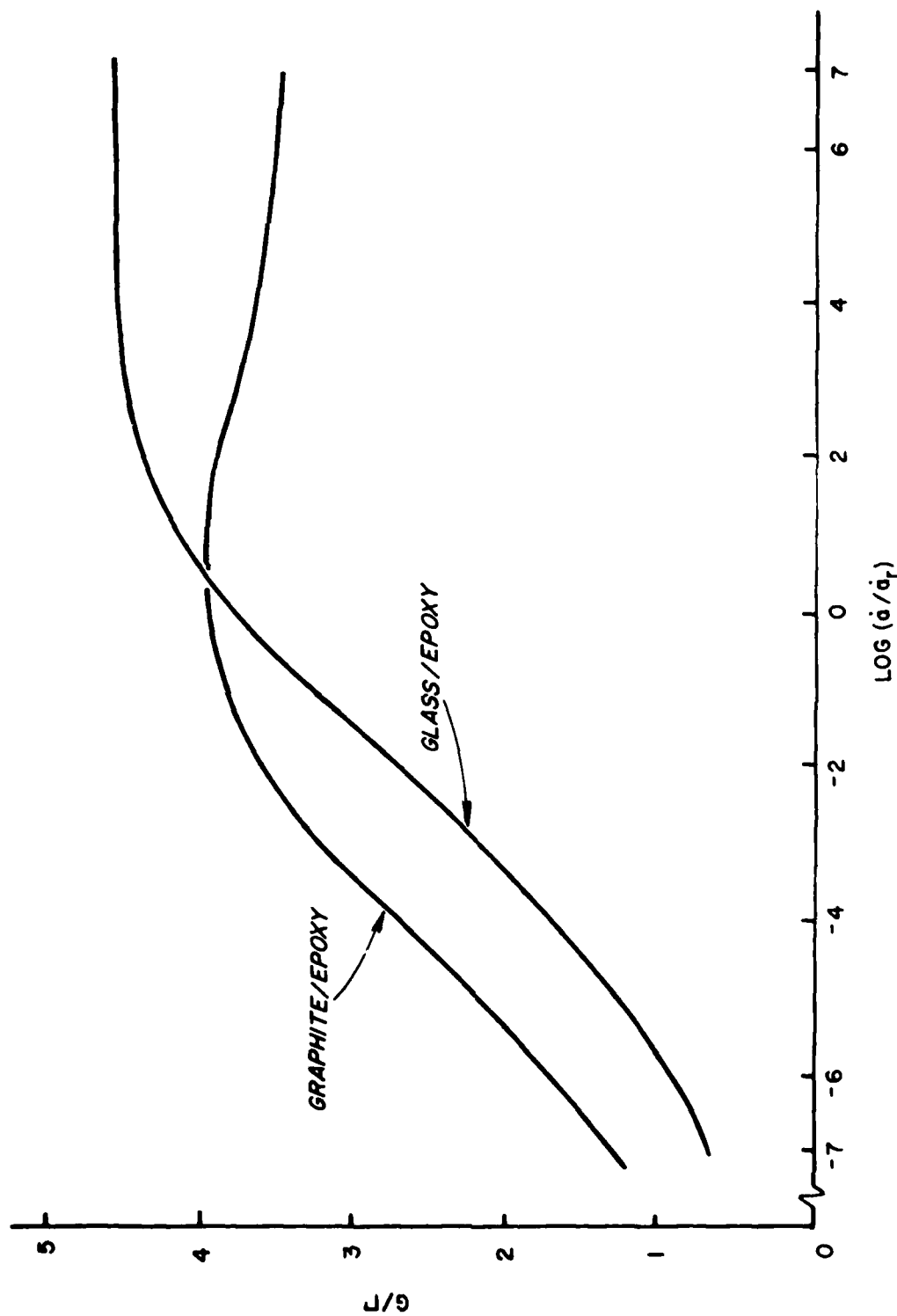


Figure 3. Theoretical prediction of normalized energy release rate versus normalized delamination crack speed based on representative fiber/matrix properties; the same matrix properties used for both composites.

SPACE ENVIRONMENTAL EFFECTS ON POLYMER MATRIX COMPOSITE MATERIALS

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PROGRAM OBJECTIVES

- (1) Determine the effect of space environment on the mechanical and physical properties of polymer matrix composite materials as a function of exposure time by means of in-situ laboratory simulation.
- (2) Evaluate the effects of ground-based accelerated testing in space simulators pertaining to thermal cycling and radiation damage.
- (3) Obtain test data from satellite composite materials experiments for comparison with (1) and (2).

An extensive program has been undertaken to study the effects of space environmental exposure on polymer matrix composites. At present, four space simulators are being used to evaluate composite materials in-situ and obtain test data on strength, stiffness, CTE and damping parameters. Analytical comparisons are then derived based on measured 'principal' values. Materials which have been investigated throughout the whole program to-date include graphite/epoxy (3M SP 288 T300), Kevlar /epoxy (3M SP 306, PRD-49-III) and boron/epoxy (3M SP 290).

DESCRIPTION OF TEST FACILITIES

Four thermal-vacuum chambers have been assembled capable of achieving 10^{-6} ~ 10^{-7} torr. A short term system for rapid thermal cycling is available with in-situ mechanical loading (axial and torsion). Another long term chamber with a mass spectrometer attachment exists for investigating large numbers of composite samples thermally cycled over periods of a year or more. Currently this facility is being modified to include in-situ tension loading as well. For thermal distortion studies, a laser interferometer thermal-vacuum system has been used to provide comparative test data for specimens containing surface bonded strain gauges. To complement these simulators is a thermal-vacuum chamber with U.V. (~ 1 solar constant) and electron radiation (Sr90). In-situ tension loading of up to 30 flat samples mounted on a rotating carousel is also available for stiffness, strength and creep tests.

TEST RESULTS AND COMPARISONS WITH ANALYSIS

To illustrate the data obtained for both short and long term exposure to vacuum for varying numbers of thermal cycles ($75^{\circ}\text{F} \leq T \leq 200^{\circ}\text{F}$), we shall

only consider coefficient of thermal expansion (CTE) and material damping (log decrement).

Figures 1 and 2 present CTE results for various $(\pm \theta)_s$ configurations for graphite/epoxy and Kevlar /epoxy, respectively. Note that for some orientations, the effect of thermal-vacuum cycling results in 'drift' in the CTE. Figures 3 and 4 demonstrate the difference between ambient and vacuum states on the CTE response. Using the principal CTE values (i.e., at 0° and 90°), one can calculate the CTE for any other laminate configuration. Good agreement has been obtained for several cases studied to-date.

Flexural damping measurements have also been made on cantilever plate specimens. Early in the program it was found that 'batch' and cure cycle variations led to substantial differences in the damping values. Figure 5 illustrates this problem in terms of the principal damping parameters. In addition, it was observed that short term vacuum exposure also produced changes in damping due to outgassing. Using measured principal values, the curves of Figure 6 were obtained for no aerodynamic effects (i.e., soft vacuum) and 24 hour, 72 hour exposure to 10^{-6} torr. However, it is of interest to note that once the principal damping terms are known, reasonably good agreement can be achieved between predictions and test data for samples prepared at the same time under identical conditions (see Figure 7).

The effect of prolonged hard vacuum with cumulative thermal cycling ($75^\circ\text{F} \leq T < 200^\circ\text{F}$) is shown in Figure 8 based on test samples subjected to 470 days at $10^{-6} \sim 10^{-7}$ torr. Substantial reductions in damping occurred when compared to 'identical' coupons that were stored in a desiccator for the same time period.

Finally, to demonstrate the application of principal damping measurements, one can calculate the log decrement for a quasi-isotropic laminate for varying ply positions (see Figure 9). Such results can be used to optimize laminate damping.

CONCLUSIONS

(a) In the presence of hard vacuum, the following effects have been observed (primarily due to moisture outgassing) on the behavior of thin epoxy matrix composites:

- (i) matrix strength and stiffness increase over a wide temperature range;
- (ii) significant changes in CTE occur, the magnitude of which depends on the laminate configuration;
- (iii) reductions in the specific damping capacity (or log decrement) occur, the magnitude of which depends on the laminate configuration.

(b) The combined effects of cumulative thermal cycling in a hard vacuum over a sufficient period of time to ensure moisture outgassing result in the following changes in thin epoxy matrix composites:

- (i) substantial reductions occur in matrix strength and stiffness;

(ii) continual drift (i.e., changes) in CTE occurs with increasing numbers of thermal cycles, at least up to ~109 which have been attained in this program (the 'asymptote' has not yet been reached).

(iii) substantial reductions in material damping (log decrement) occur.

(c) Laminate models are quite accurate in predicting the CTE and specific damping values for arbitrary laminates under ambient and thermal-vacuum conditions, if the principal values are known for the particular state being studied.

(d) Accurate predictions of laminate damping values can only be obtained if test coupons are taken from the same material batch and subjected to the identical cure cycle as the laminate.

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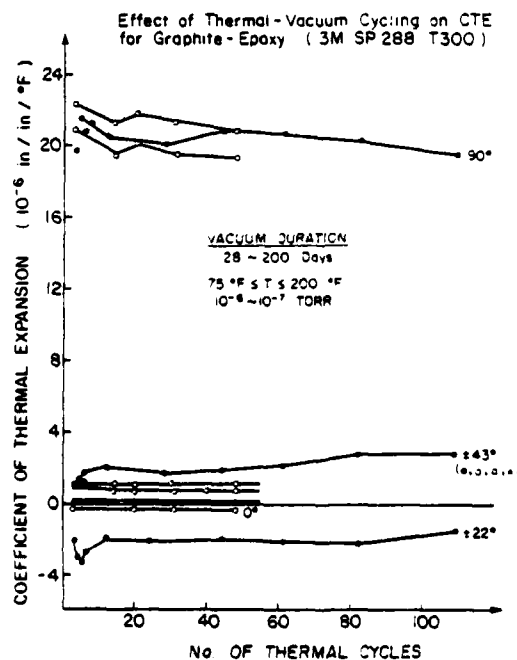


FIG. 1

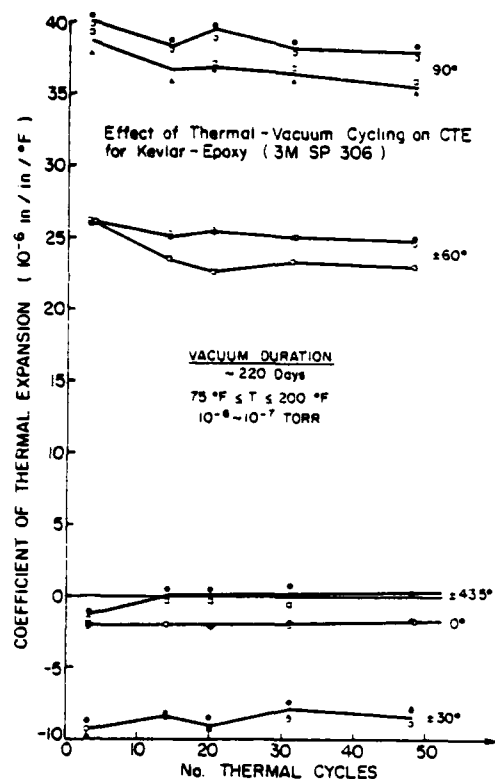


FIG. 2

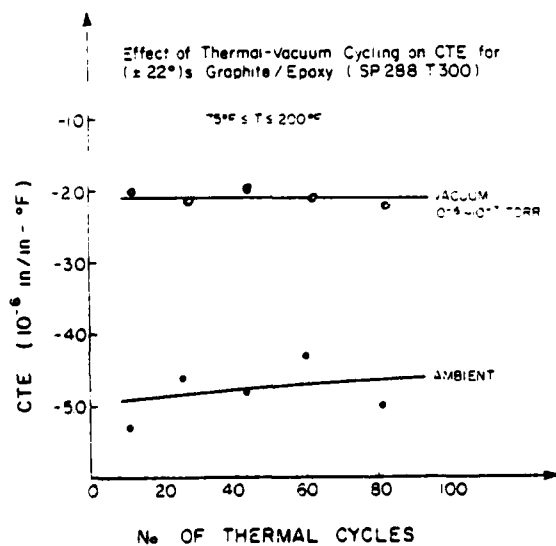


FIG. 4

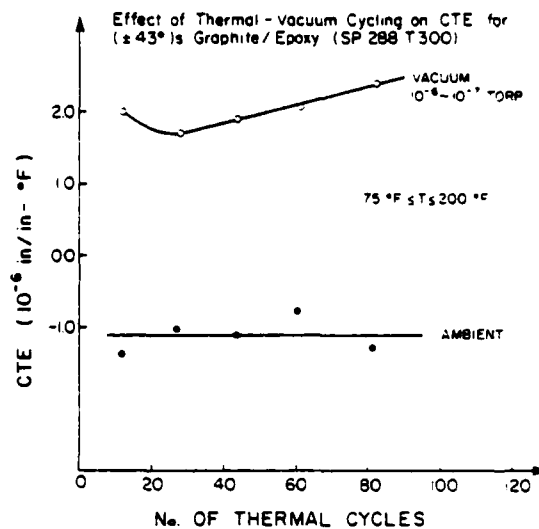


FIG. 3

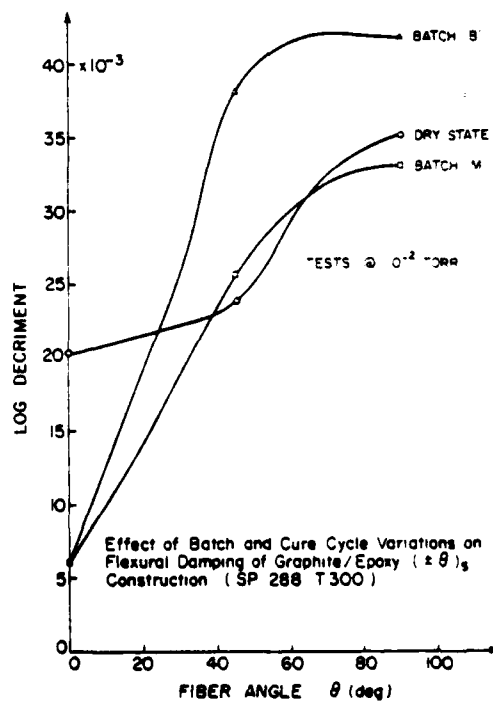


FIG. 5

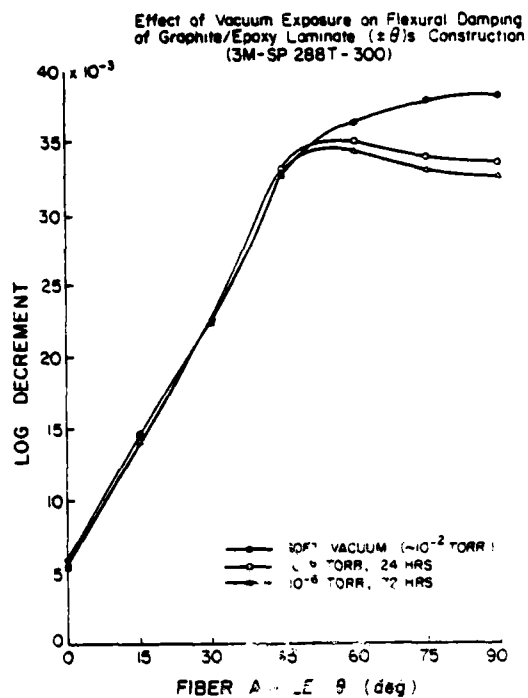


FIG. 6

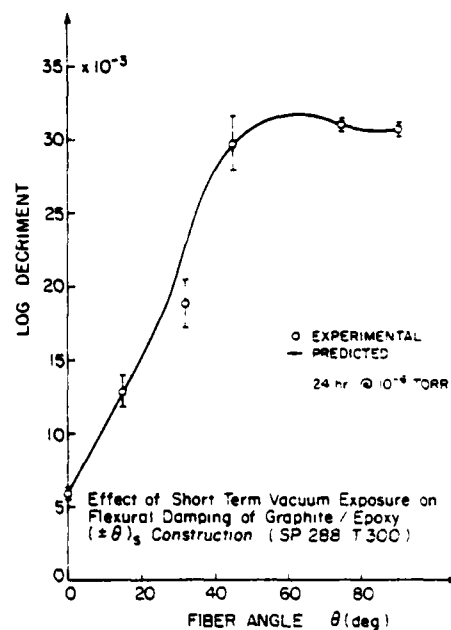


FIG. 7

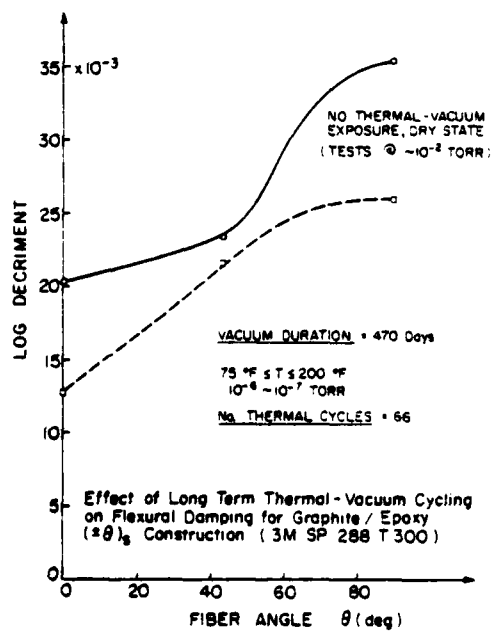


FIG. 8

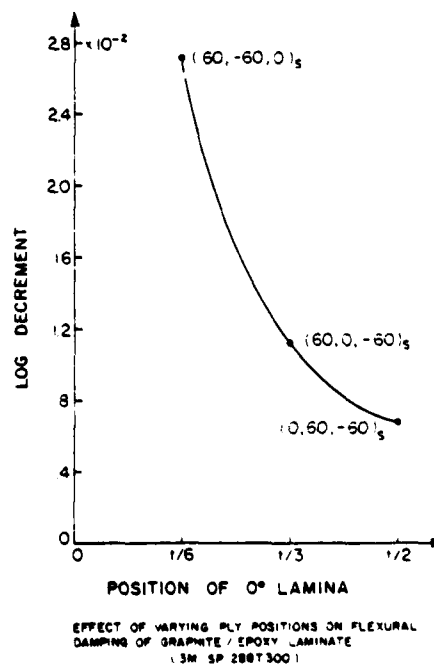


FIG. 9

DELAMINATION FRACTURE MECHANISMS

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Material not received in time for inclusion in the publication.

COMPOSITE IMPACT DAMAGE SUSCEPTIBILITY

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The potential weight reduction realizable through the use of advanced composite materials has caused the displacement of conventional metals from many primary and secondary structures in aircraft like the F-18 and AV-8B. In fiber-reinforced composites such as graphite/epoxy the high strength in the fiber direction is accompanied by low transverse strengths, and, in a laminated form, the interface between adjacent layers provides an additional weak surface. Consequently, there exist situations during the fabrication, maintenance and service of laminated composite structural components when undesirable premature failures may be precipitated, potentially imposing greater threats to composites in comparison to conventional metals. Hard object impact is one such situation and the impact damage susceptibility of composites at low impact velocities is addressed here.

In assessing the impact damage susceptibility of composite laminates, the impact velocity (V) is used to roughly define a low velocity impact situation ($V < 45.72$ m/sec), an intermediate velocity impact situation (45.72 m/sec $< V < 609.6$ m/sec), and a high velocity or ballistic impact situation ($V > 609.6$ m/sec). Of the three impact situations, the low velocity case is considered to be the most commonly occurring situation and therefore of considerable interest for laminated composites. Of major concern is the situation where extensive internal damage occurs with no visual signs of damage on the impacted or other external surfaces (see reference). Internal damage in the form of inter-laminar delaminations, for example, could reduce the service life of aircraft structures if left undetected and uncorrected. The program detailed in this paper attempts to arrive at the various combinations of the low velocity impact variables that cause critical impact damages in composites. The program is comprised of an experimental task and an analytical task. The experimental task methodically varies the different impact parameter values and records the corresponding damage sizes. The analytical task attempts to generate design curves that

summarize the observed impact response in the form of design curves relating a non-dimensional impact parameter to the damage size.

In the experimental part of the program, the impact velocity is monitored to be below 45.7 m/sec (150 ft/sec) to neglect stress wave effects, and is representative of situations involving dropped tools, runway stones, tire blow-out debris, and ground collisions. Based on limiting practical situations, the drop weight and height are assumed not to exceed 111.2 N (25 lbs) and 7.62 m (25 ft), respectively. A 48-ply, $[(+45/0)_2]_2/+45/0/90]_{2s}$ configuration, typical of an F-18 wing section, is assumed for the test laminates, with an individual ply thickness of 0.0132 cm (0.0052 in) for half the specimens and 0.0264 cm (0.0104 in) for the other half. The test panels denoted by types A and B are 0.635 cm (0.25 in.) and 1.27 cm (0.5 in.) in thickness, respectively. To study the effect of impactor geometry on the type and size of impact damage created, spherical-tipped impactors with tip radii of 0.15875 cm, 0.635 cm and 2.54 cm (0.0625 in, 0.25 in. and 1 in.) are used. Test panels are bolted on to wing-type metal substructures with a rib spacing of 60.96 cm (24 in.). The spar spacing is maintained at 20.32 cm (8 in.) for the 1.27 cm (0.5 in) thick panels, and varied between 10.16 cm (4 in.) and 20.32 cm (8 in.) for the 0.635 cm (0.25 in.) thick panels.

For a chosen test bay, four impact locations are identified in the test program: (a) midway between the spars, away from the bay ends; (b) nearer, but not on, the spar support; (c) at the corner where the spar and the rib intersect, but not on either support; and (d) directly over the support, between fasteners. The impact response is expected to be dominated by flexure for location (a), especially in the thinner laminate. When the impact location is moved from (a) to (b) to (c), support constraints affect the response considerably; and for impacts over the support (d), a local crushing effect (indentation) is predominant.

The experimental program is outlined to be carried out in three stages. The first phase involves preliminary instrumented impact tests, listed in Table 1, that are intended to cause through penetration in the panels. These tests are conducted in a drop tower arrangement. From these tests, the variations in the contact force and absorbed energy are obtained as a function

of contact time and panel deflection. The initiation of a damage is manifested by an unloading in the force versus time curve. Based on the results from the first phase, six energy levels corresponding to significant damage levels from incipient damage to laminate puncture are identified for each test case. These are then used in the second phase of the test program (Table 2). In addition, the effect of pre-absorbed moisture (1% by weight) on the impact response under room temperature, ambient conditions, and at -65°F will be investigated in the third phase of the program (Table 3). Tests for the second and third phases will be conducted in a modified drop tower arrangement.

In the analytical part of this program, impact damage is categorized into three groups: (1) flexural damage; (2) internal damage; and (3) contact damage. Flexural damage is comprised of outer ply failure in the form of fiber breakage or matrix splitting between fibers. Internal damage is assumed to be predominantly interlaminar delaminations with accompanying matrix crazing. And contact damage is simplified to mean the depth of indentation at the impact site, not accounting for accompanying fiber/matrix cracks. For each damage type, a design curve will be developed, relating the various impact parameters in the form of a non-dimensional impact parameter to the extent of damage.

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N. M. Bhatia, Impact Damage Tolerance of Thick Graphite/Epoxy Laminates, NADC-79-38-60, Northrop Corporation, Hawthorne, California, January 1979.

TABLE 1. PRELIMINARY DEMONSTRATION IMPACT TESTS

Laminate Type	Panel Size	No. of Panels	No. of Impact Locations	Impactor Tip Radius (Inches)	No. of Impact Tests
A	26-inch x 10-inch	1	4	1/16, 1/4 & 1	12
B	26-inch x 10-inch	1	4	1/16, 1/4 & 1	12
TOTAL					24

TABLE 3. TASK II: TESTS ON PRECONDITIONED* SPECIMENS

Laminate** Type	Panel Dimensions (inches)	Test Temperature (degrees F)	No. of Impact Locations***	Induced Damage Levels at Each Location****	No. of Panel Replicates	No. of Impact Tests
A	26 x 10	-65 ± 5	4	6	1	24
A	26 x 10	RT	4	6	1	24
TOTAL						48

*The 1/4 inch thick panels will be preconditioned at 160F, 95 percent RH until a moisture gain of one percent by weight has been achieved.

**Spar spacing = 8 inches; Rib spacing = 24 inches.

***a,b,c and d are the referenced locations. Impactor tip radius = 1/4 inch.

****Levels 1 to 6 correspond to incipient damage (†) and higher levels, up to laminate puncture (□), if practical.

TABLE 2. TASK I: IMPACT TESTS UNDER RT, AMBIENT CONDITIONS

Laminate Type	Panel Dimensions (in. x in.)	Spar Spacing* (in.)	No. of Impact Locations	Impactor Tip Radius	No. of Damage Levels per Location***	No. of Replicates	No. of Impact Tests	No. of Instrumented Impact Tests
A	$\begin{Bmatrix} 41 \times 6 \\ 41 \times 10 \end{Bmatrix}$	$\begin{Bmatrix} 4 \\ 8 \end{Bmatrix}$	5††	1/16 in.	6	3	90	5†
A	$\begin{Bmatrix} 41 \times 6 \\ 41 \times 10 \end{Bmatrix}$	$\begin{Bmatrix} 4 \\ 8 \end{Bmatrix}$	5††	1/4 in.	6	3	90	5†
A	$\begin{Bmatrix} 41 \times 6 \\ 41 \times 10 \end{Bmatrix}$	$\begin{Bmatrix} 4 \\ 8 \end{Bmatrix}$	5††	1 in.	6	3	90	5†
B	26 x 10	8	4**	1/16 in.	6	3	72	4†
B	26 x 10	8	4**	1/4 in.	6	3	72	4†
B	26 x 10	8	4**	1 in.	6	3	72	4†
TOTAL							486	27

*Rib spacing = 24 inches.

**a, b, c and d

***The 6 damage levels, from incipient damage to laminate puncture (if practical), are achieved by varying indenter mass from 1/4 to 25 lbs and drop height from 1/2 to 25 feet.

†These tests correspond to the maximum damage level (□) at the different impact locations for one replicate.

††Same as ** but b is repeated for both spar spacings.

DAMAGE DEVELOPMENT MECHANISMS IN NOTCHED COMPOSITE
LAMINATES UNDER COMPRESSIVE FATIGUE LOADING

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Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061

Composite materials are being used to effect decreases in weight and increases in the durability of aircraft and other vehicular structures. In order to fully realize these and related advantages, it must be possible to accurately prescribe and realistically meet specific design requirements on damage tolerance. At the present time, the understanding of damage in composites and its relationship to material response is incomplete. A primary difficulty is that damage mechanisms have been identified and studied by making lists of damage components (such as broken fibers, matrix cracks, delaminations, etc.) without regard for damage processes and damage states. A more systematic and unified approach should include the identification of a damage state (or states) which is defined by material properties and determine how this state governs the strength, stiffness, and life of the material. The general objective of damage studies should be to determine those damage states which are characteristic of material response and, therefore, are necessary for the formulation of appropriate models.

The specific objectives of this investigation are to:

- determine the nature of damage induced in graphite epoxy laminates with center holes by cyclic compressive loading,
- determine the influence of the damage state on residual strength.

The investigation is being conducted on AS/3501-6 graphite epoxy laminates with two stacking sequences:

- $(\pm 45, 0_2, \pm 45, 0_2, \pm 45, 0, 90)_{2S}$, (48 plies),
- $(\pm 45, 90, -45, +22.5, -67.5, -22.5, +67.5, \pm 45, +67.5, +22.5, -67.5, -22.5, \pm 67.5, \pm 22.5, 0_2, +22.5)_S$, (42 ply).

The specimens are 25.4 mm wide with a 6.4 mm diameter center hole (Figure 1). Compression-compression fatigue tests are run in load control at 10 Hz with a stress ratio $R=10$. Nondestructive evaluation techniques are used during the tests and at selected cyclic intervals to follow the initiation and development of damage. The nondestructive methods include: optical techniques (microscope and borescope) • ultrasonic C-scan • ultrasonic attenuation • acoustic emission • replication • X-radiography • stiffness • thermography. In addition, some specimens are sectioned to aid in the interpretation of the nondestructive data. Brief descriptions of each method are given in Reference 1.

The approach to the investigation is divided into three tasks:

Initial Characterization

- initial defects (NDE and sectioning)
- static strength (tensile, compressive)
- static stiffness (tensile, compressive)

Damage Growth Study

- two maximum cyclic compressive stress levels
- nondestructive monitoring of damage development and formation of the damage state during cyclic loading (real time tests and interrupted tests)
- define "life" at each stress level

Residual Strength Study

- determination of compressive residual strength after selected cycles at the two stress levels
- effect of different damage modes on residual strength
- determination of the precise nature of the final fracture event and how it is controlled by the damage state.

Some results are presented in Figures 2 through 4.

Preliminary Observations

monotonic compressive loading

- Initiation of buckling in a constant load rate test can be detected by the strain gages as a sharp change in the local transverse and longitudinal strain rates and a corresponding knee in the stress strain curves.
- Initiation is also accompanied by a very definite increase in acoustic emission rate.
- For the purpose of damage investigations, the change in local strain rate is a very good definition of "failure". The test can be stopped before fracture for NDE studies and sectioning.
- The average value of applied compressive stress at the initiation of buckling of the 48 ply laminates is $\sigma^* = 47$ ksi.

cyclic compressive loading

- The damage state consists of matrix cracks in the zero and 45 deg plies and delamination.
- The damage zone is confined to a region around the hole measuring 2.5 hole diameters.
- Matrix cracks in the zero deg plies develop first, followed by matrix cracks in the 45 deg plies.
- With additional cycles, new cracks and delaminations form. The primary damage mode during this period is delamination growth.

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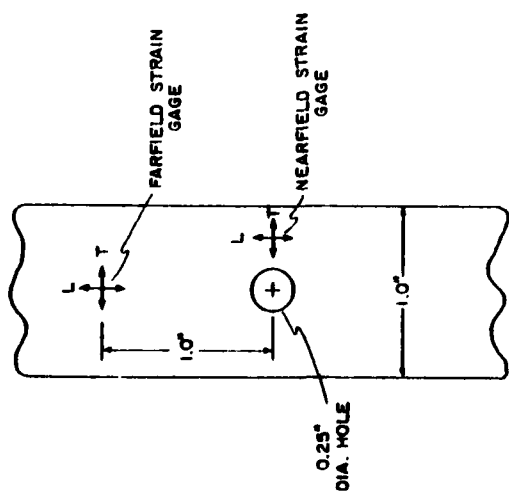


Figure 1. Compression test specimen and strain gage locations

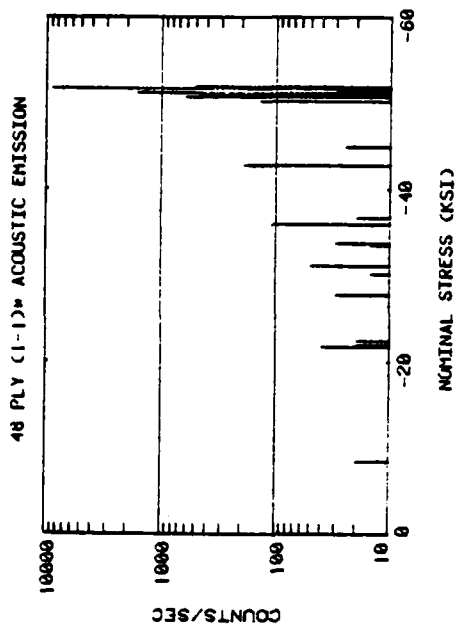


Figure 2a

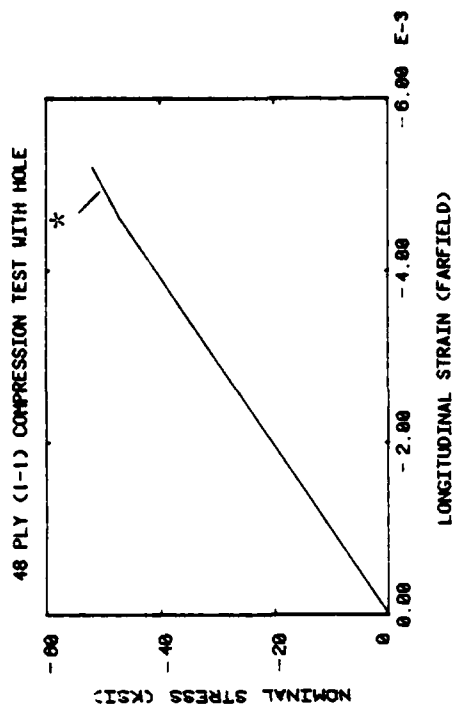


Figure 2c

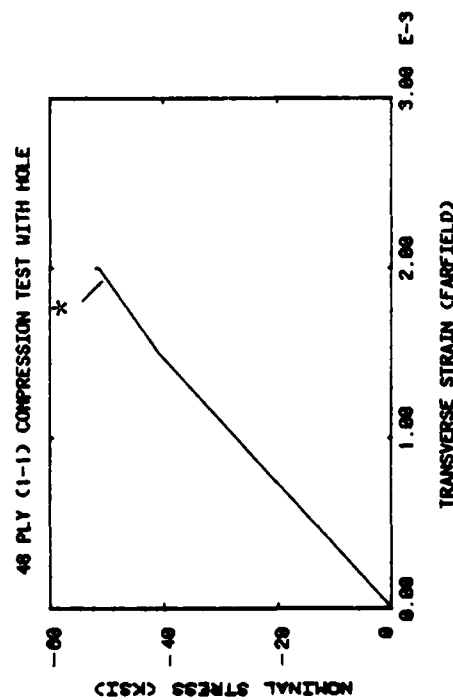


Figure 2b

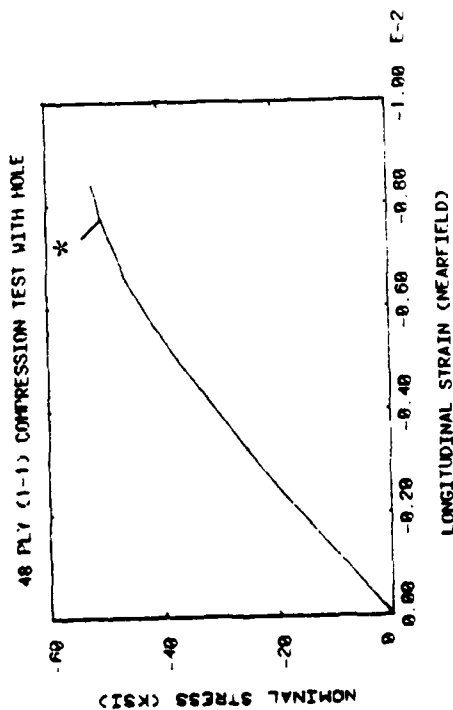


Figure 2d

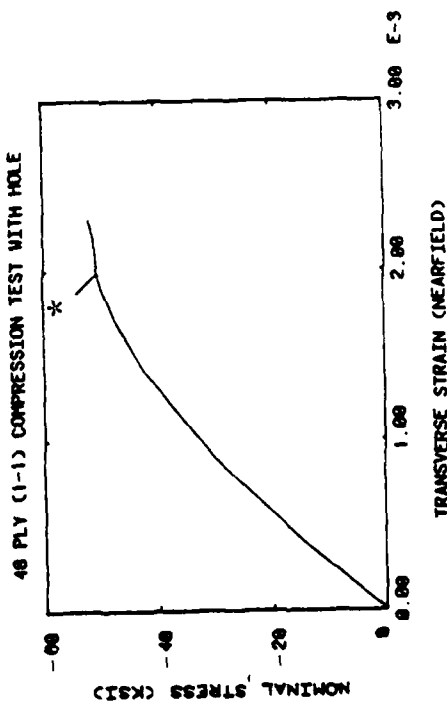


Figure 2e

Figure 2. Compressive stress-strain curves and acoustic emission data for a 48 ply specimen with a center hole (*indicates the initiation of buckling).

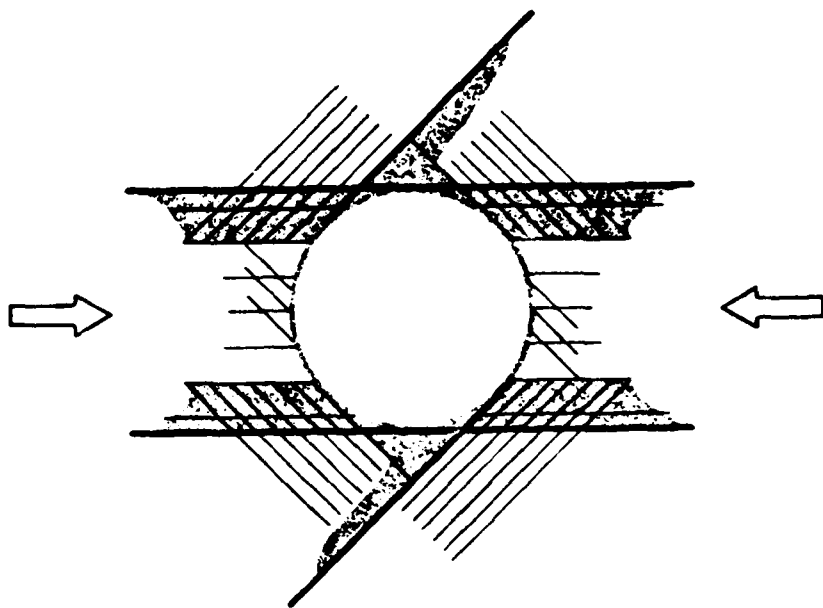
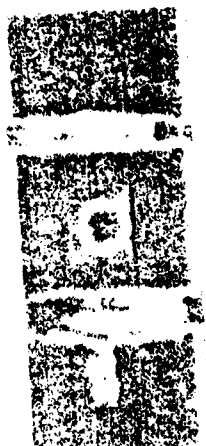


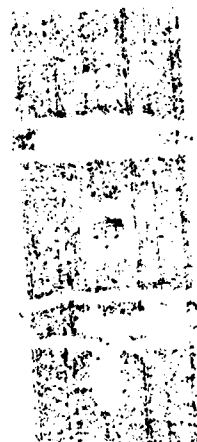
Figure 3. Schematic of the damage state around a hole in a 48 ply laminate due to compression-compression fatigue (lines represent matrix cracks and shaded regions represent delaminations).



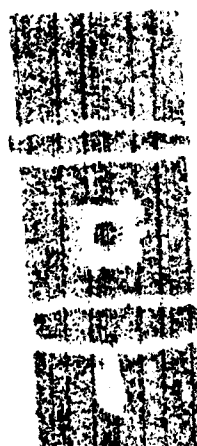
C-scan
Initial



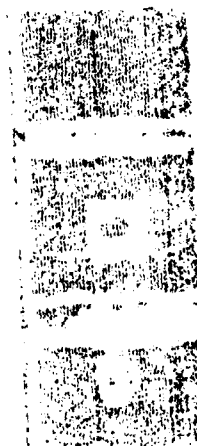
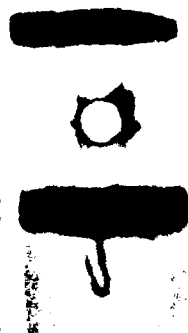
C-scan
100k cycles



C-scan X-ray
200k cycles



C-scan X-ray
300k cycles



C-scan X-ray
350k cycles



Figure 4. C-scans and TBE enhanced X-rays of damage development in a 48 ply specimen tested at a maximum compressive stress of 40 ksi.

COMPRESSION FATIGUE LIFE PREDICTION FOR COMPOSITE STRUCTURES

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The objective of this program is to develop an analytical compression fatigue life prediction methodology and supporting experimental data base for bolted joints in advanced composite structures. During the first year of this three-year program, the first of three tasks has been completed.

In Task I - Preliminary Analysis and Design, a literature search has been performed to summarize experimental data and analytical methodologies. Failure modes and mechanisms in bolted composite joints are being experimentally evaluated. Test specimens representative of commonly used joints have been designed. In Task II - Analysis/Methodologies, a methodology will be developed to predict the spectrum fatigue life of bolted joint composite structures by modeling damage initiation, growth, and failure mechanisms observed during specimen testing in Task III. Both empirical and analytical approaches will be investigated. In Task III - Experimental Data, the data required for development, validation, and use of analytical methodology will be acquired. Testing in Part A of this task will represent about one-third of the total number of tests and will provide sufficient data to evaluate and develop analytical techniques. In Part B, a sufficient number of replicates will be tested to statistically characterize life, and to determine the sensitivity of life and scatter to loading and environmental variables.

In the literature survey, Reference 1, experimental data and analytic methodologies for static strength and fatigue life of bolted composite joints have been summarized. Not a great deal has been published concerning strength analysis of bolted joints in composite materials since publication of the literature survey of Garbo and Ogonowski (Reference 2). Predictions based on characteristic dimensions, coupled with various failure criteria, have been compared with recently published test data and show good correlation.

Recent research programs have provided insight into fatigue damage initiation and propagation in composite materials. Generally, damage initiates immediately as debonds between fibers and matrix at geometric discontinuities, debonds rapidly progress to matrix cracks which progress slowly until delaminations occur. Delaminations and matrix cracking interact to rapidly degrade the matrix until fiber rupture or buckling causes final failure. The sequence of damage progression appears to be laminate dependent, with 45 degree plies observed to play a key role in the

process. Damage propagation from notches and holes in tension-tension fatigue initiates in the longitudinal directions and appears to be well contained between axial splits at hole edges or between the specimen edge and notch root for edge notches. Delaminations appear to be most severe at interfaces between 0 and angle plies nearest the free surface.

Research on the mechanisms of composite fatigue behavior is beginning to lead toward development of analytical prediction procedures. These procedures include detailed modeling of the mechanical behaviors and failure modes through finite element analyses, as well as more fundamental examination of fatigue failure modes. Fundamental approaches based on rigorous analyses of simplified models is a preferred starting point for development of a fatigue analyses methodology. These approaches are more easily applied to the wide variety of laminates and geometries found in aircraft structures and their limitations are more easily identified than are those approaches based on finite element analyses. Finally, the effect of load frequency on fatigue life of composite structures required considerably more investigation. Care must be exercised in preparing test programs so that the effects of cyclic loading frequency will not influence the trends of the data.

The general fatigue damage progression sequence identified in the literature search indicates that it is a coupling of interlaminar cracking and delamination which degrades the matrix to the point of which specimen rupture occurs. Because interlaminar cracking is controlled by inplane stresses and delaminations are controlled by interlaminar stresses, the coupling must depend on stress distributions in both planes.

A strain energy density factor formulation provides a technique for analyzing multiaxial stresses on matrix cracks. In this approach to quantifying fatigue degradation in composites, it is assumed that the matrix is the weak link of the system. Each ply of the laminate is modeled (Figure 1) as an isotropic layer of resin containing a through-the-thickness crack sandwiched between the edges of two semi-infinite orthotropic plates. E and ν are the elastic constants of the resin, while E_1 , E_2 , ν_{12} , μ_{12} are the elastic constants of the equivalent orthotropic material surrounding the resin strip. The subscripts 1 and 2 represent directions parallel and perpendicular to the ply fibers, respectively. S is the strain energy density factor ahead of the crack like flaw in the matrix. This approach has been shown to adequately correlate the fatigue behavior of unloaded fastener hole specimens fabricated with different lay-ups. It will be extended to encompass the effects of bearing loads.

In the design of specimens for the test specimens, bolted joints of the F-18 and AV-8B were used as guidelines for specimen geometries, and bearing and by-pass load levels. Specimen geometries are indicated in Figure 2, and the test program is outlined in Table I. In the experimental program, effects of several

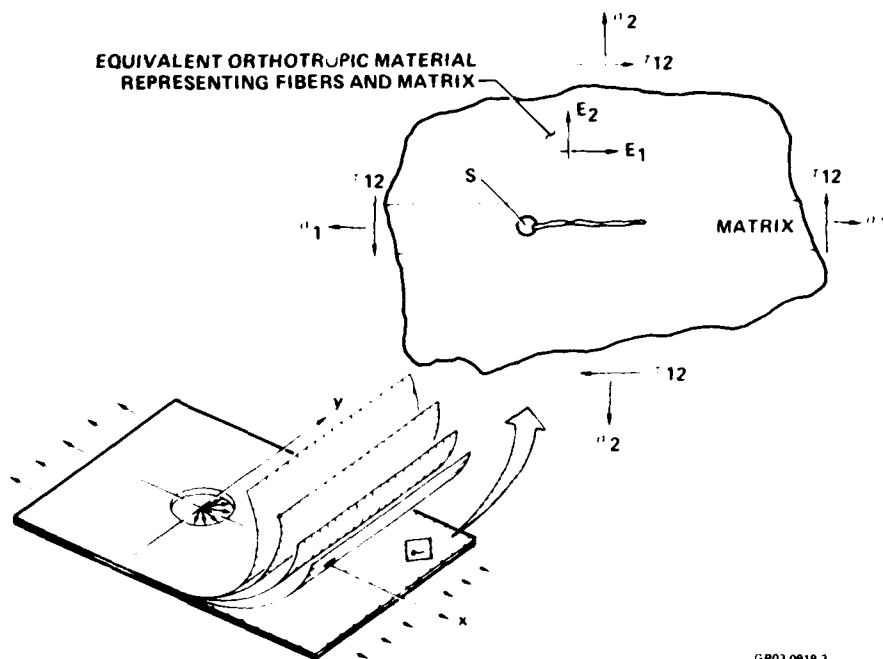


Figure 1. Cracked Lamina Model.

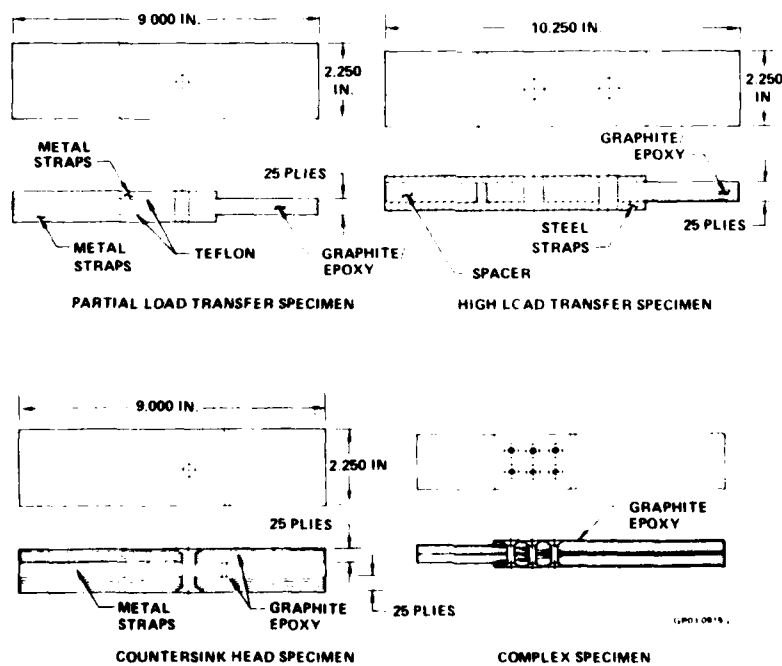
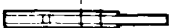


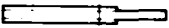
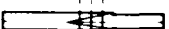
Figure 2. Test Specimens.

TABLE 1. FATIGUE TESTS SUMMARY.

PART A - PRELIMINARY

SPECIMEN TYPE	CONSTANT AMPLITUDE	SPECTRUM	NUMBER OF SPECIMENS
	240	40	280
	120	40	160
	60		60
SUBTOTAL			500

PART B - REPLICATION

SPECIMEN TYPE	CONSTANT AMPLITUDE	SPECTRUM	NUMBER OF SPECIMENS
	320	100	420
	80		80
	240	60	300
SUBTOTAL			800
TOTAL NO OF FATIGUE SPECIMENS			1300

(1003 0919)

variables will be evaluated. Tests will be performed to fully evaluate fatigue life for all combinations of four environmental conditions, three lay-ups, three stress levels, four stress ratios, two bearing by-pass load ratios and two different spectra. In addition, tests will be performed to evaluate the effects of countersink on fatigue life.

ACKNOWLEDGEMENT

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ENVIRONMENTAL EFFECTS ON COMPOSITE
DAMAGE CRITICALITY

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Hawthorne, California 90250

Conventional metals and laminated composites differ appreciably in their response to, and tolerance of, various kinds of damage. One such damage, induced by low velocity impact of hard objects on laminated composites, is notable. The inherent heterogeneity and the relatively weak interface between plies created through the lamination process make composite laminates susceptible to interlaminar delaminations under low velocity impact conditions. Comparatively, a metal would suffer less severe damage under these conditions and, furthermore does not exhibit failures like delaminations. Of concern is the impact situation where laminated composites suffer considerable internal damages with no visual signs of damage on the impacted or other free surfaces (see reference). The exposure of impact-damaged structural components to humid environment during service accelerates the moisture absorption process, thereby possibly affecting the integrity of the component under adverse operating conditions, viz. low temperature, ambient or elevated temperature, humid conditions.

This paper discusses an ongoing experimental program that investigates damage growth mechanisms and residual strength degradations in laminated composites as a result of impact-induced damages, and the possible deleterious effect of environment on pre-conditioned, impact-damaged laminates. A 48-ply, AS/3501-6 laminate, representative of an F-18 wing skin layup, and a 42-ply, AS/3501-6 laminate, representative of a Harrier wing skin layup, are tested in the program. A preliminary impact damage study was conducted on both the layups as shown in Figure 1, to determine the drop heights required for a 40N (9 lb) weight to cause two types of damages. Type I damage is defined as a delamination, 3.81 cm to 5.08 cm (1.5 to 2 in.) in diameter, with no visual evidence of damage on

the impacted and the opposite surfaces. This was produced using a blunt impactor. Type II damage is defined as a delamination, approximately 5.08 cm (2 in.) in diameter, with visible cracking on either or both faces to allow direct moisture entry into the laminate. This was produced using a sharp impactor. Two specimen configurations (Figure 2) were designed for the program to produce a uniform strain field within a $\pm 3\%$ variation, in the laterally unsupported test section, up to a maximum applied strain level of 0.005. The test section was unsupported laterally to allow for unconstrained growth of interlaminar delaminations. Stabilization was provided outside the test section to prevent gross buckling of the test specimen (Figure 3). Test specimen design and strain uniformity, based on a finite element analysis, were verified through preliminary tests on strain-gaged, undamaged specimens subjected to static compression and tension-compression fatigue loads with a maximum strain amplitude of 0.005.

After the preliminary impact tests and specimen design verification tests were successfully completed, fabricated test specimens were all impact-damaged, as required, to produce the desired damages. The test program was divided into two tasks: (1) room temperature dry tests; and (2) elevated (200F) temperature tests under humid (95%RH) environment, and low temperature (-65F) dry tests. The various tests under task (1) are listed in Table 1. The breakdown of the tests for each test series to obtain an S-N (maximum fatigue stress amplitude versus cycles to failure) curve is given in Table 2. Test specimens for task (2) were pre-conditioned, before testing according to Table 3, to absorb 1% of moisture by weight. Each test series in Table 3 will be tested as shown in Table 4 to obtain an S-N curve. During the fatigue tests in both the tasks, four specimens from each test series will be periodically C-scanned, as shown in Tables 2 and 4, to monitor damage growth in the specimens for various maximum fatigue load amplitudes. The room temperature test results will quantify the effect of impact-induced damage on the S-N behavior of the chosen laminates at $R = -\infty$, -1 and 0. Elevated and low temperature test results will quantify any deleterious effects of environment on the S-N behavior of these laminates at $R = -1$.

REFERENCE

N. M. Bhatia, Impact Damage Tolerance of Thick Graphite/Epoxy Laminates, NADC-79038-60, Northrop Corporation, Hawthorne, California, January 1979.

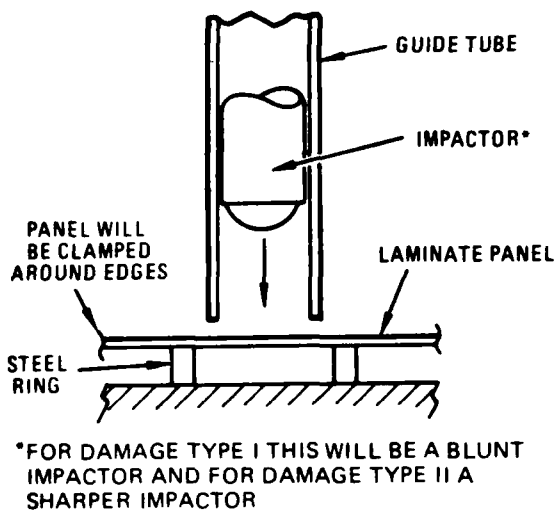


FIGURE 1. PROPOSED TECHNIQUE FOR OBTAINING IMPACT DAMAGE

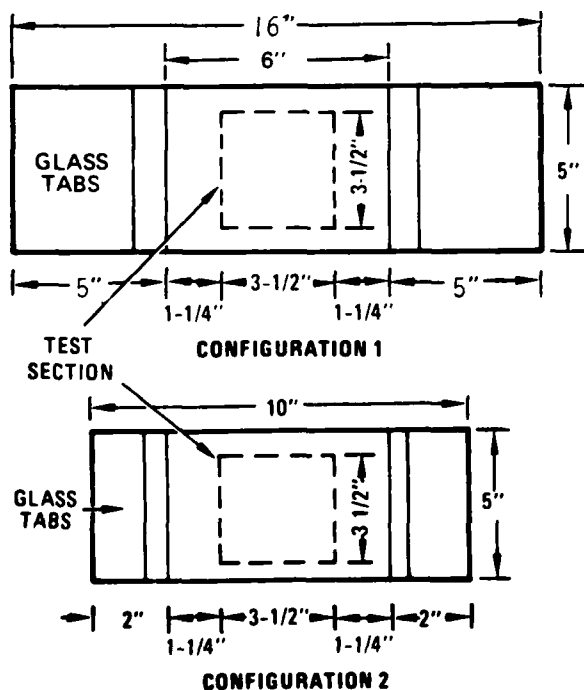


FIGURE 2. SPECIMEN CONFIGURATION USED IN THE PROGRAM

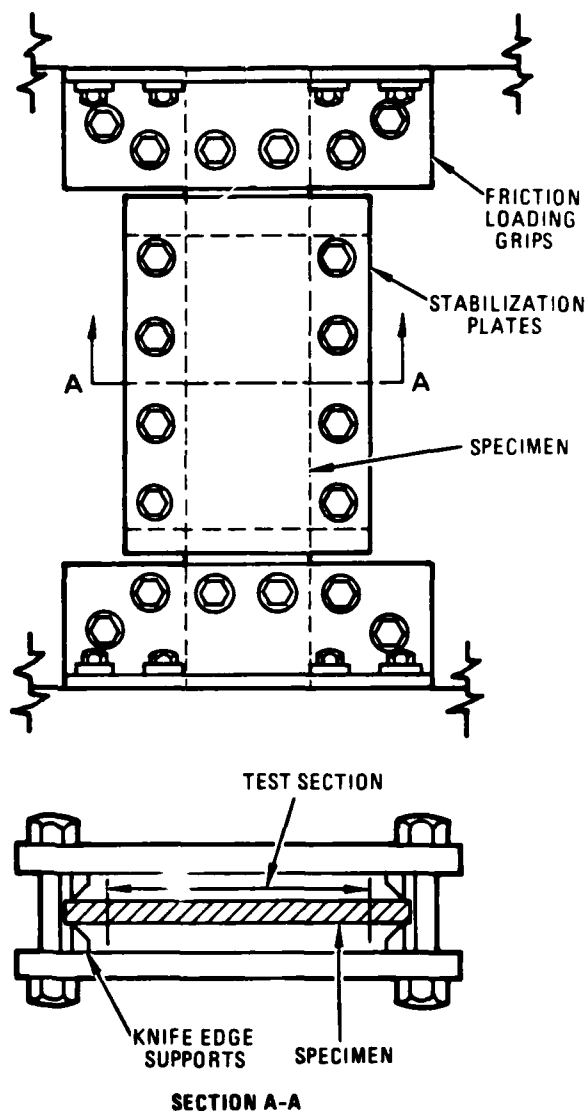


FIGURE 3. COMPRESSION LOADING STABILIZATION FIXTURE

TEST SERIES NO.	TYPE TEST	LAMIN- ATE	DAMAGE LEVEL	NO. OF SPE- CIMENS
1	TENSION- TENSION (R = 0)	48 PLY	I*	15
2	TENSION- TENSION- (R = 0)	48 PLY	II**	15
3	TENSION- TENSION- (R = 0)	42 PLY	I	15
4	TENSION- TENSION- (R = 0)	42 PLY	II	15
5	TENSION- COMPRESSION (R = -1)	48 PLY	I	15
6	TENSION- COMPRESSION (R = -1)	48 PLY	II	15
7	TENSION- COMPRESSION (R = -1)	42 PLY	I	15
8	TENSION- COMPRESSION (R = -1)	42 PLY	II	15
9	COMPRESSION- COMPRESSION (R = -∞)	48 PLY	I	15
10	COMPRESSION- COMPRESSION (R = -∞)	48 PLY	II	15
11	COMPRESSION- COMPRESSION (R = -∞)	42 PLY	I	15
12	COMPRESSION- COMPRESSION (R = -∞)	42 PLY	II	15
			TOTAL	180

*APPROXIMATELY 1-1/2-2 IN.
DIAMETER

**APPROXIMATELY 2 IN.
DIAMETER

TABLE 1. DAMAGE GROWTH TEST
PLAN - ROOM TEMPERATURE TESTS

EXPECTED CYCLES TO FAILURE*	NO. OF REPLICATES	REPLICATES TO BE C-SCANNED 5 TIMES PRIOR TO FAILURE
0 (STATIC TEST)	3 (0)**	0
10^3	3 (4)	1
10^4	3 (4)	1
10^5	3 (4)	1
10^6	3 (3)	1
TOTAL	15	4

*UNFAILED SPECIMENS WILL BE RETESTED AT A
HIGHER LOAD LEVEL

**NUMBERS IN PARENTHESES WILL BE USED FOR THE
TENSION-COMPRESSION, ROOM TEMPERATURE
FATIGUE TESTS IN FIGURE 9. STATIC TEST DATA
FOR THIS CASE WILL HAVE BEEN GENERATED IN
THE TENSION-TENSION AND COMPRESSION -
COMPRESSION FATIGUE TEST CASES

TABLE 2. VARIABLES FOR EACH
S-N CURVE IN FIGURE 9

TEST SERIES NO.	TYPE OF TEST	LAMINATE	DAMAGE LEVEL	TEST ENVIRON- MENT	NO. OF SPECIMENS
13	TENSION-COMPRESSION (R = -1)	48 PLY	11**	200+5 F 95%R.H.	16
14	TENSION-COMPRESSION (R = -1)	48 PLY	11	-65+5 F	16
15	TENSION-COMPRESSION (R = -1)	42 42 PLY	11	200+5 F 95% R.H.	16
16	TENSION-COMPRESSION (R = - 1)	42 PLY	11	-65+5 F	16
**APPROXIMATELY 2 IN. IN DIAMETER			TOTAL		64

TABLE 3. ENVIRONMENTAL EFFECTS ON DAMAGE GROWTH TEST PLAN

EXPECTED CYCLES TO FAILURE*	NO. OF REPLICATES	REPLICATES TO BE C-SCANNED 5 TIMES PRIOR TO FAILURE
0 (STATIC TEST)	4**	0
10^3	3	1
10^4	3	1
10^5	3	1
10^6	3	1
TOTAL	16	4

*UNFAILED SPECIMENS WILL BE RETESTED AT A HIGHER LOAD LEVEL

**TWO TESTS EACH IN BOTH COMPRESSION AND TENSION

TABLE 4. VARIABLES FOR EACH S-N CURVE IN FIGURE 11

EFFECTIVE THERMAL CONDUCTIVITIES
OF FIBROUS COMPOSITES

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Composites are known to be susceptible to thermal environmental factors such as moisture and temperature gradient. The former causes swelling leading to a loss of strength; the latter results in thermal stresses especially damaging to those composites whose constituents have widely different coefficients of thermal expansion. The diffusion of heat and moisture are, therefore, important considerations in the design and application of composites.

Early studies of thermal properties for composites proceeded mainly on the basis of models - a useful qualitative approach having limitations on its quantitative validity. This type of investigation is exemplified by the work of Hashin and Shtrikman, Beran and Silnutzer, and Springer and Tsai, just to mention a few of the pioneering works. The results of these model studies are usually expressed in terms of an effective thermal conductivity, k_e . As an example, the transverse thermal conductivity for a fiber-matrix composite is given by Behrens (4) as:

$$(k_e/k_m) = [\beta + 1 + V(\beta - 1)] / [\beta + 1 - V(\beta - 1)] \quad (1)$$

where k_m is the thermal conductivity of the matrix material. The ratio of fiber conductivity to the matrix conductivity is denoted by β , and the volumetric content of the fibers is expressed by V , as a fraction of the total volume.

It is of course recognized that formulas of this type do not wholly consider the influence of relative fiber position of the heat conduction. Thus, they cannot be expected to yield quantitative accurate results for the myriads of possible fiber geometrical patterns, especially under conditions of high-density fiber packing. However, these various model equations do indeed serve useful purposes in estimating the conductive capacity of composites.

The purpose of the present work is primarily to investigate a class of heat conduction problems in composites for which the proximity effects of the embedded fibers are significant. The method of approach is a rigorous solution of the steady-state heat conduction equation.

This paper addresses itself to the problem of transverse heat conduction in the steady state through composite materials, in particular those having isotropic fibers uniformly dispersed in an isotropic matrix. Specifically considered is a class of fiber-matrix composites having two geometrical arrangements for the fibers: (i) fibers in a uni-directional orientation, and (ii) layered composites with fibers laid alternately along two mutually perpendicular directions - often referred to as the 0/90 arrangement. Results from these investigations are expressed in terms of an effective thermal conductivity which is useful in the following context: Fibers or bundles of fibers in a matrix are usually of such a small dimension that, for most engineering applications, the scale of desired resolution spans a number of fibers or bundles of fibers. A typical 0.5 - inch thick graphite-fiber composite slab consists of 50 or more layers as graphite-tapes. Thus from a "macroscopic" viewpoint, an effective thermal conductivity reflecting the phenomenological aspect of the dispersed fibers is a first and, in most instances, an adequate requirement to analyze the temperature gradients in such a composite body. This equivalence idea is not unlike the representation of a real gas by a continuum with phenomenological coefficients.

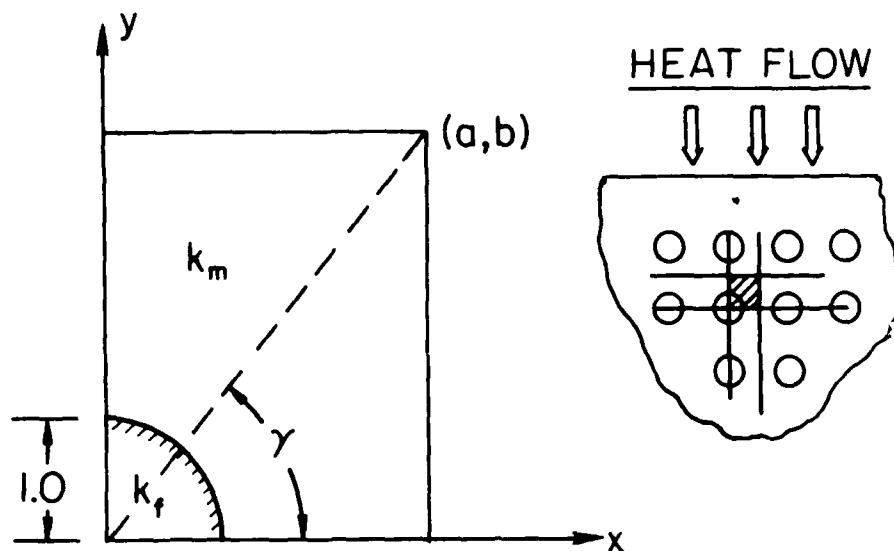
Because of the disparity between the two scales - scale of resolution and scale of the fiber dimension - rigorous evaluation of effective thermal conductivity can be obtained only from consideration of a unit cell (Figure 1a), of which the remainder of the composite is either a replica or a mirror-image. For unidirectional fibers, a regular dispersion pattern (with the fiber centers forming either a rectangle or a triangle) is assumed. Figure 1a depicts a unit-cell isolated from a fiber-composite with fibers forming a rectangular array. Such a unit-cell construction assumes a one-dimensional heat flow downward at its boundaries. Consequently, both the horizontal line midway between fibers and the one passing through the fiber centers are isotherms and the corresponding vertical lines are adiabats.

Figure 1b depicts a unit-cell constructed for fiber-composites with fiber-centers forming a triangular pattern.

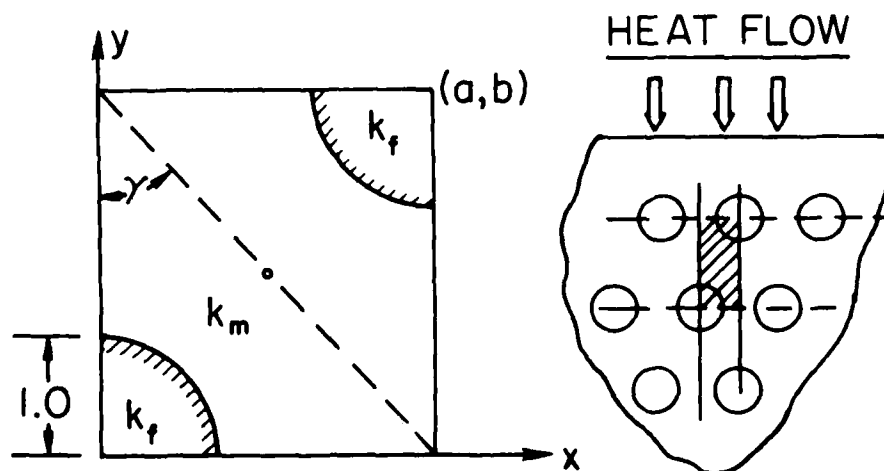
For fiber-composites with 0/90 arrangement, the unit cell is a box; it consists of two quarter-fibers located along two perpendicular but non-intersecting edges of the matrix box. Figure 2 depicts such a unit cell.

The results of analysis are typified by Figure 3 which shows the ratio of an effective conductivity k_e to be conductivity of the matrix material k_m . The ratio (k_e/k_m) is dependent on many factors which are: 1) k_f/k_m the ratio of the fiber to matrix conductivity (k_f/k_m), 2) the volume ratio which describes the denseness

of packing and 3) the fiber to fiber orientation or the angle γ . The effective conductivity k_e is seen to vary over a wide range. Complete results of the analysis can be obtained in AFWAL-TR-80-3012, Mar. 1980.



a) RECTANGULAR PATTERN



b) STAGGERED PATTERN

FIGURE 1. UNIT CELL OF A UNIFORMLY-DISPERSED FIBER-MATRIX

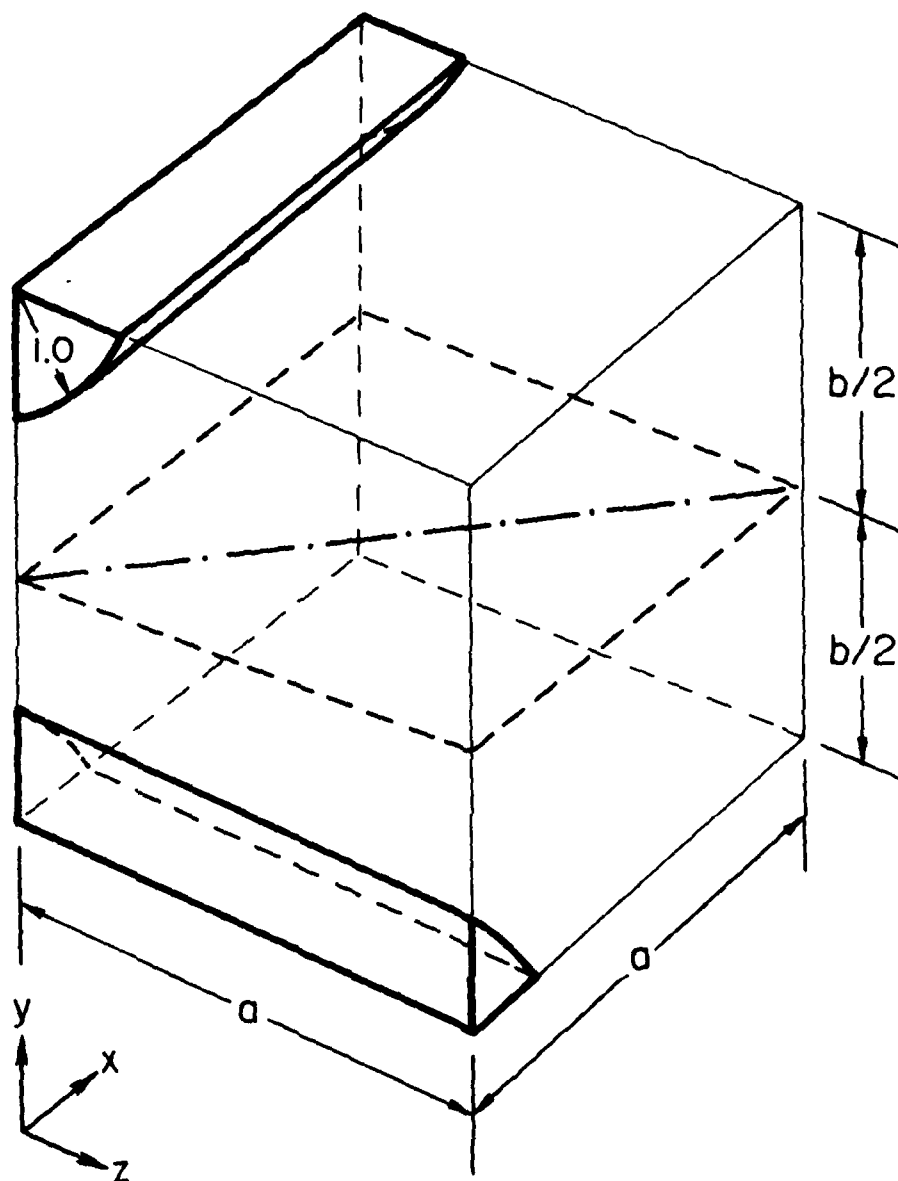


FIGURE 2. UNIT CELL FOR (0/90) FIBER-COMPOSITE

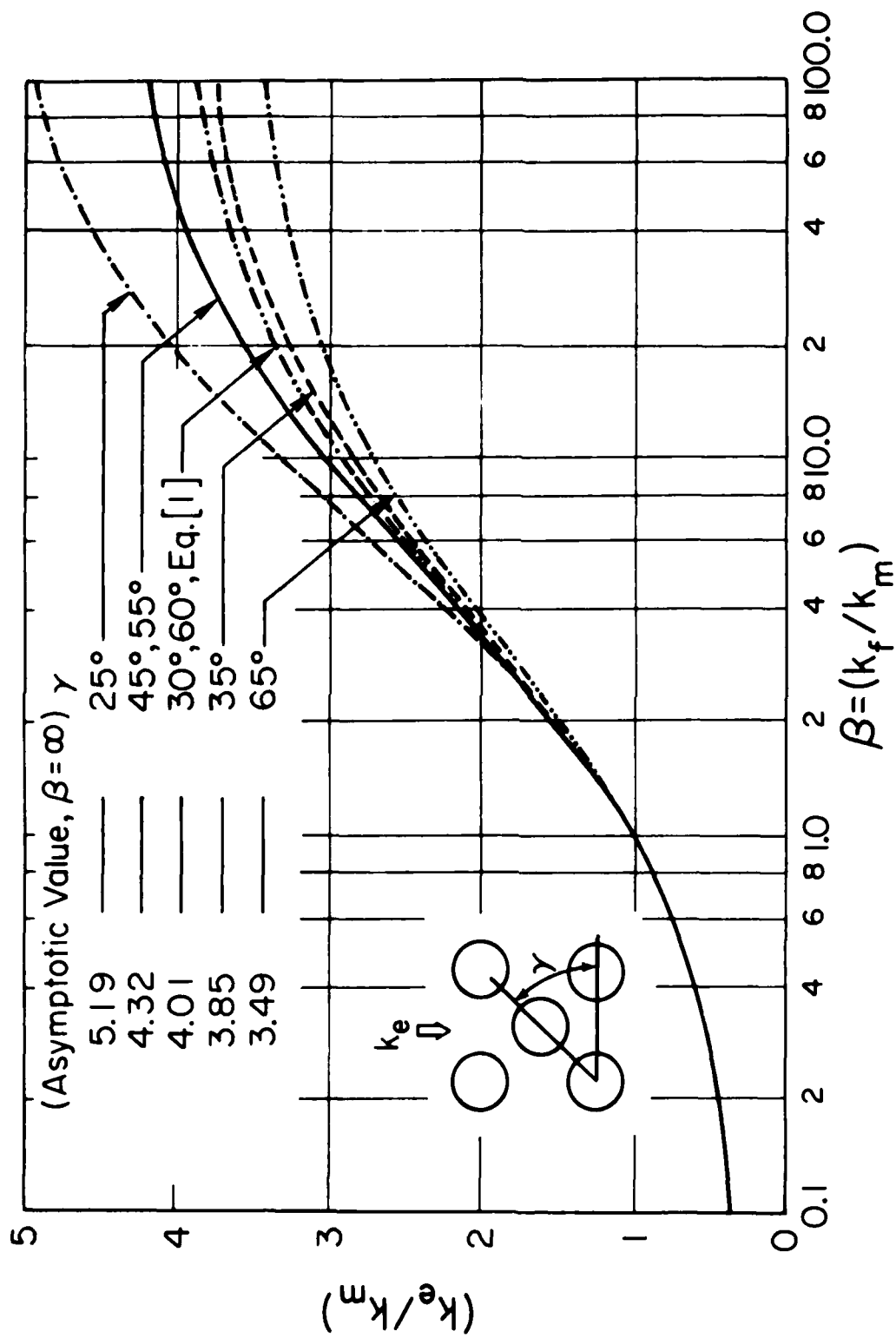


FIGURE 3. EFFECTIVE CONDUCTIVITIES FOR UNIDIRECTIONAL FIBER-COMPOSITES
(Staggered Array, $\nu = 0.6$)

ADVANCED RESIDUAL STRENGTH DEGRADATION
RATE MODELING FOR ADVANCED COMPOSITES

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Introduction of advanced composite materials into aircraft structural applications in recent years has turned attention towards the development of analytical methodologies to assure the same level of structural reliability found in comparable metal structures. Composite materials, however, are not well suited to a simple application on extrapolation of analysis methods used in metals. Basic differences in scale and structure between composites and metals prevent such an approach. Behavior of composites under static and dynamic loads is dominated by material structural discontinuities which manifest as anisotropy and inhomogeneity.

Unlike metals where most types of defects could conservatively be considered as cracks, initial damage in composites may take many different forms having entirely different initial characteristics and possibly propagating in a different manner. A common type of damage which is unique to composites is that due to low velocity impact which can readily occur due to a tool being dropped or pieces bumping together. In service a similar damage can occur at higher velocities (but normally much lower impactor mass) due to hail or rocks flipped from tires, etc..

Associated with the difficulties in defining the exact nature of damage and its effect on service life is the added problem of developing adequate nondestructive inspection methods to detect the damage and analysis methods to define its severity. Development of methods of defining, detecting, evaluating, and analyzing these types of damage in a framework consistent with current durability and damage tolerance requirements is the problem area addressed by the current effort. Specifically, the program was designed with the aim of developing a methodology for predicting the residual strength and its rate of change as a function of fatigue loading for advanced composite structures used in modern aircraft construction. The systematic approach to the accomplishment of this aim included:

- Selection of a well defined, repeatable initial damage configuration.
- Tracking of the fatigue induced damage growth by a nondestructive inspection procedure.
- Correlating the actual damage dimensions with the results of the NDI procedure.
- Relating the residual strength to the damage configuration.

The program was divided into three major phases: Task I was designed to screen the static and fatigue induced damage growth characteristics of two damage types. Based on these results a single damage condition was to be selected for study in Tasks II and III; Task II was devised to generate statistically significant data sets for the static and fatigue life behavior and for the fatigue induced damage growth and residual strength behavior from which a model for predicting the damage growth characteristics and the subsequent residual strength can be developed; Task III included the study of three variations in the loading/environmental parameters to evaluate the applicability of the model over a range of loading/environmental conditions and update the model if required. Task I has been completed and a final report (AFFDL-TR-79-3095) issued. The testing phases of Tasks II and III have been accomplished and the analysis and reporting efforts encompassing these two tasks are in progress.

TASK I - Preliminary Screening

Two laminates of T300/5208 graphite/epoxy material were selected for this study: 1) a 24-ply, 67% - 0° ply, (0/45/0₂/-45/0₂/45/0₂/-45/0)_s and 2) a 32-ply quasi-isotropic (0/45/90/-45₂/90/45/0)_{2s}. These complied with the requirements that one laminate be delamination prone (32-ply) while the other would not (24-ply) under fully reversed tension-compression fatigue loading. In addition, these particular stacking sequences were chosen since results from this program could then contribute to a comprehensive static and fatigue data base being developed for these laminates of the same material under AFML Contracts F33615-77-C-5140 and F33615-78-C-5090 and previously developed for a different material (T300/934) under AFML Contracts F33615-75-C-5118 and F33615-77-C-5045.

A three-inch wide by fourteen-inch long specimen with a nine-inch gage section was used for both fatigue and static tests. Two damage types were included in the study: 1) A poorly drilled hole with multiple delaminations surrounding the hole and 2) low velocity impact damage produced by dropping an impactor on the panel.

Static tension and compression properties were determined for both laminates with ten replicates per a condition. Compression tests were conducted using both the fatigue supports which are full platen restraints with a 2.15 in. (55 mm) square window and 4-bar buckling guides to evaluate the inherent local buckling characteristics of the damaged laminate. Fatigue tests conducted to obtain the R = -1 S-N characteristics for each of the four laminate/damage conditions also provided the basic fatigue induced damage growth characteristics. Damage growth was monitored using a Holosonics Series 400 Holscan ultrasonic unit. A subset of both static compression and fatigue tests was conducted to provide a statistically based answer as to the effect of TBE on subsequent material behavior.

Static tension data for the initial "as damaged" condition indicate that while the damaged hole causes a major drop (~50%) in static strength, the impact condition results in little if any strength reduction. However, under static compression loading both damage conditions produced a significant strength reduction in both laminates with the more severe loss observed for the damaged hole condition.

Under cyclic loading a distinct similarity in the general shape of the S-N curve was evident for the two damage types of the 24-ply laminate, but the damage growth behavior differed. Damage in this laminate extended very slowly

(if at all) for the first 60% to 70% of the life of the impacted specimens followed by an increasing growth rate to failure. In the 24-ply damaged hole specimens damage progressed at a substantial rate during the initial 20% to 30% of specimen life, slowing to a much lower value until near failure where the rate again accelerated until fracture occurred.

A typical S-N curve with less than an order of magnitude data dispersion was displayed by the 32-ply laminate containing a damage hole. Damage growth characteristics of this laminate were comparable to those observed for the 24-ply damage hole specimens with initial rapid growth followed by progression at a slower rate until near failure where the rate again accelerated. Results for the impact damaged 32-ply specimens did not, however, show consistent damage growth or S-N behavior with a large number of the failures occurring away from the damage region. This impact condition was apparently too near the threshold size to act as the dominant cause of failure under fatigue loading. In cases where valid failures were obtained, a growth pattern similar to that observed for the 24-ply impact damage specimens was evident.

An important observation which is a factor in the consideration of the significance of the damage growth data is the effect of the anti-buckling guide geometry. Careful study of the results indicates that for certain load ranges and laminate/damage conditions damage may extend at a stable rate to the boundary of the anti-buckling guide opening. At this point damage growth may be stopped or slowed due to the clamping forces exerted by the guide which in effect defines the limit of the velocity of the damage growth rate data.

A comparison of the TBE enhanced x-ray and Holskan ultrasonic methods of monitoring damage revealed similar damage sizes for the subset of fatigue test coupons. No significant change in static compression strength following TBE examination was discovered. Periodic TBE inspection also appeared to have no measurable effect on the subsequent fatigue behavior of the 32-ply laminate, but results were less definitive for the 24-ply laminate where there is some indication of a possible shortening of the fatigue life at lower stresses. However, the limited sample size is small enough that the apparent reduction could result from the inherent scatter under fatigue loading.

TASK II - Damage Growth and Residual Strength Degradation Prediction

The damaged hole condition was selected for further study with monitoring of damage development to be accomplished by use of the Holskan ultrasonic unit.

Static tension and compression strength distributions were determined for damaged and undamaged laminates at cross-head rates of 0.05 in./min. and 20 in./min. to determine the effect of the higher rate which is experienced in fatigue testing on the fracture stress.

Damage zone growth characteristics under static loading were examined by loading specimens to various percentages of ultimate, inspecting, then reloading to failure.

Twenty replicate specimens for each laminate were fatigue tested to failure at a single stress level ($R = -1$) with the damage growth for each specimen monitored a minimum of ten times during its life to determine the fatigue life and damage growth distributions and pertinent statistical parameters. Based on these results, five cycle levels were selected for the residual strength study.

Twenty-three specimens of each laminate were inspected, cycled to one of the five preselected N values and Holscanned again. Three of the replicates were destructively analyzed while the other surviving specimens were tested in static tension or compression. This sequence was repeated for each of the five N Values.

No change in modulus or shape of the stress-strain curve at the higher strain rate was noted for either of the damaged laminates under static tension or compression loading but there was a significant decrease in strength and failure strain for both cases at the higher loading rate producing a larger drop in compression than in tension. Damage growth studies under static loading revealed no significant change in the final failure properties as compared to the baseline tests.

Fatigue cycling of the 24-ply specimens at ± 35 ksi (241 MPa) ($R = -1$) produced data which were dispersed over nearly two orders of magnitude with a characteristic life of 209,000 cycles. Data scatter was slightly more than one order of magnitude for the 32-ply coupons tested at a stress level of ± 22 ksi (152 MPa) with a characteristic life of 113,000 cycles. Damage growth behavior of the 32-ply specimens appeared more well-behaved than that of the 24-ply and unlike the 24-ply laminate where damage height appeared to be the more consistent growth parameter, damage extension occurred primarily in the width direction. Residual static properties of either laminate do not appear to be affected by $R = -1$ fatigue cycling up to the 80% probability of survival life. A very slight but insignificant increase (6-11%) in tensile residual strength and similar decrease in compression was noted for the 32-ply laminate. Both tension and compression residual strength tended to increase slightly as the number of cycles completed increased for the 24-ply laminate. For neither laminate, no consistent trend in damage size with increasing number of fatigue cycles experienced could be discovered, due in part to the dispersion of the damage size data but also to the fact that no definitive change in residual strength was observed.

TASK III - Effect of Fatigue Loading/Environment Perturbations

The purpose of this task is to ascertain the effect of changes in the fatigue test conditions on: 1) fatigue life; b) damage growth behavior; and c) residual strength of initially damaged specimens from which data the range of applicability of current life prediction models may be assessed.

Three major variations in the fatigue test parameters were selected for evaluation:

- CASE A - Data from the first two tasks indicate that the damage growth behavior of a fatigue specimen subjected to considerable compression loading during fatigue will be a function not only of the normal load (stress and range ratio), frequency and environment but also of the method of stabilizing the specimen. Moreover, the fatigue life and residual strength are potentially functions of the test support method and geometry establishing this as a major variable for evaluation. The range ratio ($R = -1$), stress levels and environment were maintained as on Task II but a four-bar column buckling support with 1.8 inch (46mm) spacing replaced the fatigue platens.
- CASE B - Of the major loading variables the compressive stress component has been recognized as having a significant effect on both notched and

unnotched fatigue behavior. Thus, the primary variable preferred for isolation in this case was the influence of the compressive load portion of the cycle. Tests were conducted at a range ratio of $R = -0.3$ employing the same environment, stress range ($\sigma_{\max} - \sigma_{\min}$) and fatigue supports used in Task II.

- CASE C - Environment is another obvious variable which can be expected to influence fatigue behavior and can be important in assessing the applicability of analytical models. An environment of 180°F (82°C) dry laboratory air was selected for this case while all other conditions were maintained as in Task II.

The number of specimens per condition in this task is much smaller than in Task II, often too small to definitively assess the effects.

Three replicates per each laminate for each of the fatigue cases (A, B or C) were tested to discern the basic fatigue life and damage growth characteristics which were monitored at selected intervals during the life with the Holscan unit. From these results 3 cycle N values were selected and for each N level six replicates per laminate per condition (A, B or C) were then fatigued after which three were then tested in static tension and three in compression.

Static tension and compression tests also formed a part of this task to provide an initial strength distribution should any changes have occurred during shelf storage and to generate baseline data with the 4-bar buckling support and for the 180°F (82°C) temperature condition.

Although the constraint condition employed in Case A appeared to yield the same average compression buckling strength as with the fatigue guide it did result in shorter fatigue lives with correspondingly larger damage growth earlier in life.

Under the $R = -0.3$ Case B loading both laminates completed 2 million cycles without failure. Most notable was the change in damage development, especially for the 24-ply laminate for which essentially no growth in the width direction was evident with extensive growth in the length direction. This longitudinal damage growth reduced the notch effect resulting in significant increases in tensile residual strength with increasing number of cycles completed. Damage growth for the 32-ply laminate at $R = -0.3$ was also greater in the length direction, but growth in both directions did occur.

Although the sample size is small, there appears to be an order of magnitude decrease in life due to the 180°F (82°C) temperature exposure during cycling with more rapid initial damage growth.

DESIGN SPECTRUM DEVELOPMENT AND GUIDELINE HANDBOOK

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The objective of this program was to evaluate the effect of fighter wing load spectrum variations on the life behavior of composite structures. With the introduction of composite materials into aircraft structures, a number of questions have arisen concerning their durability under the variety of loading conditions found in a fleet of multi-mission military aircraft. The impact of load history variations upon composite structure durability was not well known, although these variations are of significance with metallic structures. This program was performed to quantify and evaluate these potential effects in realistic composite structures, in order to develop minimum load spectrum requirements to adequately evaluate structural durability in both the design and test stages. The program was performed in five phases.

In Phase I, Load Sequence Generation, eleven spectra were generated using the procedures given in References 1 and 2, which were derived from load factor spectra for the F-15 aircraft. The upper wing skin of the F-15 aircraft was used as a basis to convert the load factors to stresses. The cycle-by-cycle stress histories were generated using digital techniques based on random noise theory, as described in Reference 1. Modifications of these time histories used to create the spectra variations were developed using the procedures described in Reference 2. The spectra variation types considered were: (a) clipping to 90% test limit stress, (b) addition of stress overloads, (c) addition of low loads to match the original measured mix, (d) truncation to 70% test limit stress, (e) clipping of tension loads, (f) increased severity and number of air-to-air loads. These spectra variations are shown in Figures 1-3.

In Phase II, Test Specimen Design and Manufacture, a simple single hole compression test specimen was designed to simulate fatigue critical areas of fighter wing structure. Two different lay-ups: (a) fiber dominated (48/48/4), (b) matrix dominated (16/80/4), (% of 0° plies, % of +45° plies, and % of 90° plies) were selected. Fiber dominated lay-ups (laminates with a high percentage of 0° plies) are representative of wing fatigue box skin areas where the design is strength controlled. Matrix dominated lay-ups (laminates with a high percentage of +45° plies) are used in stability critical structural components such as fixed trailing edges. Both the fiber and matrix dominated laminates were fabricated with 25 plies of .0104 inch nominal thickness graphite-epoxy prepregs. Hercules AS/3501-6 graphite-epoxy was the material system.

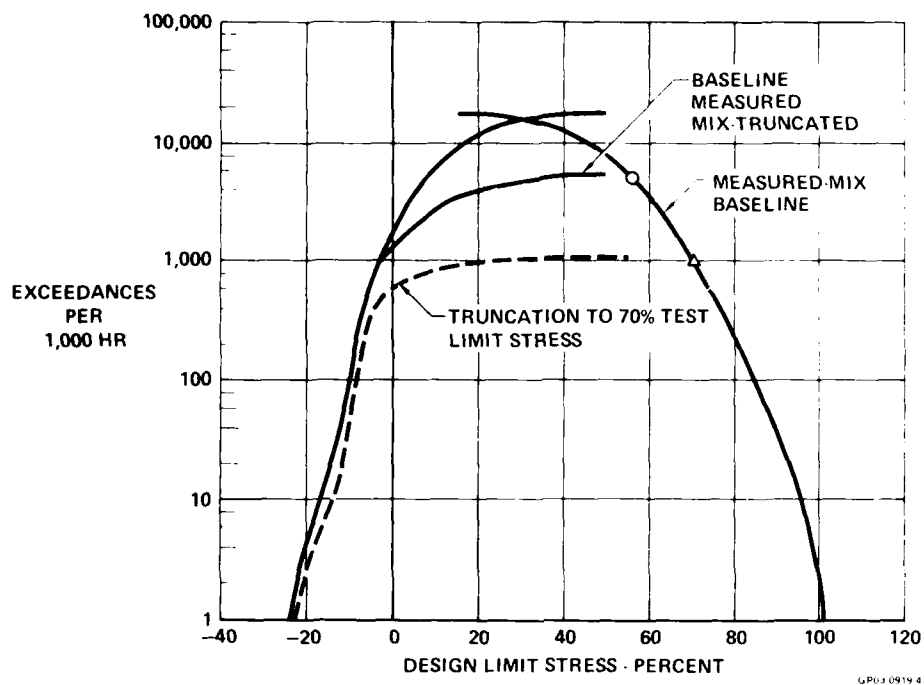


Figure 1. Exceedance Curves for Truncation Variations.

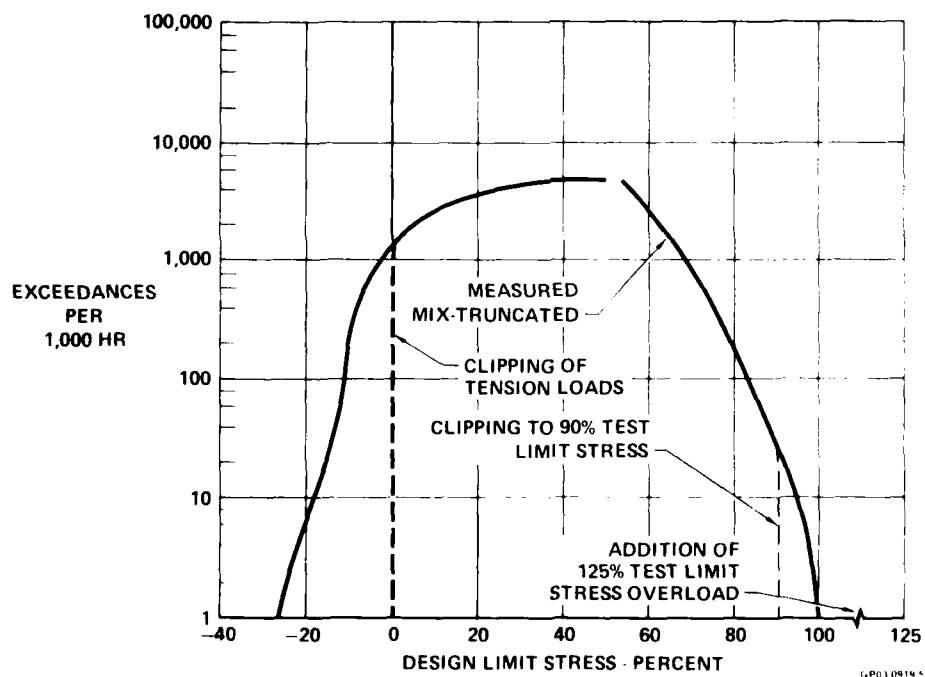


Figure 2. Exceedance Curves for Clipping Variations.

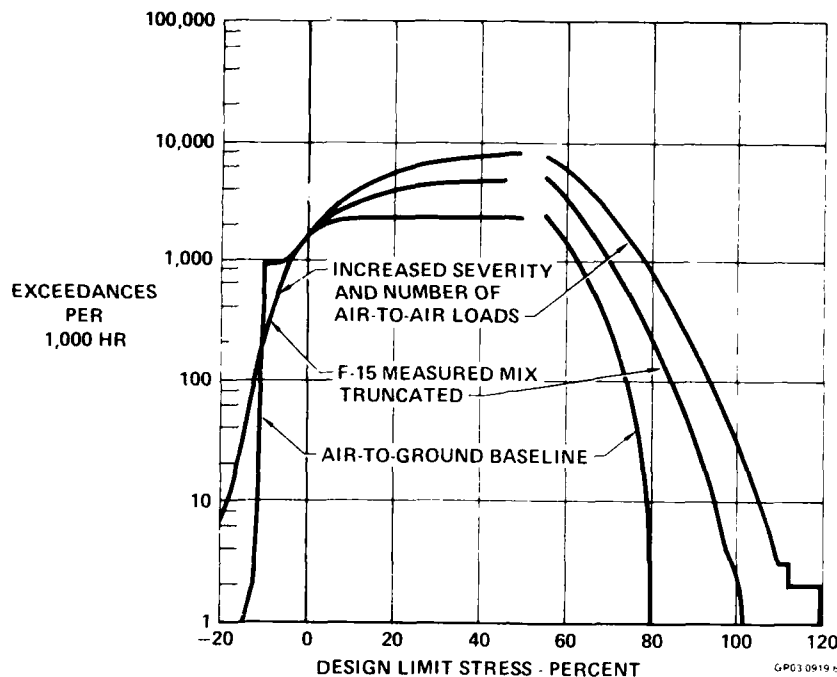


Figure 3. Exceedance Curves for Usage Variations.

In Phase III, Analytical Life Predictions, the spectrum fatigue life for each of the variations was predicted. These predictions were made by using constant amplitude fatigue data, a correlation parameter based on the concept of strain energy density factor for micro-cracks in the laminate matrix, an empirical modification to account for stress ratio effects on life, and linear residual strength reduction (Figure 4) fatigue damage model.

In Phase IV, Experimental Verification, 36 constant amplitude and 177 spectrum tests were performed. The purpose was to evaluate the effects of spectra variations on life, and to provide data useful for defining guidelines for structural verification of future aircraft. Figure 5 summarizes the baseline measured mix truncated test data for the two lay-ups. These data demonstrate scatter that can occur in composite laminate testing. Because this scatter makes the selection of typical lives difficult, Weibull statistical analyses was performed to determine the test mean life. The Weibull test mean life of the Baseline Measured Mix-Truncated is shown in Figure 5 for the two lay-ups at three different load levels. Figure 6 is a comparison of the Weibull mean test lives with predicted lives. Figure 7 is the comparison of predicted and measured effect of spectrum variations on life, normalized with respect to the Baseline Measured Mix-Truncated life.

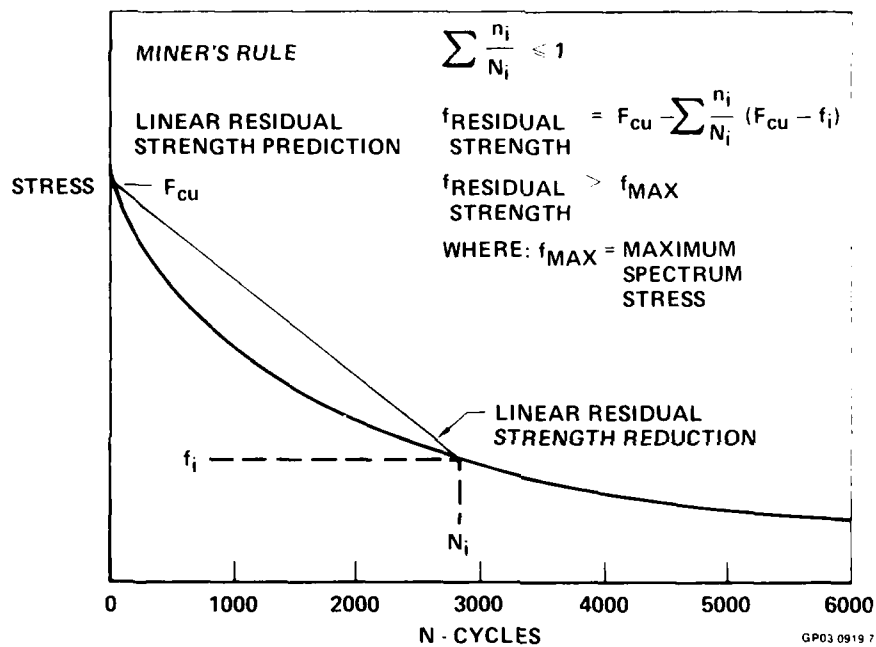


Figure 4. Linear Residual Strength Reduction Fatigue Model.

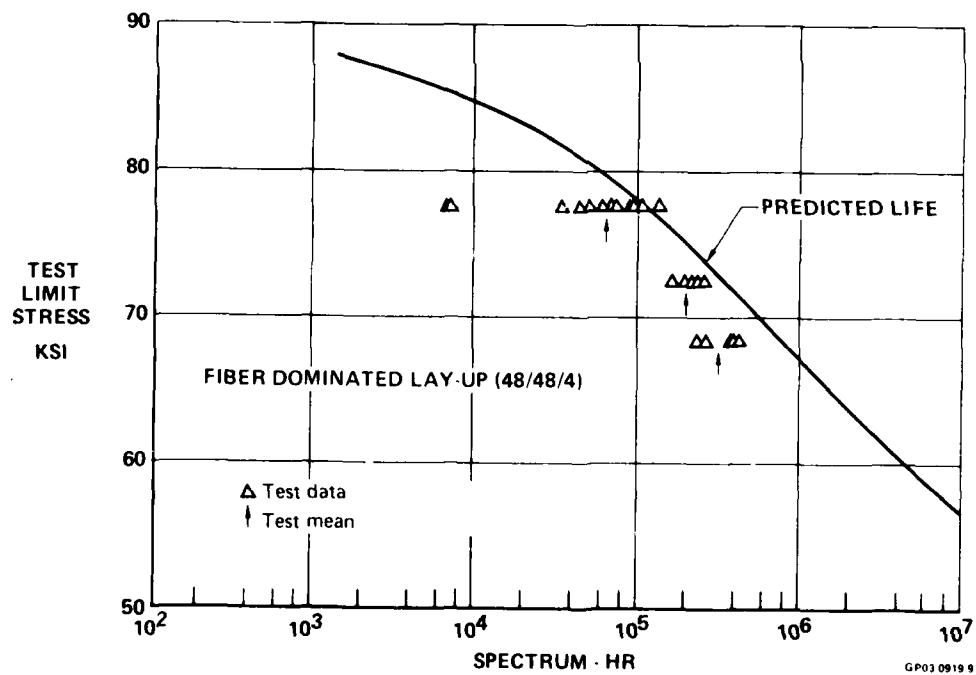
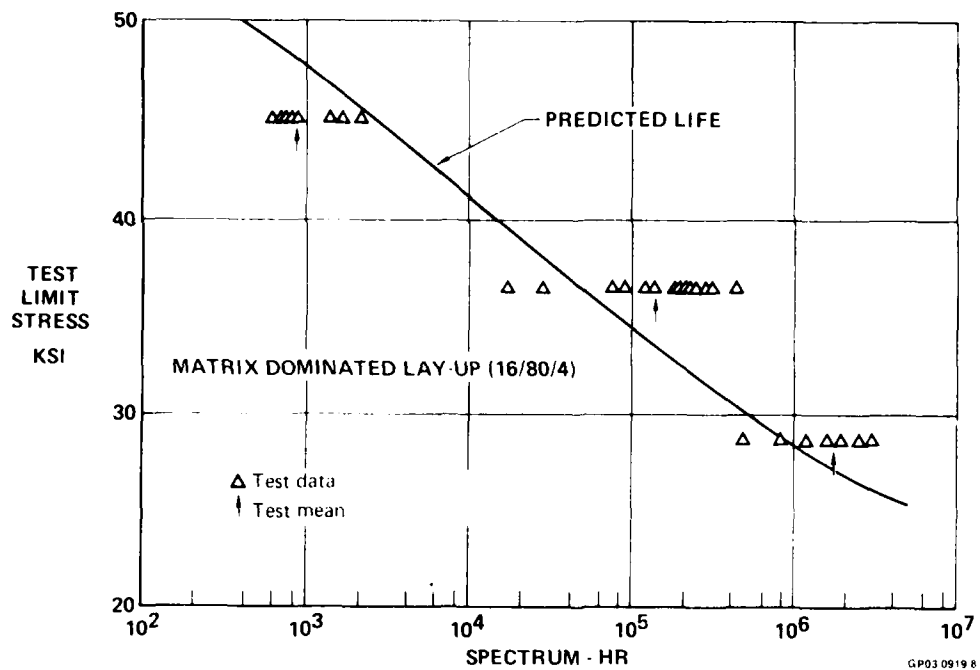


Figure 5. Experimental and Predicted Lives for Baseline Measured Mix Truncated Spectrum

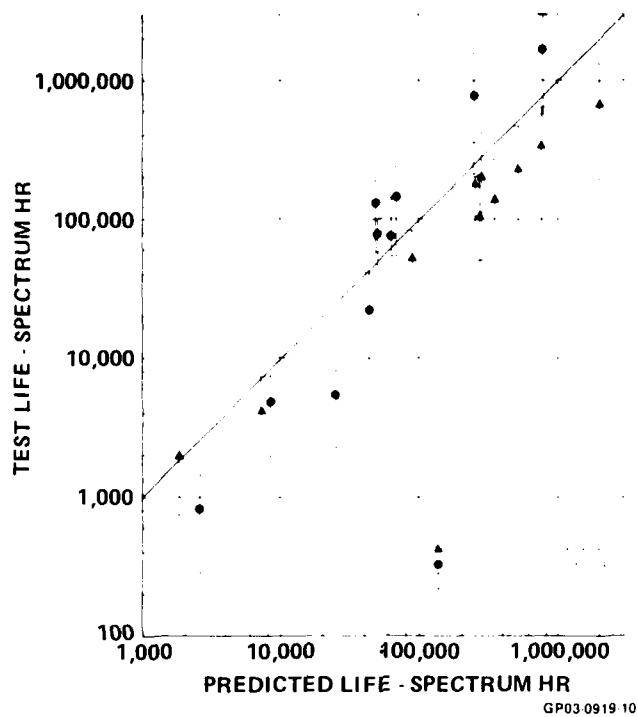


Figure 6. Comparison of Predicted and Test Lives.

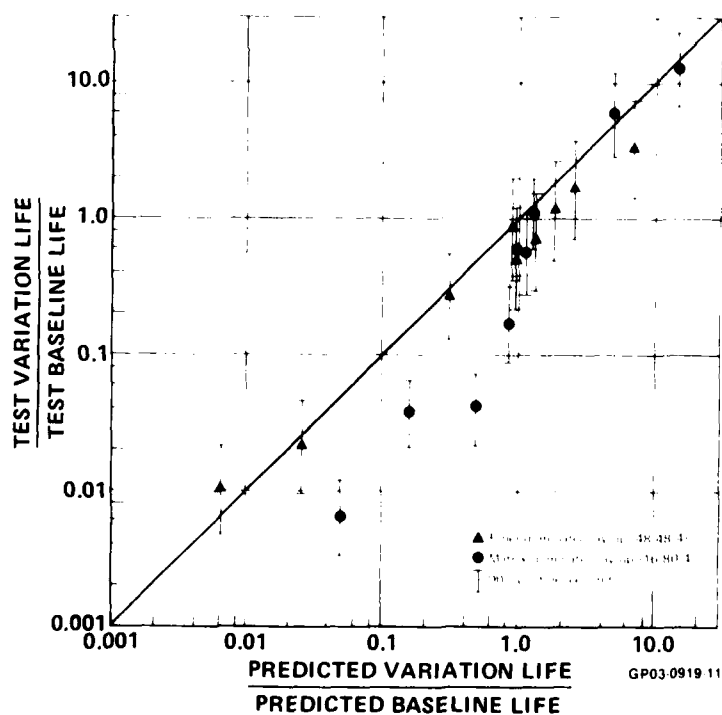
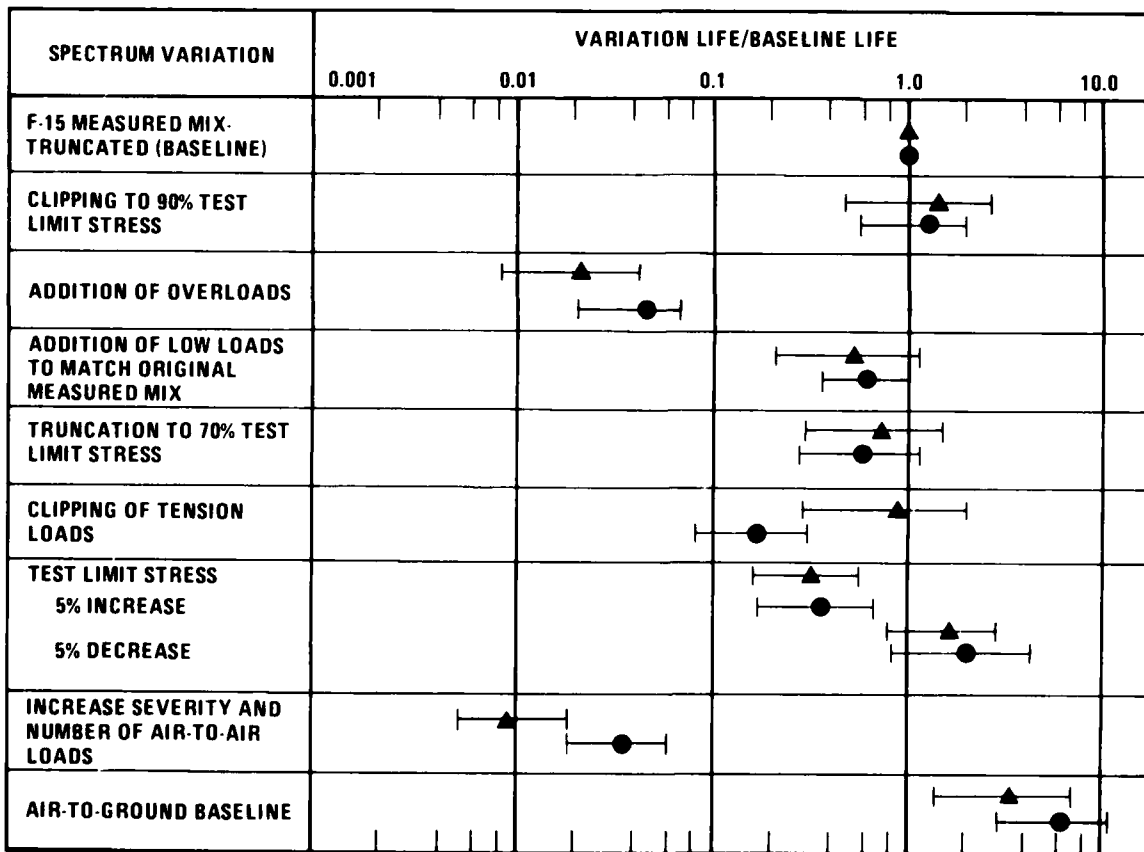


Figure 7. Comparison of Predicted and Measured Effect of Spectrum Variation on Life.

In Phase V, Recommendations and Guidelines, the experimental data were evaluated and summarized. The impact of spectrum variations upon the fatigue life of composite laminates is summarized in Figure 8. In this figure, test mean lives are normalized with respect to the baseline life - the life developed with the F-15 Measured Mix-Truncated. The variations found to have the greatest effect were those that increased the frequency or magnitude of the high loads in the spectrum. Addition of Overloads, and Increased Severity and Number of Air-to-Air loads caused more than an order of magnitude reduction in life. A 5% increase in test limit stress caused 60% decrease in life. Variations found to have smaller impact on life were those that change the lesser loads in the spectrum. These variations are the Addition of Low Loads to Match the Original Measured Mix, Truncation to 70% Test Limit Stress, and Clipping of Tension Loads.



▲ Fiber dominated lay-up (48/48/4)
 ● Matrix dominated lay-up (16/80/4)
 ┌─┐ 90% confidence limits

GP03 0919 12

Figure 8. Effects of Spectrum Variation on Life.

Recommendations and guidelines for deriving design and test spectra for multi-mission fighter aircraft are being developed.

ACKNOWLEDGEMENT

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FATIGUE SPECTRUM SENSITIVITY STUDY FOR
ADVANCED COMPOSITE MATERIALS

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Analysis and testing of primary composite aircraft structure for certification is hampered by the lack of suitable techniques for simulating real time environments in the laboratory. Decisions are generally made on the types of spectrum load, environment, and time simulation in fatigue testing for the sake of expediency and without a rational basis for measuring the effect of these decisions against actual fatigue performance in real time. This problem is particularly compounded in the case of high performance fighter aircraft where the combined effects of high flight temperatures (up to 250°F) high heatup rates (90°/min or more), minute amounts of absorbed moisture, and high loads occurring frequently during combat maneuvers may be detrimental to certain types of composite structures. For instance, it is well known that certain hygrothermal conditions can reduce the strength of composite structure in substantially different ways depending on the temperature and moisture absorption values and the manner in which they are combined in a laminate. Durability test methods must take this into account. The standard practice in metallic fatigue testing of applying loads as rapidly as possible within the test equipment limits may not be acceptable in composite structural testing because of these affects. This would leave real time testing as the only acceptable approach in some cases.

This Air Force sponsored program studied the effects of various aspects of load and environmental realism in fatigue testing of composite joints. An accelerated test method has been developed that combines in various ways typical fighter aircraft flight load and temperature spectra, and the effects of a 20 year life cycle ground storage moisture environment. Starting with temperature exceedance data, a distribution of discrete temperature values is determined based on the proportion of times a given value occurs in real time. These discrete values are then arranged in a Lo-Hi-Lo block representing 1/8 of a lifetime (Figure 1), and are applied over the same test time span required for the accelerated flight-by-flight load application. This method has the advantage that temperature ramping time is reduced significantly, adding only about 15 percent of the test time required without temperature (at a 50-degree/minute heatup/cool-down rate and 5 Hz load application frequency). While the blocked temperature and the superimposed flight-by-flight loading are random, the distribution of load

BLOCK LENGTH = 408 MISSIONS OR 1/8 LIFE

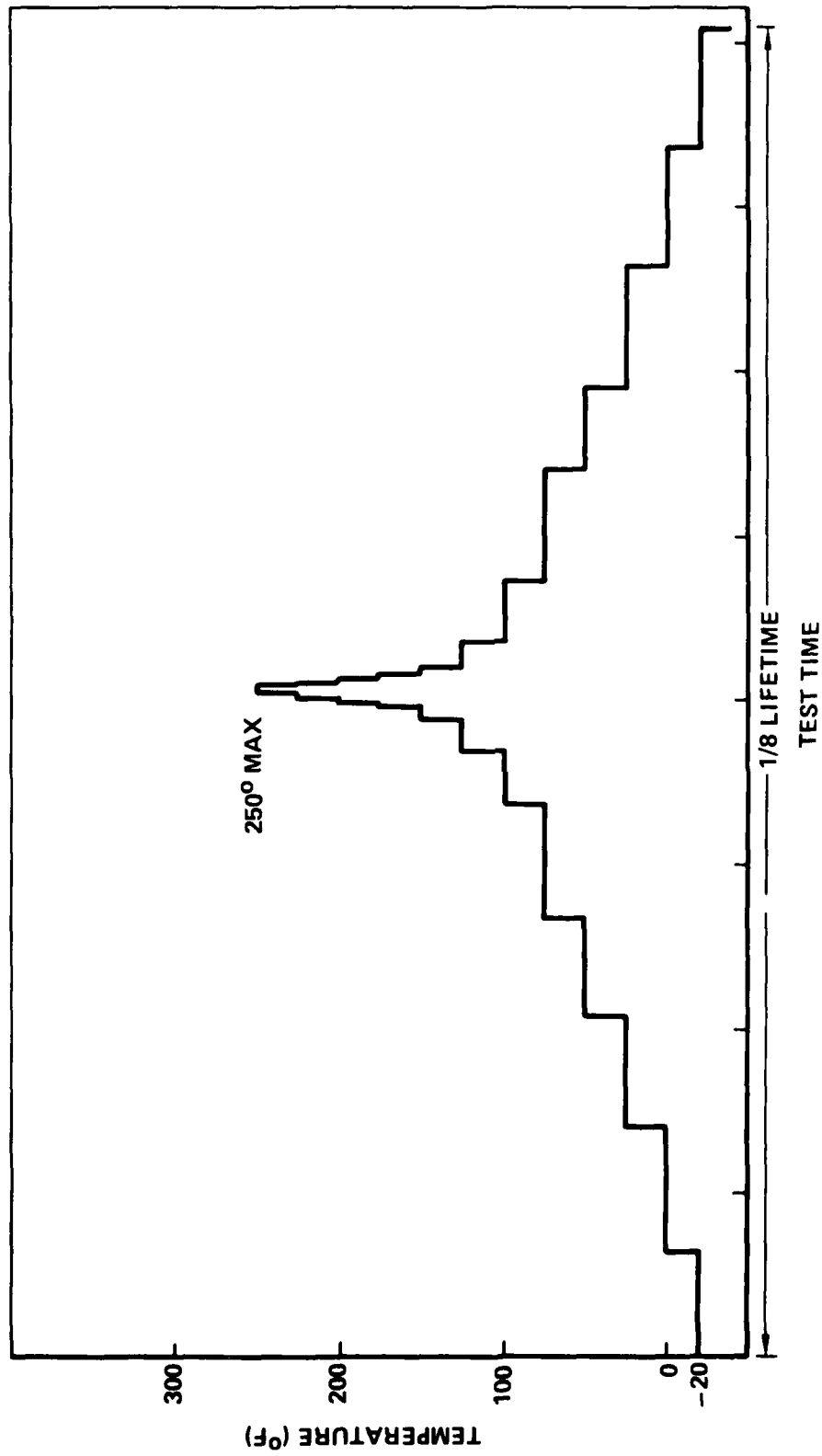


Figure 1. Temperature Versus Numbers of Missions

magnitudes to temperature levels is reasonable with loads as high as 71 percent of the max spectrum occurring at 200F and as high as 60 percent at 250°F. This method represents a possible lower cost approach to replicating the real time fatigue test in failure modes and durability measured by residual static strength degradation. This methodology is used in an extensive experimental study of the load and environmental sensitivity of composite-to-metal bolted and step-lap bonded joints. This study evaluates the most frequent assumptions made in load truncation, RMS load level, load frequency content, test temperatures, moisture contents, and mission mix, in development of test spectra to simulate actual flight time fatigue effects. Both accelerated and real flight time tests are conducted on composite joints. Information is developed on the effects of spectrum power spectral density content and sustained load maneuvers on joint residual strength by varying the loading rates and peak load dwell times.

A total of 104 fatigue test series, and 12 static test series (without fatigue exposure) were completed. Also residual strength tests were run on the approximately 80% of the fatigue specimens that survived 2 lifetimes of fatigue (or 15 lifetimes in the case of extended lifetime tests). All test series contained 20 specimens per series except the real time tests and bolted joint extended lifetime tests which were 10 specimens each, and baseline test series 3 and 15 which were 40 specimen each (Table 1).

A 20-year aircraft life cycle moisture model is developed and simulated in the test specimens by accelerated moisture conditioning. Extensive moisture control travelers (MCT's) were used during fatigue testing to monitor moisture gain/loss (Figure 2). The travelers were made up to represent the solid laminate and the three laminate plus bondline sections that occur in the bonded joints.

The two-parameter Weibull distribution is used for statistical analysis of bolt strength and fatigue life data. Two methods are used to estimate the population shape and scale parameters: (1) the maximum likelihood estimate (MLE) for censored fatigue data (tests with only a partial number of fatigue failures) and uncensored fatigue and residual strength data; (2) the least squares (LS) method for residual strength data from test series with early fatigue failures.

Within each environmental type (RTD, RTW, MPTW)* parametric test data are compared with baseline data. It is shown that these spectrum effects are much more prominent in the step-lap bonded joints than the bolted joints. Loading waveform, time of load, and real time parameters are demonstrated to have potentially harmful effects on residual strength and fatigue life of matrix or bondline dominated joint designs. It was further shown that batch-to-batch variations in materials and fabrication panel quality can mask these effects if proper quality control destructive and non-destructive test methods are not used to isolate and correct for

TABLE 1. TEST CONDITION AND TEST SERIES MATRIX

TASK	ENVIRONMENT	DURATION (LT)	LOAD RATE	FREQUENCY (Hz)	TRUNCATION	RMS	STATIC	BASELINE	FREQUENCY EFFECTS	TRUNCATION EFFECTS	STRESS LEVEL EFFECTS		EXTENDED LT EFFECTS
											K ₁ X STANDARD	K ₂ X STD	
I	RTD	BONDED & BOLTED	T C T	5	9/2	STANDARD	STANDARD	STANDARD	7.33/2	10/2	9/1	STD (9/2)	STD (9/2)
IIA	LTV	BONDED & BOLTED	T C T	5	9/2	STANDARD	STANDARD	STANDARD	7.33/2	10/2	9/1	STD (9/2)	STD (9/2)
IIB	MPTW	BONDED & BOLTED	T C T	5	9/2	STANDARD	STANDARD	STANDARD	7.33/2	10/2	9/1	STD (9/2)	STD (9/2)
IIB	RTD, RTW, MPTW	BONDED & BOLTED	T C T	5	9/2	STANDARD	STANDARD	STANDARD	7.33/2	10/2	9/1	STD (9/2)	STD (9/2)

Batch Effect Characterization														Repeat Test Series 3-1, 107-1, 109-1, 131-1, 149-1, and 10A - Repeat Test Series 15-1													
Aircraft Usage and Specimen Characteristics														Test Series 77, 78, 79 and Large Scale Test Series 76													
APPLICABLE														Large Scale Test Series 72													

☒ BASELINE SPECTRUM (SAME FOR ALL TASKS)

2 SEE TABLE 5 FOR MULTIPLICATION FACTORS (K_1 AND K_2)

3 NUMBERS SHOWN DENOTE UPPER AND LOWER POSITIVE LOAD FACTOR RANGE (E.G. 9/2=9g UPPER, 2g LOWER)

☒ THESE SPECIMENS TO BE HTW ONLY (NO MISSION PROFILE

✓5 DENOTES TEST SERIES NUMBERS; ALL TEST SERIES 20 SPECIMENS PER SERIES EXCEPT 9, 13, 21 106, 1111

111C, 119, 134T, 134C, 141, 150-10 SPECIMENS PER SERIES, 3, 15-40 SPECIMENS

PER SERIES.

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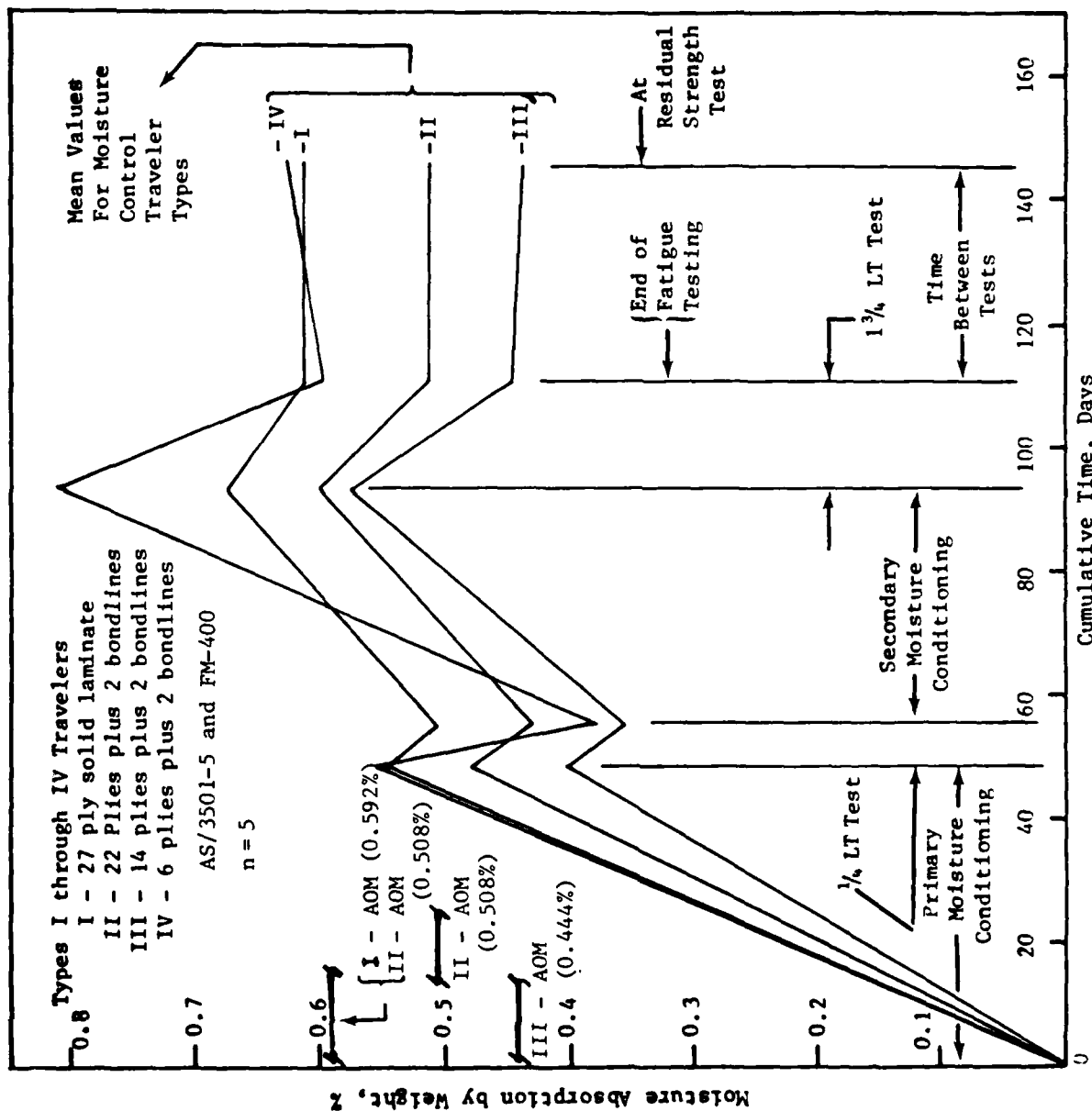
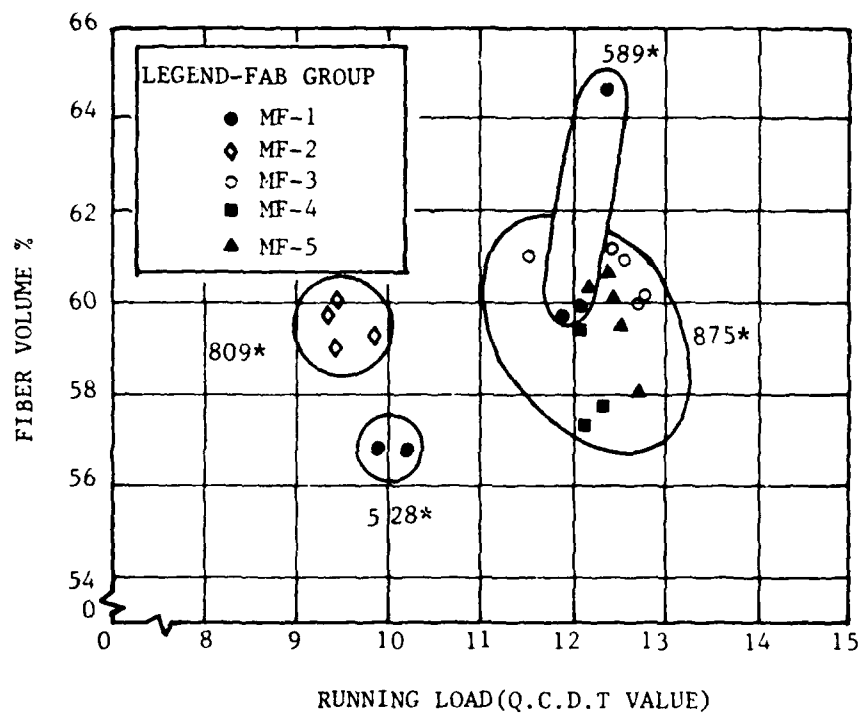


Figure 2. Types I Through IV Traveler's Moisture Absorption Versus Time for Test Series 135 (NPTW).

these variations (Figure 3). Almost 3300 specimens were fabricated from approximately 150 composite panel assemblies containing 20-24 specimens each. During this production run over a period of about four years extensive quality control procedures were used. Thus, any variations in specimen quality were discovered and corrective action taken to either repeat the test, scrap the panel, or adjust the data reduction techniques to compensate.

The effects of both moisture and temperature on bonded and bolted joint strength degradation are discussed. Guidelines for using accelerated fatigue testing and certification of fighter aircraft composite joints to satisfy requirements of cost effective real time/real environment simulation are presented.

- * RTD - Room temperature dry
- RTW - Room temperature wet
- MPTW - Mean profile temperature wet



*Denotes Prepreg Batch

Figure 3. Bolted Joint Quality Control Destructive Test Data vs Fiber Volume

COMPOSITE WING/FUSELAGE PROGRAM

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INTRODUCTION

The overall objective of the program is to demonstrate structural integrity and durability of composite wing/fuselage primary structure. The two main specific program objectives are to develop low cost durability validation testing procedures, and to develop and verify durability design methodology. The secondary objectives are to validate generic detail design concepts, to exercise innovative manufacturing techniques in a production environment, and to develop a detailed cost data base for future cost predictions. These goals are achieved by selecting test specimens from the wing/fuselage structure addressed in the preliminary design phase, fabricating the required replicates of these specimens, and performing testing with various combinations of test parameters to establish a data base. This data base is used to validate the structural design concepts, develop durability design methodology and establish low cost durability qualification procedures. This paper discusses three critical aspects of the Wing/Fuselage Program: (1) selection of candidate accelerated test schemes, (2) durability test strain level philosophy and (3) durability design methodology.

SELECTION OF ACCELERATED TEST SCHEMES

To meet the objective of low cost validation testing procedures, four accelerated test schemes are being evaluated against a real flight time test scheme. The Baseline Accelerated Scheme shown in Figure 1 is designed to replicate the real time test scheme as faithfully as possible (within the constraints of a 300 hour/lifetime test time) and is the most sophisticated accelerated test scheme. The essential features of the baseline scheme are:

- o The temperature spectrum is divided into forty-six equal length blocks. Each block contains nine high amplitude thermal cycles, one cycle from 250°F to -20°F and the remaining eight from 215°F to -20°F.
- o The temperature profile is divided into 3 load application zones, which are indicated by solid lines in Figure 1.

- Loads corresponding to high temperatures are applied between 185°F and 250°F at a frequency of 0.08 Hz.
- Loads corresponding to low temperatures are applied between 50°F and -20°F at 1 Hz.
- The remaining loads are applied at the weighted mean temperature of 145°F at 1 Hz.
- Four hundred and fourteen such temperature/load cycles constitute one lifetime.
- Moisture absorption is simulated by partial preconditioning of the test specimens to 0.75 end of lifetime moisture level (ELTM) followed by reconditioning to ELTM after 60 percent of one lifetime. During the test period moisture loss due to thermal spiking is recouped by third shift and weekend reconditioning.

The rationale for the three other (alternate) accelerated test schemes is as follows. The alternate schemes should be more cost effective than either the baseline accelerated or real time tests, which implies a reduced complexity of load/environment simulation and a decreasing order of complexity of temperature and moisture environment simulation.

The first alternate is shown in Figure 2. In this scheme, discrete temperature levels are arranged in a Lo-Hi-Lo block for a predetermined fraction of the lifetime. The Lo-Hi-Lo block profile is determined by the proportion of times a given temperature value occurs in real time. The loads are applied flight-by-flight, and thus the sequence is maintained. The temperature spectrum consists of high amplitude, low frequency thermal cycles, with no attempt made to match loads and temperatures. Loads are applied at accelerated frequencies for temperatures below 185°F, and at 0.08 Hz for temperatures above 185°F.

The second alternate accelerated scheme is shown in Figure 3. It is a simplified version of the accelerated baseline scheme where the freeze-thaw cycles are reduced to 46 per lifetime. As in the case of the baseline scheme the number of thermal spikes above 200°F per lifetime equals that in the real-time spectrum. Loads are segregated by temperature segments so as to ensure load/temperature correspondence above 185°F. The test specimens are moisture preconditioned to ELTM levels followed by periodic reconditioning on 3rd shifts and weekends.

The third alternate is shown in Figure 4 and is the least complex by way of environmental simulation. The scheme consists of simply applying a flight-by-flight load spectrum at a predetermined constant temperature of 145°F. In this scheme, the load sequence is maintained, and the specimens are moisture preconditioned to ELTM and then intermittently reconditioned during the test.

DURABILITY TEST STRAIN LEVEL PHILOSOPHY

In certification clearance, the fatigue test load spectra is related to the ultimate design load irrespective of the relationship of the static test failure load to the ultimate design load. However, the objective of the Composite Wing/Fuselage Program is not to certify a specific composite structure, per se, but rather to set the durability test strain levels such that the test data are optimized for screening of alternate test schemes and a retrospective comparison of the real flight time test to the accelerated test schemes. The durability test strain levels must, therefore, be related to some function of the static failure load rather than the ultimate design load.

To help in durability test strain level selection, a re-examination of the conventional zero margin of safety design concept (based on the design allowable being equal to 0.8 x average test failure value) was conducted. Firstly, it appeared that there was little hard data to justify the past use of an across-the-board 80 percent of average for design values. This value appears to have been used because it is considered to be "conservative." To check the validity of this factor the design data used in the Wing/Fuselage Program were examined in detail. For each data set the ratio of Weibull B-Basis value to average test value was calculated. The test data showed that: the mean B-Basis/Average ratio for the 56 design data sets was 0.86. The design value for Wing/Fuselage Program fatigue testing was therefore selected as 0.86 x the average test value. The upper truncation load level for the Wing/Fuselage Program is 125 percent of limit load (9.16g) with no low load or negative load truncation. The Wing/Fuselage Program test load levels are:

Load	% Average Test Failure Load
Ultimate Design Load	86.0
Maximum Fatigue Load	71.6
Limit Load	57.3

DURABILITY DESIGN METHODOLOGY

The objective of this work is to develop durability analysis procedures which are suitable for use in practical design applications.

Three published analytical models have been selected for study: (1) a cumulative damage rule, (2) a deterministic model based on the observation of delamination growth as the dominant compression fatigue failure mechanism, and (3) a probabilistic life prediction technique which uses a data pooling method that does not require a prohibitively large data base. As needed, these models have been extended and modified to increase their versatility and reduce their complexity in design applications. In addition, environmental fatigue life prediction methodology is being developed which takes into account load/temperature/moisture relationships in the fighter aircraft environments. Two approaches are being used: a simple engineering approximation based on the Sandeckyj's model and a micromechanics analysis procedure.

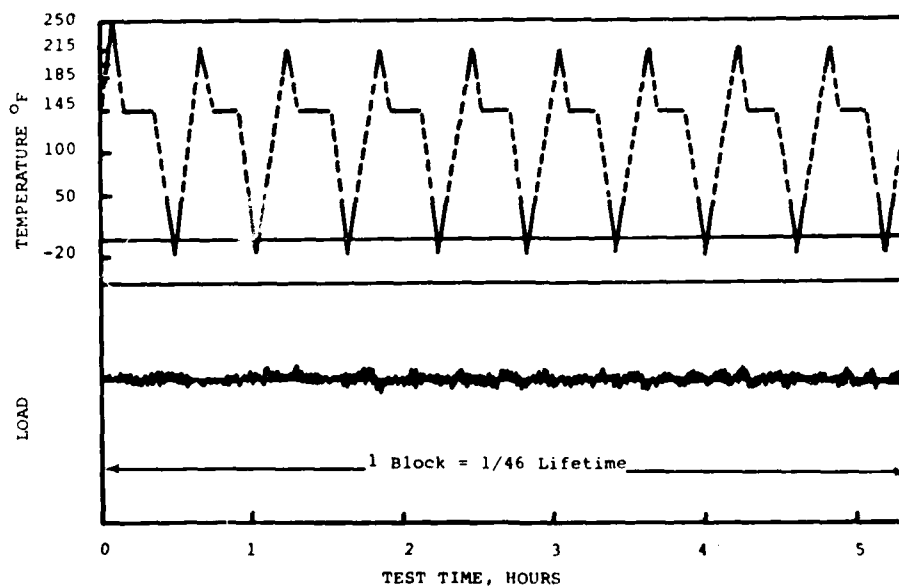


Figure 1 Baseline Accelerated Test Scheme

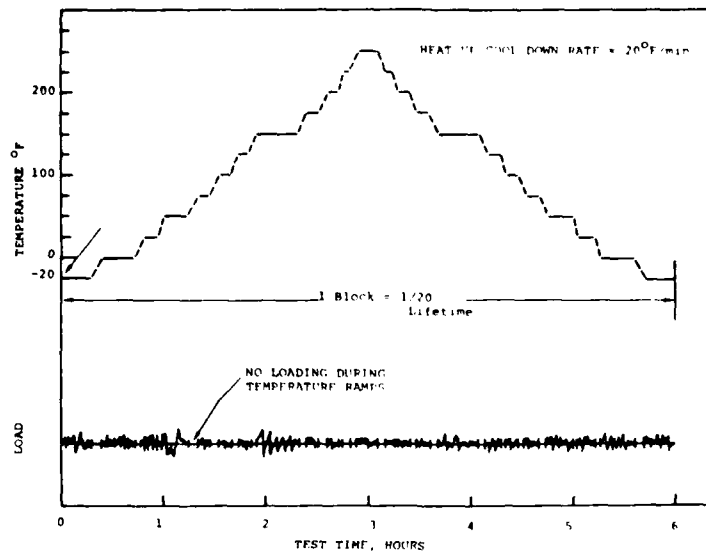


Figure 2 Alternate Accelerated Scheme No. 1

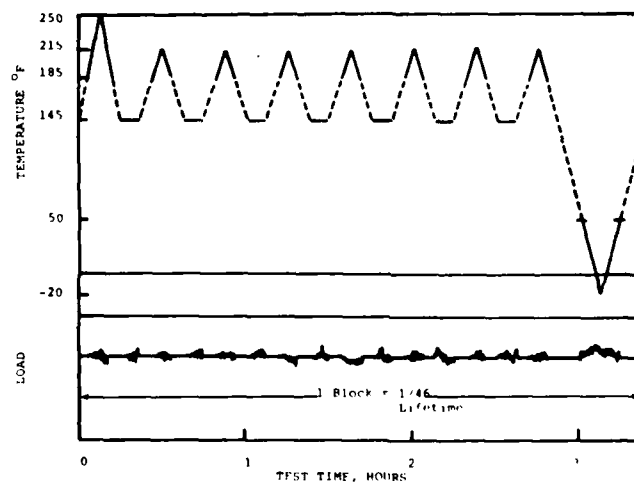


Figure 3 Alternate Accelerated Scheme No. 2

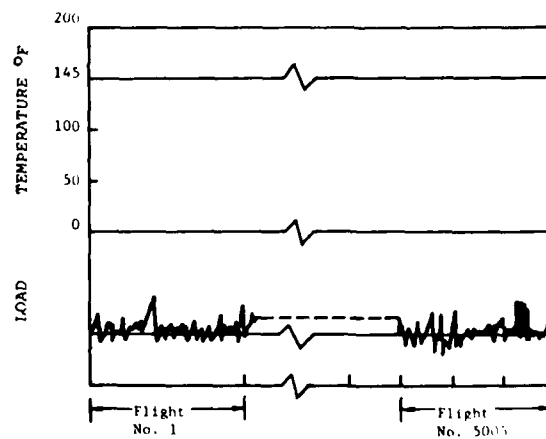


Figure 4 Alternate Accelerated Scheme No. 3

THE EFFECT OF LOAD
HISTORY ON FATIGUE LIFE

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During the past decade the increased use of composite materials in aircraft and their proposed use for primary structural applications has introduced a greater need for development of predictive capability for fatigue life and strength changes. To accomplish this goal, an understanding of the effects of various load histories on the fatigue life of composites is essential. The achievement of this understanding necessitates an awareness of damage mechanisms and a comprehension of their relationship in order that at least a qualitative prediction of load history effects can be formulated. For qualitative understanding of laminated composites up to the onset of 0° fiber breakage, the underlying damage mechanism which needs to be considered is matrix cracking influenced by the anisotropic nature of the material (inter and intralamina cracking). Thus, in formulating life and strength prediction techniques, attention must center upon gaining an understanding of the cause of and driving mechanism behind crack initiation and extension. This understanding may need to be combined, for certain applications, with a detailed study of fiber breakage in 0° plies.

Associated with the difficulties in defining the nature of damage growth and the ascertainment of associated failure modes for laminated composites, are the added considerations of: 1) developing and selecting adequate inspection methods to detect and monitor damage; 2) developing analysis methods to define the severity of damage; 3) developing adequate terms and methods for classifying failure modes; 4) determining the state of the stress/strain/energy field within the material. This program necessarily addresses these problems in the course of determining some of the effects of load history on the fatigue life of a graphite/epoxy composite laminate. The primary objectives of the program are to: 1) study in detail how mechanical loading parameters affect the life of graphite/epoxy laminates; 2) determine the effects of environmental and geometrical perturbations on the effects studied under objective 1; and 3) analyze the results in a manner which gives at least a qualitative understanding of the experimental results. The analysis is being conducted in a manner such that a foundation is laid for formulating fatigue life models based on knowledge of failure mechanisms.

A point of view as to the nature of damage initiation and growth for laminated composites has been formulated during this program as follows:

- The dominant type of damage of analytical concern for descriptive and mathematical analysis is matrix cracking.
- Matrix cracking can be intralamina (transverse) or interlamina (delamination).

- The mechanism of crack extension (atomic debonding) can be conveniently classified (for mathematical formulation purposes) as being due to: load increase, creep, and cyclic cracking.
- The damage state at any time is deterministically related to the initial state.
- The damage state at failure is dependent on the loading path.

This point of view leads to several deductions, among which are:

- The laminate stacking sequence determines the inter and intralaminar normal and shear stresses which, in turn, determine: location and eventual density of intralaminar cracks; location and propagation path of delaminations. These characteristics can be calculated, at least approximately, by known mathematical procedures.
- Final fracture must be ascertained by determining the manner in which load is transferred to the 0^0 plies. This requires analysis of the influence of local delamination on the fracture of 0^0 fibers and analysis of the effect of such fiber fractures on 0^0 ply integrity and coupon failure behavior.
- Temperature, width, length, and notches affect both static strength and fatigue life because they alter either matrix properties or stress state or both.
- Stiffness changes with time during fatigue cycling.
- Fatigue life depends on the definition of failure. If failure is defined as an amount of stiffness change, fatigue life will be much different than if defined as the number of cycles to a particular crack damage state or as fracture of the specimen into two or more pieces.

The point of view outlined above allows the results of the planned experiments to be related to the program objectives. The formulated point of view is based upon a qualitative analysis of the nature of damage initiation and growth in laminated composites. The effect of a general loading history depends upon the three identified mechanisms of crack extension. Hence, the relative dominance of these mechanisms must be assessed to understand the effect of any general load history. The relative importance of the three mechanisms and how they influence the damage accumulation and failure process needs to be assessed. The experiments planned in Task II and III of this program will discriminate among these mechanisms. In Task II, five different load histories will be evaluated: progressive loading, time under load, block fatigue, preload, overload.

TASK I - Diagnostic Experimentation (1978-1979)

One laminate of T300/5208 graphite/epoxy material is being used in this program. The laminate is a 16-ply quasi-isotropic layup of the following configuration: $(0/+45/90/-45_2/90/+45/0)_S$. This material and laminate were selected to provide maximum continuity in the formulation of a comprehensive data base since this material and laminate are the same ones for which extensive fatigue and static test data is being developed under AFML Contracts F33615-77-C-5140 and AFFDL Contract F33615-77-C-3084.

Material lamina properties were determined under three environmental conditions: 1) room temperature, dry; 2) 180°F, dry; 3) 180°F, wet. Static tests to obtain strength and modulus were conducted at each of these conditions using five (5) different loading conditions of 16-ply laminates: 1) 0° unidirectional tension; 2) 90° unidirectional tension; 3) +45° tension; 4) 0° unidirectional compression; 5) 90° unidirectional compression. In addition, thermal and moisture diffusivities, equilibrium moisture contents, and expansional strains due to temperature and humidity were obtained. Laminate fatigue properties were determined under constant amplitude loading at a frequency of 10 Hz. A stress-life scan was conducted consisting of tests at five (5) stress levels and four (4) load ratios ($R = -1, -0.5, 0, +0.5$) chosen such that any dominating effect of maximum stress or stress range on the fatigue life can be discerned.

As expected, 0° unidirectional tensile properties were found not to be significantly affected by environment. Compressive properties decreased by ~20% in the presence of high temperature, but only when coupon moisture content was high. The stress-strain curves of tensile loaded 0° coupons were non-linear displaying a non-Hookean, increasing curvature consistent with other experimental observations. High temperature was found only to affect the tensile properties of those 90° unidirectional coupons which had a high moisture content. The 90° unidirectional compression properties decreased by ~15% due to high temperature, but were not further affected by high moisture content. Shear stress properties were affected by both temperature and coupon moisture content. Static tension data of quasi-isotropic laminates obtained at a loading rate of 0.01 mm/mm/min. displayed a low coefficient of variation (3.5%) and high Weibull exponent (34) indicative of low scatter. This was due to the elimination from the data set of those coupons containing adjoining tape edges called line discontinuities. The average tensile strength of quasi-isotropic coupons tested at a loading rate of 6 mm/mm/min. was ~8% lower than those tested at 0.01 mm/mm/min. The cause of the reduction was hypothesized to be due to the manner in which matrix cracks propagate at the different strain rates. High strain rate compression test results for quasi-isotropic coupons were not significantly different than low strain rate results except that data dispersion was greatly increased.

The fatigue properties of this laminate were not significantly affected, at room temperature, by epoxy resin type or by batch fiber properties. Fatigue lives of coupons cycled at $R = +0.5$ were one to two orders of magnitude longer than those at $R = 0.0$ when compared on the basis of σ_{max} . This indicated that both the maximum stress and stress range significantly affect fatigue life. Based upon the tension-compression results and analytical concepts, a different type of constraint fixture is being used in Tasks II and III. A limited residual strength, experimental study of unfailed fatigue coupons supported the conclusion reached on the basis of fatigue data results that damage is extended by fatigue cycling and not just by a creep mechanism occurring at maximum load.

NDI techniques were considered and evaluated based upon four criteria that a selected procedure was expected to meet: 1) be reproducible; 2) locate damage regions; 3) differentiate among the three damage mechanisms of matrix cracking, ply delamination and fiber breakage; 4) indicate expected failure locations. Based upon a literature review and RFP requirements, six NDI techniques were selected for evaluation: 1) enhanced radiography; 2) ultrasonic pulse echo; 3) acoustic emission; 4) plastic-cast edge replication; 5) stiffness monitoring; 6) temperature monitoring. The selected NDI techniques were evaluated using quasi-isotropic coupons statically loaded to various percentages of their average

ultimate tensile strength and by coupons fatigue loaded at $R = 0.0$ to various cycle lives. Two NDI techniques were selected for Tasks II and III based upon the experimental evaluation: enhanced radiography and edge replication. These two techniques are complementary and together meet all of the criteria established for NDI evaluation except for detection of fiber breakage. No simple, non-labor intensive, and inexpensive technique for detecting details of fiber breakage is presently available. Evaluation of the data obtained by enhanced radiography strongly supported the hypothesis that the fracture process, and thus coupon strength, in laminated coupons is path dependent after the onset of delamination.

TASK II - Effects of Loading Parameters (1980-1981)

The stress-life relations were obtained in detail at three stress levels chosen on the basis of Task I results. Tests were conducted at R ratios of 0, and $-\infty$. A number of different loading histories are being investigated to determine their effects on constant amplitude fatigue properties of the selected laminate and material at R ratios of 0 and $-\infty$. Only the experiments at $R = 0$ have been completed.

The effect of progressive cyclic loading on the static strength has been determined at the R ratio of 0. These coupons were progressively loaded to failure while under cyclic loading at 10 Hz at two different maximum stress versus time rates. The rates of maximum stress versus time were chosen such that failure occurred in approximately 1 minute or 1 hour (~600 and 36,000 cycles, respectively). The average strength of coupons progressively loaded to failure in one minute was ~ 10% lower than the static population, while those loaded in one hour was ~ 25% lower. This data leads to the conclusion that considerable damage is caused by the load cycling.

The effect of time under load was investigated at $R = 0$ by comparing results for two waveform types: 1) sine wave; and 2) trapezoidal wave. Trapezoidal waveform fatigue tests were conducted at two maximum stress levels, 45 and 60 ksi. Trapezoidal tests were conducted with the same rise and fall times as the previous sinusoidal tests, but with a 6 or 60 second hold period at maximum hold. The data shows that the different wave types and hold time periods gave identical results when compared on the basis of cycles. Therefore, one can conclude that hold time does not affect fatigue life and thus creep is not a dominant mechanism at room temperature.

The concept of summation of fatigue damage at different stress levels for prediction of fatigue life has been investigated at an R ratio of 0. Two of the maximum stress levels studied in the stress life scan of Task I and II were selected, 60 and 45 ksi. Specimens were tested at four different conditions of low and high stress. Results follow those expected on purely mechanical grounds namely; high-low blocks give a Miner's sum ≥ 1 and low-high blocks give a sum ≤ 1 .

The effects of overloads on constant amplitude fatigue life and damage is being investigated. Two different levels of single cycle overloads and one overload level, but with the overload cycle interspersed at two different regular intervals (every 100 or 1000 cycles), have been studied at $R = 0$. The overload levels selected were 60 and 70 ksi and the fatigue level was 45 ksi. These overload levels are approximately 75% and 90%, respectively, of the average static strength population.

The effect of preload is being studied to ascertain the possible relationship between the static strength and subsequent fatigue life. Preload tests consist of a preload followed by constant amplitude fatigue cycling at $R = 0$ or $-\infty$. The preload stress level and subsequent fatigue stress levels will be determined by the static strength and stress-life scan results of Task I and II.

An extensive NDI investigation is underway using enhanced x-ray and edge damage photography. This investigation is being conducted to document damage initiation and growth and failure modes.

TASK III - Effects of Environment and Geometry (1981-1982)

The environmental conditions selected for this task are: 82.2°C (180°F), dry condition (coupon moisture content as manufactured); 82.2°C (180°F), $95 \pm 5\%$ R.H. equilibrium moisture conditioning ($\sim 1.1\%$ by weight); and a two and eight week exposure to thermal spiking. All coupons to be used in this task are machined from the same panels manufactured under Task I so that commonality of material is maintained. Before mechanical testing all specimens will be conditioned in their appropriate environments until equilibrium is obtained. Environmental conditions will be maintained throughout the fatigue tests. The static tension and compression strength properties of the laminate will be determined for dry and wet coupons at 82.2°C (180°F) and after exposure to the two thermal spiking durations. In addition, the same static properties will be obtained at room temperature using coupons which contain a hole.

A constant amplitude fatigue stress-life scan will be conducted at two R ratios (0.0 and $-\infty$) for four different environmental conditions: 1) 82.2°C (180°F) dry; 2) 82.2°C (180°F) wet; and 3) and 4) two conditions of repeated thermal spiking. The stress life scan will also be conducted on the hole geometry specimen. Stress levels for these tests will be based upon previous test results of Tasks I and II. A limited study of the viscoelastic effects on the laminate fatigue properties will be conducted. This study will be undertaken at elevated temperature, in both the dry and wet environmental conditions, and will explore the different viscoelastic effects as evidenced under different fatigue test frequencies and those due to time under load. The effect of frequency will be explored at one stress level (selected based upon previous test results) using constant amplitude sinusoidal wave forms at frequencies of 1 and 10 Hz. The modifying effect of a hole upon the previously determined load history results will be obtained at $R = 0$ and -1 under four fatigue loading conditions: high-low stress, low-high stress, overload, and preload. These tests will all be conducted at a room temperature, laboratory air test condition. For all fatigue testing conducted in this task, one-half of all fatigue runout (1×10^6 cycles) coupons will be tested in static tension and one-half in static compression. Selected coupons tested in this task will be inspected for damage initiation and growth. The damage monitoring and inspection procedures will be the same as used in Task II.

TASK IV - Data Analysis and Reporting

In this task, a statistical analysis is being performed on each data set. The analysis includes determination of mean, and standard deviation and the Weibull distribution parameters. Comparison and correlation studies of the static and fatigue results with failure modes and fatigue life are being conducted to serve as a basis for formulating fatigue life models based on knowledge of failure mechanisms.

RESEARCH AND DEVELOPMENT INTO THE DESIGN TECHNOLOGY
OF ADVANCED COMPOSITES

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The present report describes some investigations into the vibrations of composite graphite/epoxy plates and tubes conducted at M.I.T. All specimens were made using Hercules AS1/3501-6 graphite/epoxy material which has a fiber fraction of about 60%.

1. Vibration Modes of Cantilever Plates (E. Crawley)

In this investigation, the natural frequencies and mode shapes of a number of cantilever graphite/epoxy plates were determined experimentally. See Figure 1. The samples tested included 8-ply composite plates with ply orientations $[0_2/+30]_s$, $[0/+45/90]_s$, $[+45]_s$ and length to width ratios of 1 and 2. Also tested were some composite cantilever cylindrical shell sections as well as an isotropic aluminum plate for reference purposes. Natural frequency and mode shapes results were compared with those calculated by a finite element analysis. Agreement between calculated and observed mode shapes was excellent, while only fair agreement is found for the frequencies. The discrepancy in frequencies seems to be due to uncertainty in the material properties. The dynamic flexural modulus E_L appears to be about 25% less than the static in-plane measured modulus, even after conventional shear effects have been taken into account. The above investigation is reported more fully in Ref. 1.

Also, in connection with this investigation, a convenient method for rapidly estimating the natural frequencies of composite plates was developed, based on a partial Rayleigh-Ritz (Kantorovich) analysis. This expresses the n^{th} bending, torsion, and chordwise modes of the plate in the form,

$$\omega_{Bn} = k_{Bn} \sqrt{D_{11}/m l^4} \quad (1)$$

$$\omega_{Tn} = k_{Tn} \sqrt{48 D_{66}/m l^4 C} \quad (2)$$

$$\omega_{Cn} = k_{Cn} \sqrt{D_{22}/m C^4} \quad (3)$$

where D_{11} , D_{66} , D_{22} are the composite plate stiffnesses, m is the mass per unit area, ℓ is the plate length, c the plate width, and k_{Bn} , k_{Tn} , k_{Cn} are appropriate coefficients. Reasonable correlation with the more accurate finite element plate results were obtained for the lower plate modes, for symmetric laminates with modest bending-twisting coupling terms D_{16} and D_{26} . For strong bending-twisting coupling terms, more general coupled differential equations are presented. This investigation is reported more fully in Ref. 2.

2. Damping of Cantilever Composite Plates (D. Boyce)

In this investigation, the material damping properties of a number of graphite/epoxy double cantilever beams were investigated. The specimens tested included $[0]_8$, $[0]_{12}$, $[90]_8$ and $[\pm 45/\mp 45]_8$ laminates subjected to base excitation at their fundamental resonant frequencies. See Figure 2. All tests were conducted in air. Isotropic aluminum specimens were also tested for reference purposes and to estimate the amount of air damping and clamp support damping in the tests. The damping of the aluminum specimens agreed reasonably with previously reported data by Granick and Stern in Ref. 3. The graphite/epoxy specimens gave damping loss factors η_s of the order of .0008 to .002 for the $[0]_8$ specimens which seemed to be comparable to that of the aluminum specimens. The $[90]_8$ and $[\pm 45/\mp 45]_8$ laminates gave somewhat higher loss factors, but these were still small. The major source of damping in practical structures appeared to originate from joints rather than from the material itself. Also, it was found in this investigation, as in the previous investigation, that the dynamic flexural modulus E_L seemed to be about 25% less than the static in-plane measured modulus. The above damping and stiffness investigation is reported more fully in Ref. 4.

3. Torsional Vibrations of Tubes (O. Bauchau)

In this investigation, the torsional vibration and damping properties of a number of graphite/epoxy tubes were determined experimentally. See Figure 3. The samples tested were tubes with inner diameter 7.67 cm (3.0 in), length 33.0 cm (13.0 in), and ply orientations $[\pm 45]_3$, $[\pm 30]_3$, $[0_2/\pm 45/\pm 45/0_2]$, $[\pm 15]_3$ and $[0]_{12}$. One end of the tube was clamped to a torsional shaker base, while the other was attached to a heavy disk to lower the torsional frequency of the assembly. Also tested was an aluminum tube for reference purposes. From the torsional natural frequencies of the various specimens, the elastic shear modulus G was derived, and was found to be in good agree-

ment with nominal static in-plane measured values from flat plate specimens. The tubes were also tested statically in a torsion testing machine, and in this case, the shear modulus was found to be some 10% less than that found from the dynamic tests. This was possibly due to the large strains in the static torsion tests as compared with the very low strains of the dynamic vibration tests. This investigation is reported more fully in Ref. 5.

Attempts to find the damping in torsion of the composite material itself were unsuccessful, as the tube damping was dominated by the joint between the tube and the disk. Much change in damping was obtained by varying the bond here. The least damping occurred with the disk bolted tightly by eight equally spaced bolts. Again, the importance of joint details in determining the torsional damping of a shaft was brought out.

Further work is continuing with the construction of a tubular driveshaft 1.37 m (4.5 ft) long to be used in a small gas turbine engine. Torsional stiffness and strength as well as bending stiffness and strength are being determined, paying attention to the joint details.

4. Flutter of Symmetric Composite Plates (S. Hollowell)

In this investigation, the flutter and aeroelastic characteristics of several rectangular cantilever graphite/epoxy plates are being investigated in a small wind tunnel. See Figure 4. The plates to be tested are 6-ply unbalanced symmetric laminates with ply orientations, $[+45_2/0]_S$, $[+30_2/0]_S$, and $[0_2/90]_S$ with generally strong bending-twisting coupling (except for the last case). By reversing the flow direction, both favorable and unfavorable angle-of-attack changes result depending on the D_{16} terms. Preliminary experimental results from a $[+45_2/0]_S$ plate of length/width ratio equal to 4 have indicated a flutter condition for the flow from one direction and a divergence condition for the flow from the opposite direction. Analysis is being done to confirm the experimental trends for this series of aeroelastically tailored plates.

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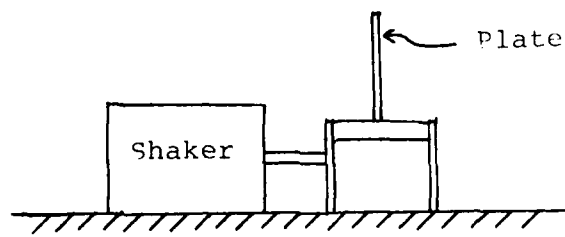


Figure 1 Vibration Modes of Cantilever Plate

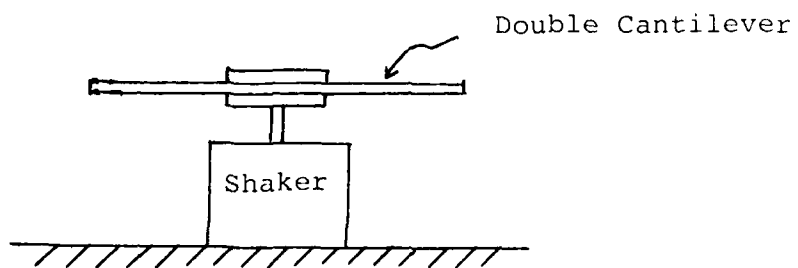


Figure 2 Damping of Cantilever Composite Plates

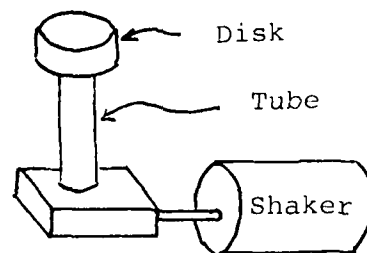


Figure 3 Torsional Vibrations of Tubes

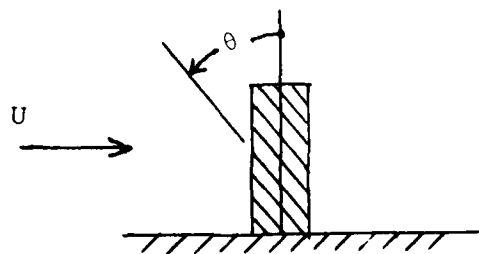


Figure 4 Flutter of Composite Plates

FATIGUE FAILURE OF COMPOSITE LAMINATES

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For most materials the failure properties such as strength and lifetime exhibit more scatter than other properties. The reason is known to be that these failure properties are sensitive to local defects which vary significantly from element to element even though all the elements are made of the same material under the same manufacturing condition.

Composite materials are no exception. Interestingly, however, several investigations have shown a possible existence of a relationship between static strength and fatigue life. The relationship is such that a stronger element also has a longer fatigue life.

The strength-life relationship, once proven, will no doubt be very helpful in proof testing of composite structures because one can then provide a certain degree of assurance as not only to the inherent strength of the structure but also to the expected lifetime. Aside from these practical benefits, an investigation on such relationship will lead to a better understanding of the fatigue failure mechanisms and of the variability of fatigue life in composites.

The main objectives of the program were thus to delineate the effect of proof test both on the subsequent strength and on the subsequent fatigue life, and to identify the sources of the scatter in fatigue life. The laminate chosen was $[0_2/90/45]_s$ Gr/Ep with an average fiber volume fraction of 0.66. Specimens were 12-mm wide and 150-mm long with 75 mm of gauge section. Twenty specimens were tested in each test series unless otherwise indicated. Fatigue tests were carried out at the stress ratio of 0.1 and the loading frequency of 5 Hz.

Table 1 shows the test matrix of the program together with the strength properties of the panels used. There is little variation of strength from panel to panel with the exception of panel #11. Noting that panel #11 has only three 0-deg plies rather than the intended four, one calculates the expected strength to

be 590 MPa. Since the measured strength is slightly higher than this value, panel #11 has been included in the test series as well.

Figure 1 compares the initial strengths with the residual strengths after a proof test to $0.87 \bar{X}$ and $0.95 \bar{X}$ (average static strength), respectively. In the lower strength region, the residual strengths are slightly higher than the initial strengths, after a proof test to $0.87 \bar{X}$. However, in the higher strength region, an opposite trend is observed. When the proof stress is $0.95 \bar{X}$, the residual strengths are consistently lower than the initial strengths, although the difference is rather small.

A relation between fatigue stress and fatigue life (S-N relation) is shown in Figure 2. The line drawn through the 50% probability of survival does not show any possibility of a fatigue limit down to $0.7 \bar{X}$ for this laminate. Nevertheless, all ten specimens tested at $0.6 \bar{X}$ survived 10^6 cycles.

A postulated relationship between static strength and fatigue life at $0.8 \bar{X}$ is shown in Figure 3 together with the fatigue life data after proof test. First of all, the minimum fatigue lives after proof test fall on the postulated strength-life curve. Also, the number of specimens surviving the minimum fatigue life at each proof stress level is the same as would be expected from the curve. All these results lead to the conclusion that the postulated curve really describes a relation between static strength and fatigue life for this laminate.

The effect of proof test on the subsequent fatigue life distribution at $0.8 \bar{X}$ is shown in detail in Figure 4. After a proof test to $0.87 \bar{X}$, the fatigue lives are shorter than the initial ones. However, a proof test to $0.95 \bar{X}$ does not show the same deleterious effect although more ply failure occurs at this stress level. At the fatigue stress of $0.7 \bar{X}$, the fatigue lives were longer after a proof test to $0.87 \bar{X}$, in contrary to the results of Figure 4. Therefore, one can conclude that the effect of proof test on the subsequent fatigue life distribution is not substantial.

Figure 5 shows a definite correlation between modulus and fatigue life at the fatigue stress of $0.87 \bar{X}$. Similar correlations were obtained in the other test series. Since a higher modulus can be taken as an indication of a higher volume fraction, the corresponding fiber stress will be lower at the same fatigue stress level. Thus the fatigue life will be longer when the modulus is higher.

At the same fatigue stress level, the longer a specimen survives, the more delamination it shows before the final failure. Only a very limited amount of delamination at the fracture site is characteristic of the static failure mode. Since there is a clear distinction between the fatigue and static failure modes,

a correlation between failure mode and fatigue life at $0.8 \bar{X}$ can be established as in Figure 6. It is interesting that a rather well defined transition region exists around 20,000 cycles. Thus, the failure sequence of this laminate is the ply failure followed by delamination.

To delineate the effect of gripping on fatigue failure, fatigue lives at $0.8 \bar{X}$ are plotted against the corresponding failure zones in Figure 7. The gage section was divided into five zones of equal length. These zones were labeled alphabetically with the zone E being the nearest to the moving grip. The results of Figure 7 show very little difference in fatigue life resulting from different failure zones.

In conclusion, the results of our investigation on the tension-tension fatigue behavior of $[0_2/90/\pm 45]_S$ Gr/Ep laminate can be summarized as follows:

1. A relation exists between static strength and fatigue life such that a stronger specimen has a longer fatigue life. Thus a minimum fatigue life can be assured by a proof test.
2. The effect of proof test on the subsequent strength and life is not significant.
3. At the 50% probability of survival the S-logN relation is linear down to $0.7 \bar{X}$. All ten specimens tested at $0.6 \bar{X}$ survived 10^6 cycles.
4. A higher modulus is an indication of a longer fatigue life.
5. The amount of delamination increases with the fatigue life at the same fatigue stress. For example, the transition from a typical static failure mode to a typical fatigue failure mode occurs around 20,000 cycles when the fatigue stress is $0.8 \bar{X}$.
6. There is no correlation between the location of failure and the corresponding strength and fatigue life. Thus the effect of gripping is negligible.

Table 1. Test matrix

Test series	Panel Number	
Strength	10	
Average, MPa	11a	
Coeff. Var., %	789.7	
Proof stress, strength	609.1	
% Ave. strength	7.23	
Max. fatigue stress	87	
% Ave. strength	95	
Proof stress, fatigue	80	
% Ave. strength	70	
	87	
	70	
	87	
	95	
	95	
	782.0	
	5.37	
	788.1	
	5.21	
	--	
	--	
	--	
	92.5, 60	
	92.5, 60	
	--	
	--	

^a[0₂/90/±45/±45/90₂/0] laminate.

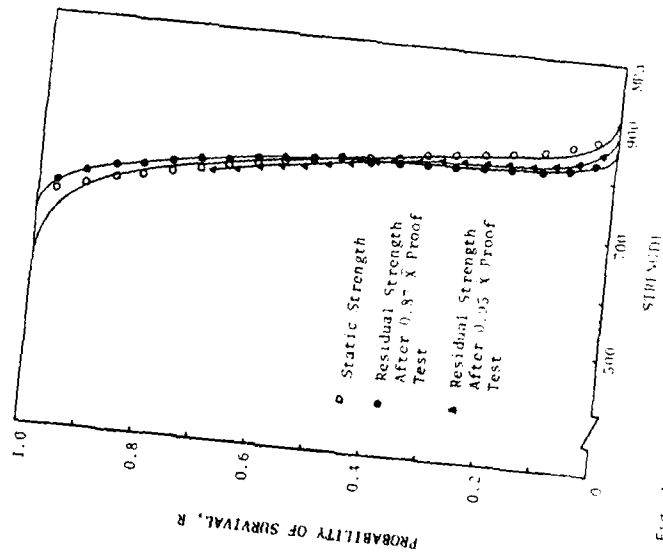


Fig. 1. Effect of prog test on subsequent stress.

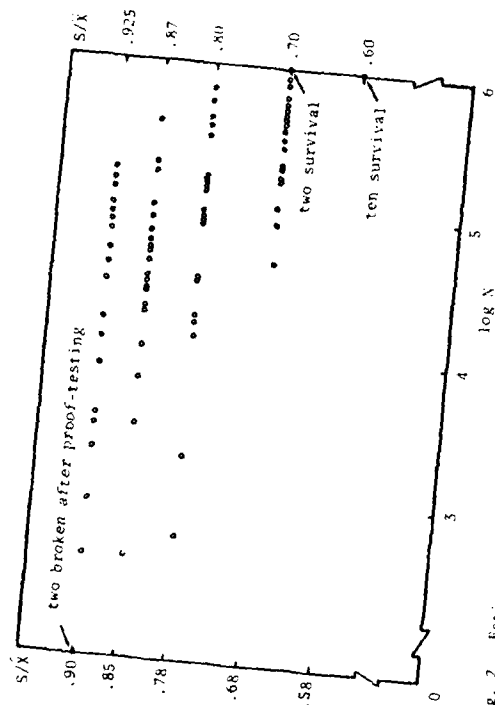


Fig. 2. Fatigue stress life (S-N) relation

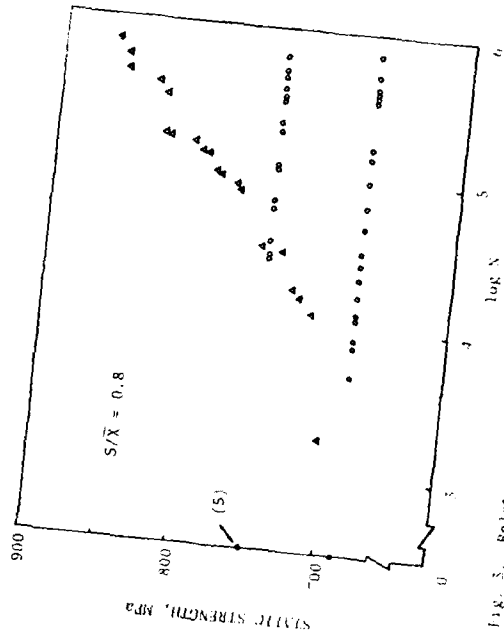


Fig. 5. Relation between strengths and fatigue lives at $0.8 \bar{\sigma}$. Subsequent fatigue lives after proof tests are also shown.

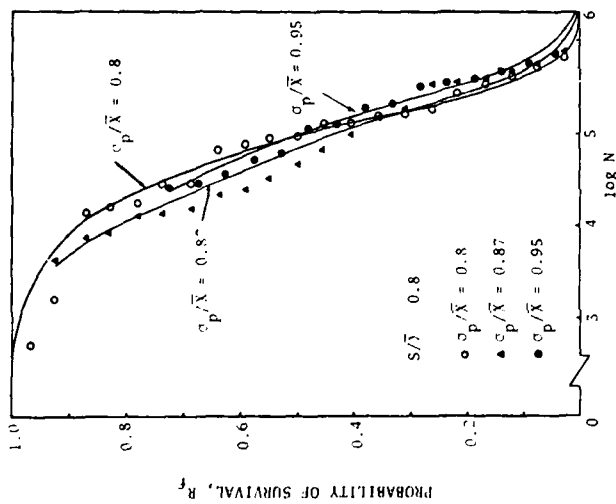


Fig. 1. Effect of proof test on subsequent fatigue life distribution.



Fig. 6. A correlation between failure mode and fatigue life at $0.8 \bar{X}$.

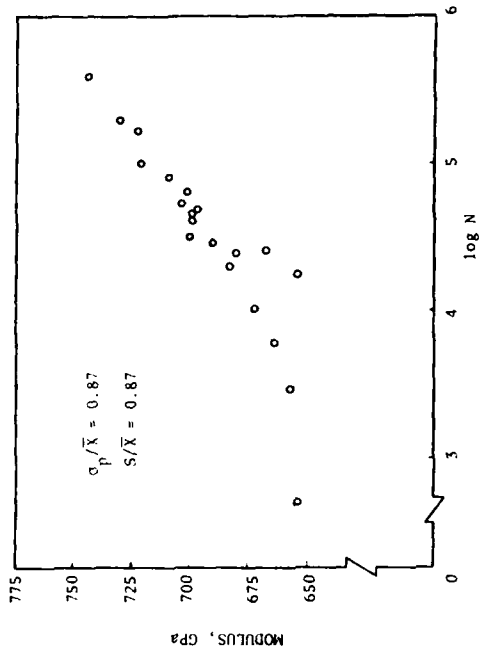


Fig. 5. A correlation between modulus and fatigue life at $0.87 \bar{X}$.

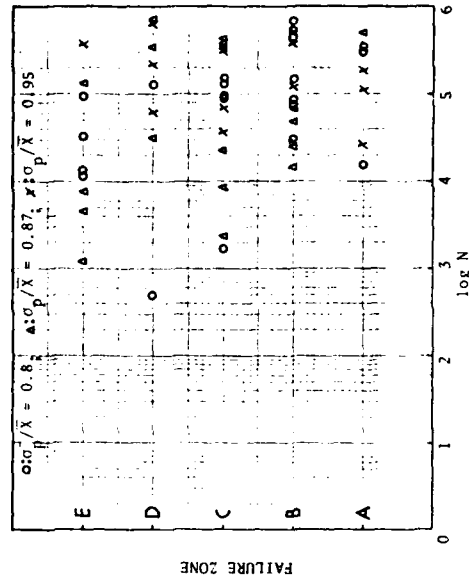


Fig. 7. A correlation between failure zone and fatigue life at $0.8 \bar{X}$.

FIRST PLY FAILURE OF COMPOSITE MATERIALS

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A knowledge of the first ply failure strength is essential for designing with composite materials. The aim of the present work is to develop a simple criterion for the first ply failure of multidirectional laminates. This criterion simplifies considerably the process of designing for strength.

The strength of a unidirectional composite is given by the roots of the quadratic equation (1,2)

$$F_{ij} \sigma_i \sigma_j + F_i \sigma_i = 1 \quad (1)$$

The strength parameters (F's) can be determined by simple tests on a unidirectional composite referred to its orthotropic axes, except for the interaction term between the two normal stress components. This term is assumed to have a value

$$F_{xy} = -\frac{1}{2} \left\{ F_{xx} F_{yy} \right\}^{\frac{1}{2}} \quad (2)$$

Eq. (1) describes the failure surface in stress space. The failure surface in strain space is given by

$$G_{ij} \epsilon_i \epsilon_j + G_i \epsilon_i = 1 \quad (3)$$

The strength parameters in Eq. (3), or the G's can be calculated from the F's using the on-axis stress-strain relations, if we assume the composite to be linearly elastic up to failure.

The failure surface of Graphite-Epoxy composites in the normal strain space is shown in Figure 1. These are 19 ply orientations corresponding to 0°, 5°, 10°, ... up to 90°. The failure surface for each ply was obtained using Eq (3) and substituting the appropriate values of $G_{ij}^{(i)}$ and $G_i^{(i)}$. The strength parameters obey the transformation laws for tensors, which enables us to determine $G_{ij}^{(\theta)}$ and $G_i^{(\theta)}$. Alternately, the on-axis ply strain can be calculated from the strain applied to the laminate using the strain transformation relations and the failure surface determined using Eq. (3) and the strength parameters for the 0° ply. As the ply orientation changes, the size and the location of the failure surface also changes. The failure surface for a laminate in strain space is obtained by simply superposing the failure surfaces of the laminae, because the strain is assumed to be the same through the thickness of the laminate. The failure surface is independent

of the ply ratios, but the loading path depends on the compliance of the laminate which is a function of the ply ratio. In the case of a multi-directional laminate, the failure surfaces intersect and form an inner envelope which corresponds to the first ply failure. It is seen that a minimum first ply failure strength can be defined for T300/5208 Graphite-Epoxy laminates which is the innermost envelope in Fig. (1). This envelope is independent of the ply orientations and the ply ratios. The minimum FPF envelope is formed by the intersection of 0° and 90° plies in the normal strain space. In the case of T300/5208, this envelope can be approximated by an ellipse.

We have carried out similar calculations with other composites AS/3501 (graphite-epoxy), Boron/Epoxy, Glass/Epoxy and Kevlar 49/Epoxy. The results for Kevlar 49/Epoxy are shown in Fig. (2).

Further simplifications result if we plot the failure surface in the q - r space at constant values of p , where p , q , r are the coordinates of Mohr circle (2,3). The failure surface of 0° ply is again an ellipse. The failure surface of any other θ° ply is obtained simply by the rigid body rotation of the ellipse by $2\theta^\circ$. The innermost envelope of these failure surfaces can now be approximated conservatively by a circle whose radius can be expressed in terms of G 's. Fig. 3 shows the FPF strength at a constant value of p ($=0$) for all composites studied. Aluminum is shown for comparison but the ultimate strength (and not the yield strength) is shown. It is now possible to make comparisons between conventional materials and composites and evaluate the effect of materials substitution. The simple procedures developed in the present work may be helpful in the optimization of composite laminates for strength.

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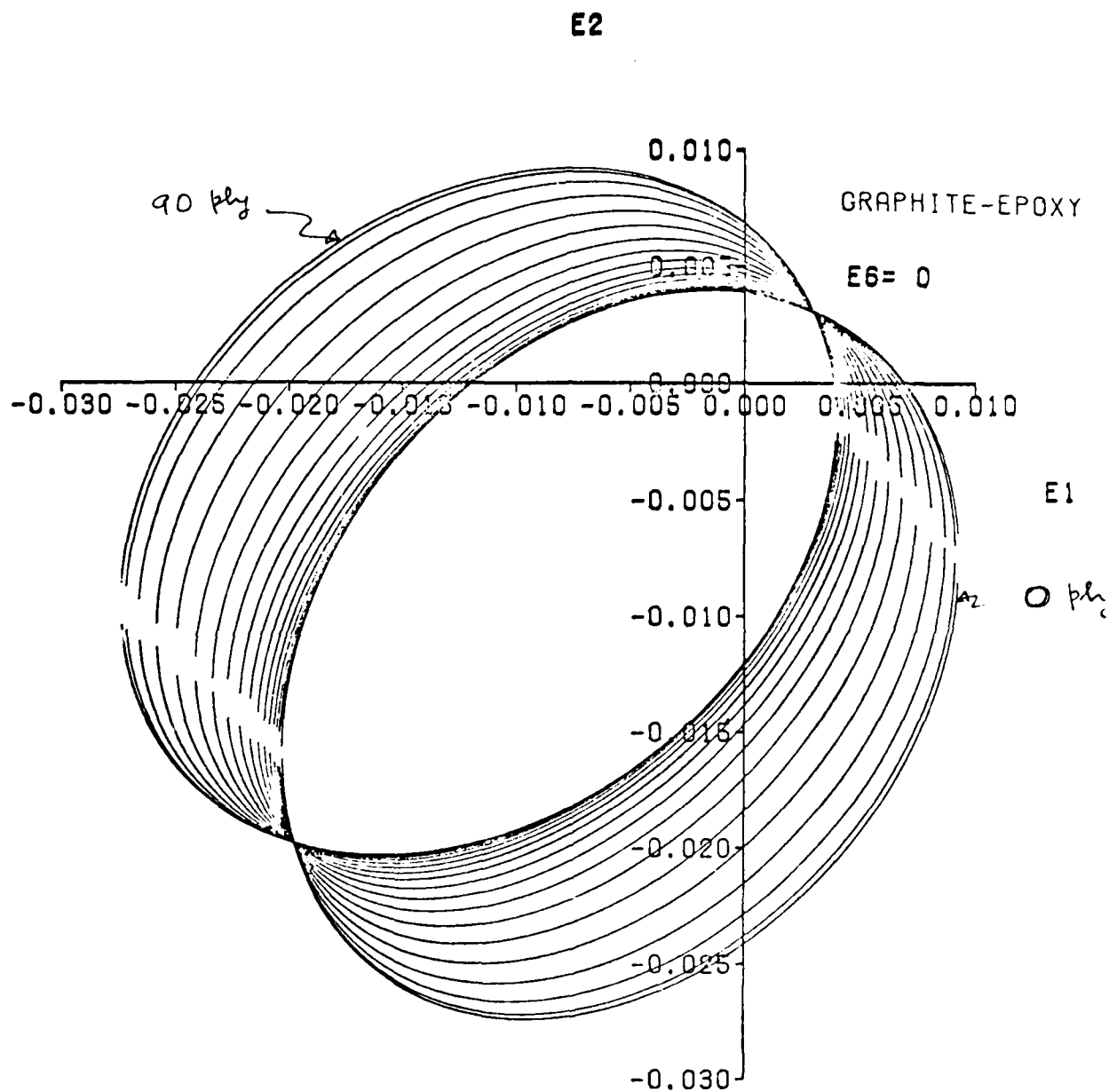


Figure 1. The failure surface of T300/5208 composites. The innermost envelope represents the minimum first ply failure strength in normal strain space for this material.

E2

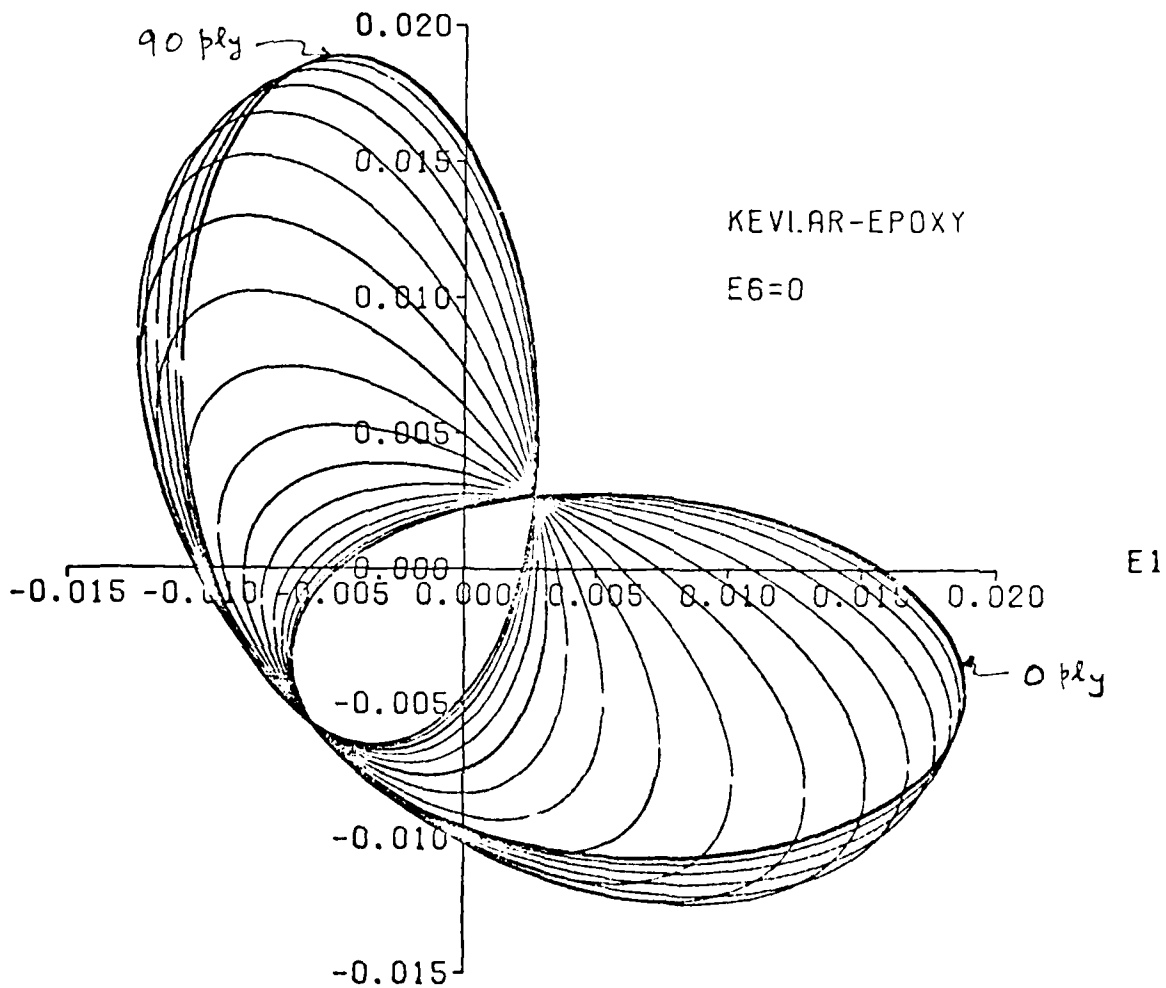


Figure 2. The failure surface of Kevlar-Epoxy composites. The innermost envelope represents the minimum first ply failure strength in normal strain space for this material.

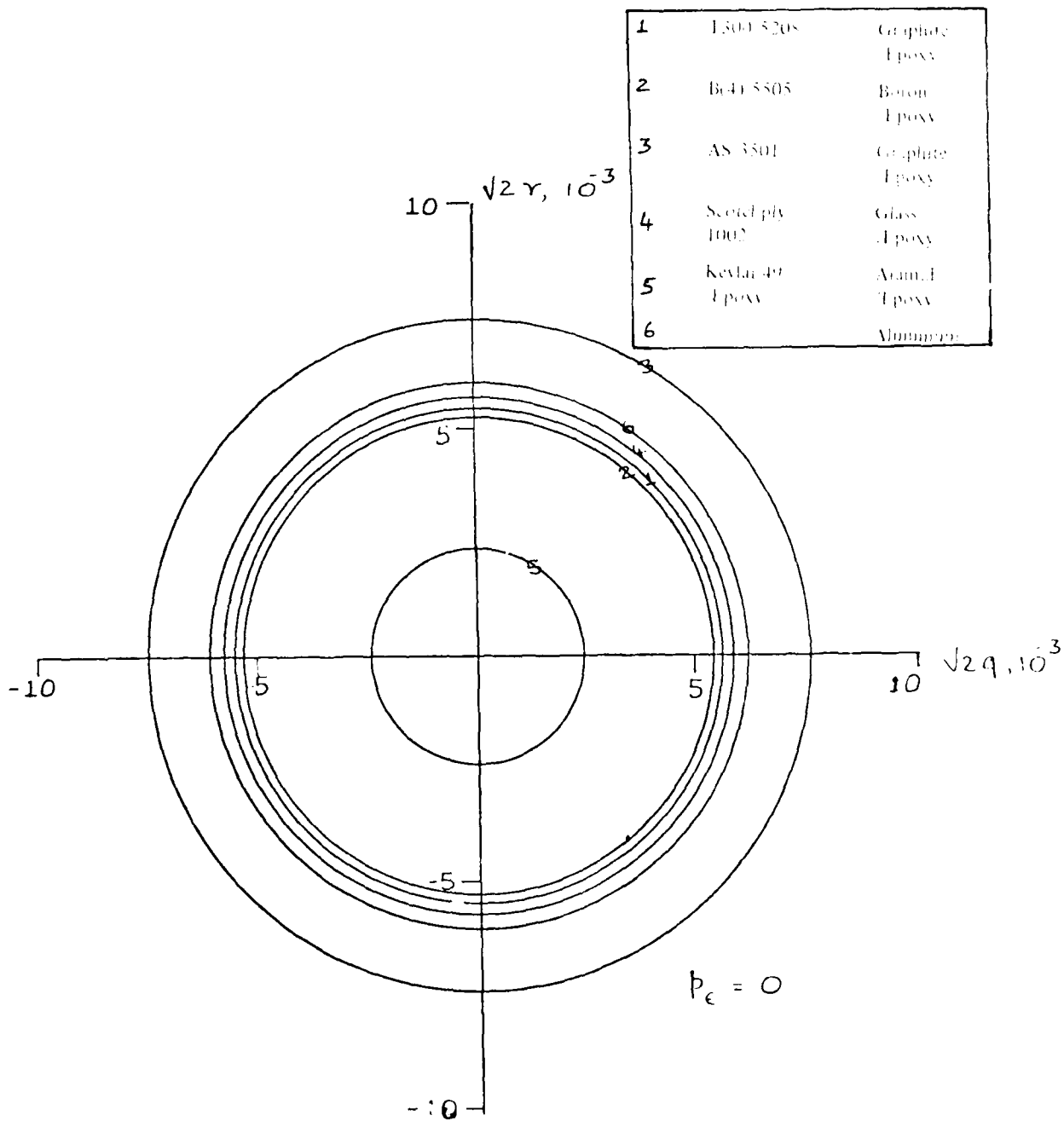


Figure 3. The first ply failure strength of composites in strain space using Mohr coordinates. The curves are drawn for a constant value of $p_{\epsilon} = 0$.

INPLANE STRESS ANALYSIS OF MULTIDIRECTIONAL
COMPOSITE LAMINATES WITH A LOADED FASTENER
HOLE-USING STRESS DISTRIBUTION IN THE
CONSTITUENT ANGLE PLY LAMINATES

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It has been shown that given the stress distribution in angle ply laminates with a loaded fastener hole, the stress levels in a multidirectional composite laminate with any volume fraction of the angle ply laminates can be computed. This investigation has been carried out within the framework of laminated plate theory. A finite element technique has been utilized to conduct the stress analysis of multidirectional laminates and constituent angle ply laminates, with the same set of boundary conditions. A simple averaging technique has been suggested to approximate the stress components for a multidirectional laminate from the constituent angle ply laminate stress levels. It has been demonstrated that this approximation gives results very close to the finite element results obtained for the composite laminate. This approximation is very useful in optimum design of bolted joints in composite laminates.

For the optimum design of bolted joints in composite laminates, a knowledge of stress distribution around the fastener hole due to the applied load is very important. With the variation in ply orientations and volume fractions in the laminate, the stiffness properties change and consequently the stress levels pertaining to the same boundary conditions differ. For optimum strength requirements, one needs to compute stress levels in the laminate for given boundary conditions with different volume fractions and ply orientations. Because, in many practical situations the closed form elasticity solutions are not available, a finite element method has to be implemented. In the present investigation, the finite element method has been used to conduct the stress analysis of the laminate for a number of ply orientations and volume fractions. A simple averaging procedure has been suggested to approximate the stress levels in the composite laminates, with any combination of ply volume fractions, from the stress distributions in constituent ply laminates with the same set of boundary conditions.

The present study consists of the computation of stress distribution in composite laminates with a loaded fastener hole using finite element method. The results are calculated for various multidirectional laminates including the constituent angle ply laminates. The stress levels at various points in the constituent angle ply laminates are used to approximate the states of stress for multidirectional composite laminates with

different ply volume fractions. There exists a very good agreement between the results from finite element method and the results from the approximation method for multidirectional laminates.

Figure 1 shows a laminate with a fastener hole, coordinate axis and relevant dimensions. The loaded hole boundary conditions are considered and are imposed by introducing radial displacement boundary constraints at the circular hole boundary and a prescribed load at the opposite plane edge. The objective of the present investigation is to present a simple method for obtaining the stress distribution around the circular hole of a multidirectional laminate using the stress levels calculated for constituent angle ply laminate systems. To achieve this, the following assumptions have been made:

1. The laminate obeys the laws of classical laminated theory,
2. The contact surface between the laminate and the bolt is semi-circular,
3. The whole is filled with a rigid core,
4. No transverse load, due to the bolt, is acting at the laminate.

Finite element method has been used for conducting the stress analysis of the laminate. Using the finite element results for individual angle ply laminates, the stress levels for composite laminates made of any combination of these angle plies are approximated.

The stress analysis of the laminate is conducted using the finite element computer code, NASTRAN. The uniaxial tensile loading conditions are considered. Due to the symmetry of the laminate and applied loads about the x-axis, half of the laminate has been modeled for finite element analysis. This part is divided into 372 quadrilateral and triangular elements. The radial displacement along the bolt contact semicircular boundary is taken to be zero and a known tensile load is applied at the opposite plane edge. The gross laminate material properties based on the laminated plate theory (Reference 1) have been used. A finite grid plot, as obtained during the NASTRAN computations, is given in Figure 2. Various numerical exercises with different finite element grids show that the present model is good enough to give acceptable results for all practical purposes.

Stress distribution has been computed for angle ply and multidirectional laminates with a loaded fastener hole, using finite element technique. The finite element results for angle ply laminates have been used to approximate the stress levels in multidirectional laminates. The following procedure has been adopted to approximate the stress levels for $(0_m/90_n/(\pm\theta)_p)_s$ multidirectional laminates from stress levels computed for individual ply systems with the same boundary conditions:

$$\sigma_x = \frac{1}{m+n+2p} \{ m\sigma_x^0 + n\sigma_y^0 + 2p\sigma_x^{(+\theta)} \}$$

$$\sigma_y = \frac{1}{m+n+2p} \{ m\sigma_y^0 + n\sigma_x^0 + 2p\sigma_y^{(+\theta)} \}$$

$$\tau_{xy} = \frac{1}{(m+n+2p)} \{ m\tau_{xy}^0 + n\tau_{xy}^0 + 2p\tau_{xy}^{(+\theta)} \}$$
(1)

It has been shown that the approximation method results agree very well with the finite element results (Figures 3 and 4). If we follow the energy formulation of the finite element method, the above expressions can be derived. This procedure of conducting stress analysis of multidirectional laminates with different volume fractions of constituent angle ply is very economical and useful for optimum design of bolted joints.

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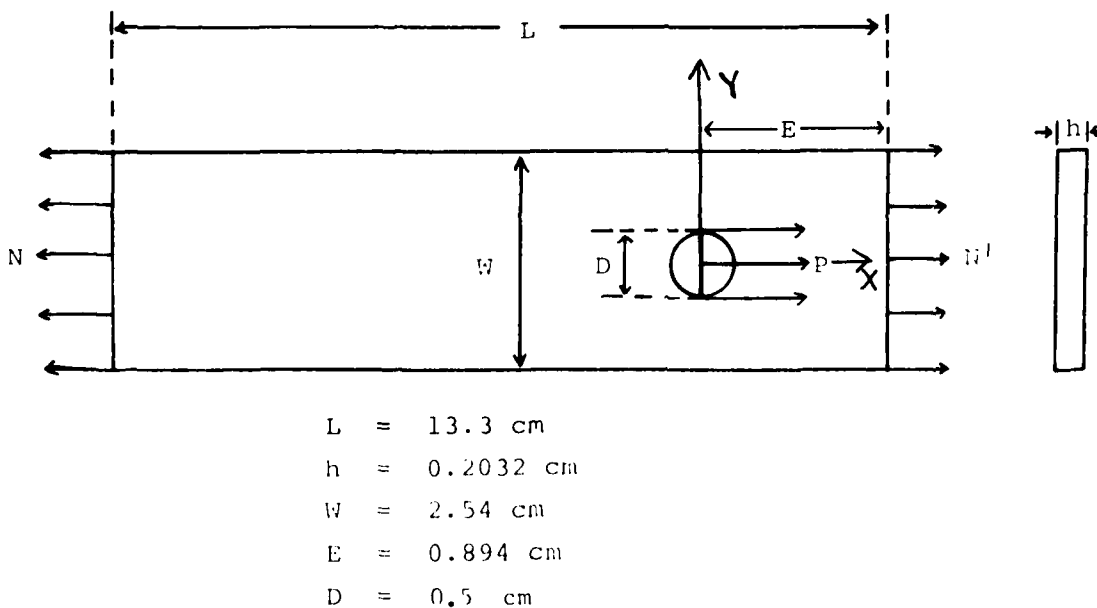


Figure 1. Laminate with Coordinate Axis and Dimensions

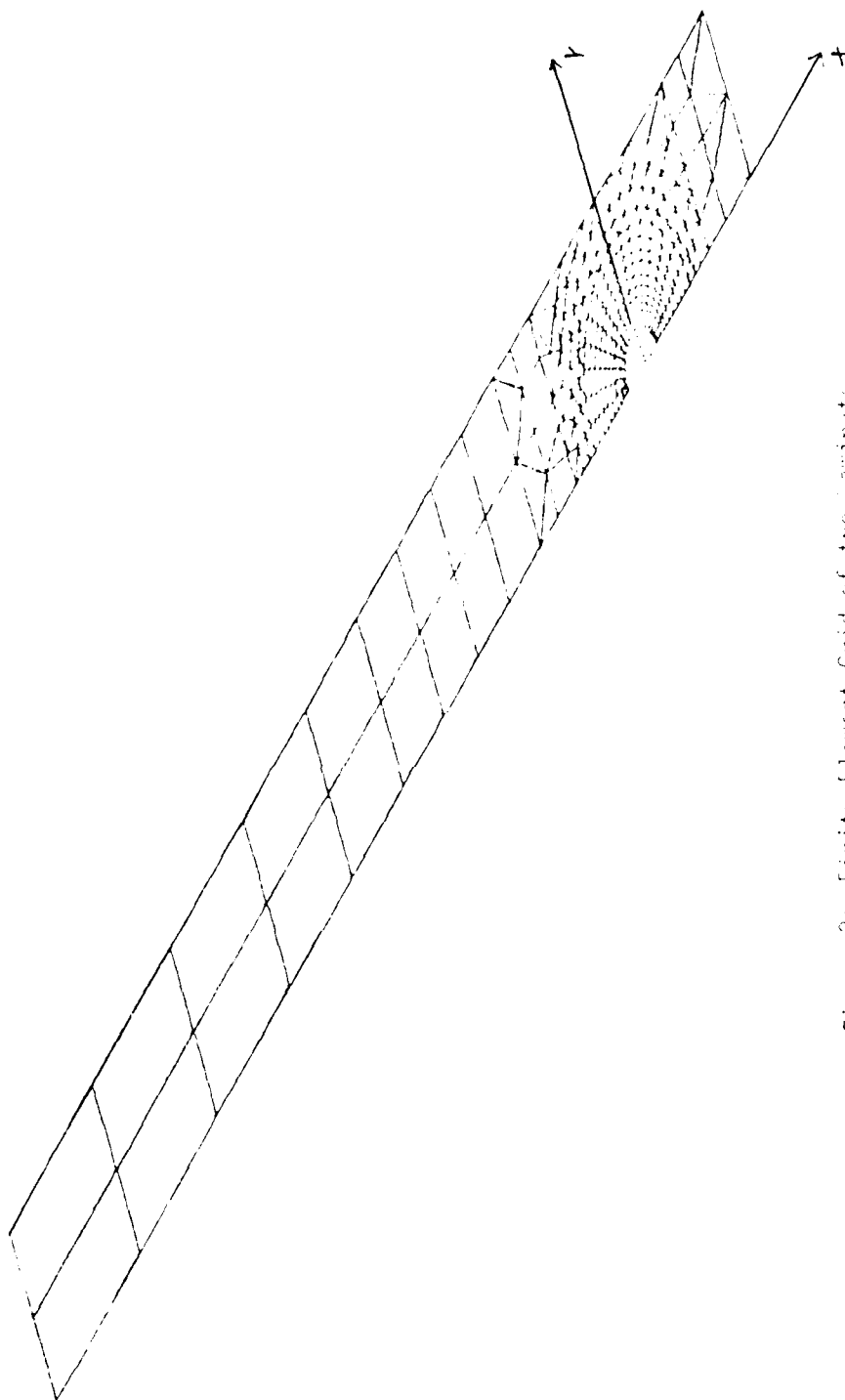


Figure 2: Finite Element Grid of the Cantilever

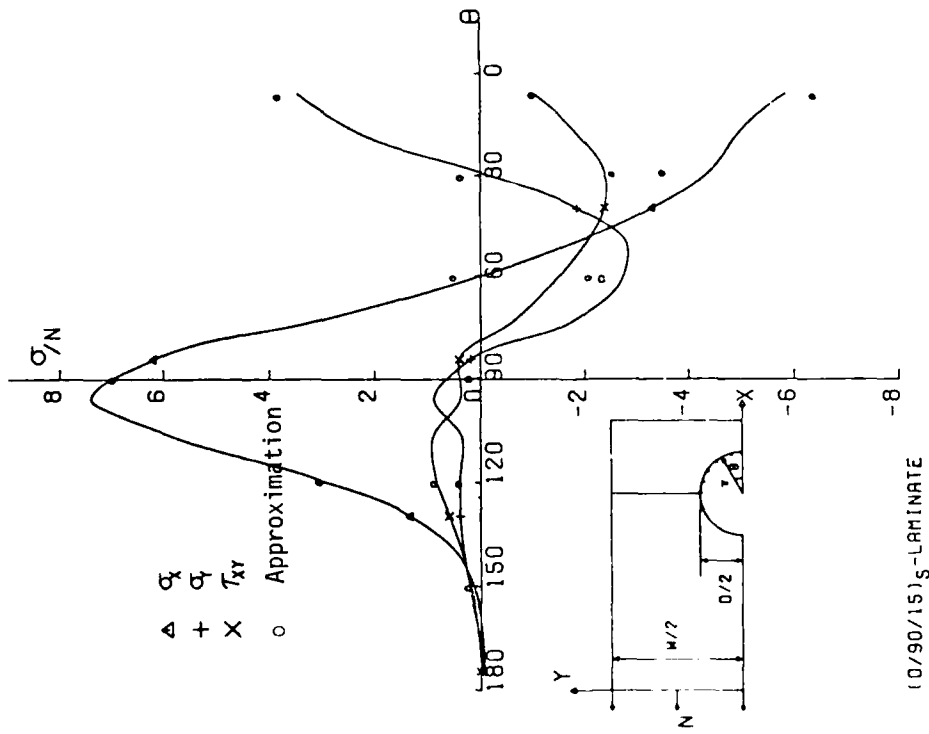


Figure 3.

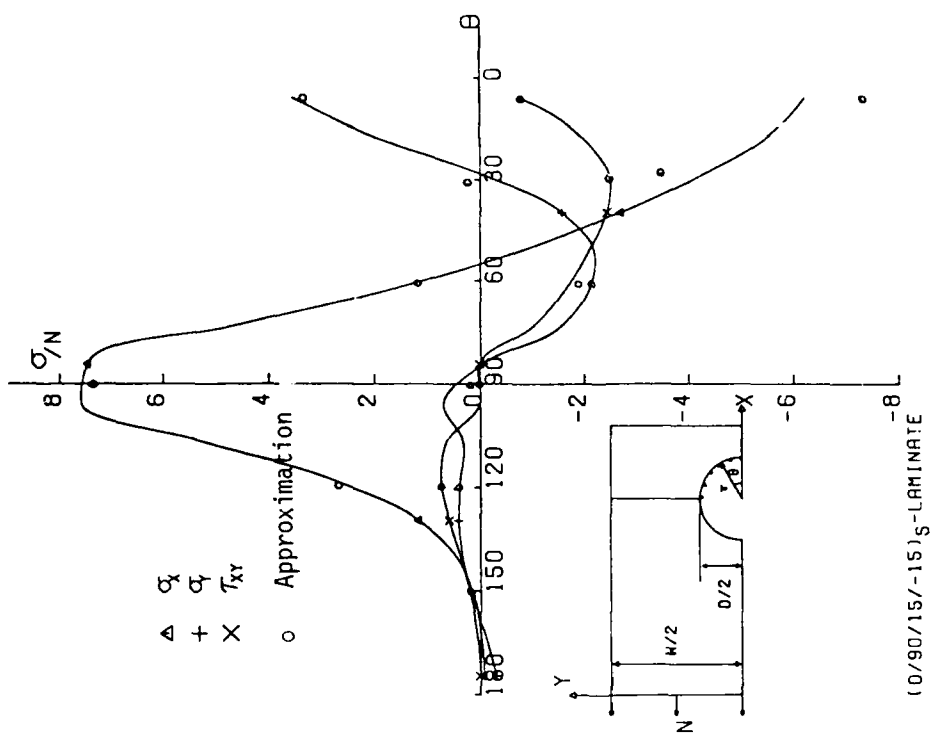


Figure 4.

Stress Distribution Around the Hole Boundary

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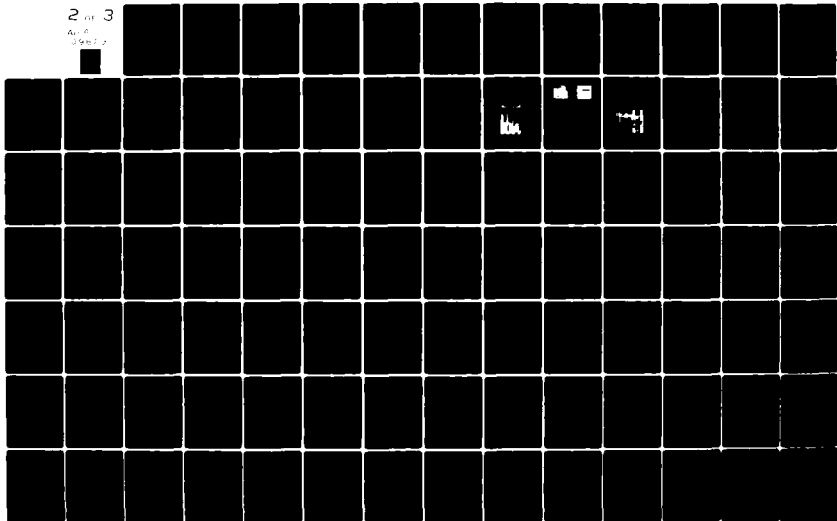
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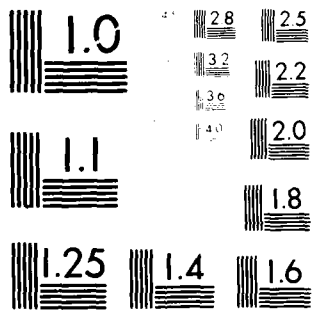
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ANALYSIS OF INSTABILITY-RELATED DELAMINATION GROWTH

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Local buckling of delaminated plies can precipitate rapid delamination growth and structural collapse. To predict the rate of delamination growth, accurate stress analyses are needed. This paper discusses two analyses, developed to calculate parameters such as stress or strain-energy-release rates, which might be used to predict growth of through-width delaminations (see fig. 1). One analysis was a geometrically nonlinear finite-element analysis. It provided rigorous solutions. The other was a "strength of materials" solution based on the insight gained from the rigorous solutions.

The finite-element analysis was custom designed for efficient analysis of postbuckled through-width delaminations. A series substructuring technique was used to conveniently exploit the linear response of most of the structure; only the buckled column was treated as a nonlinear structure. A reduced integration scheme (ref. 1) was used to eliminate the inherent excessive bending stiffness of the isoparametric elements used in the program. Hence, a small number of elements could accurately model the bending deformation. Because the incremental stiffness of a buckled column is nearly zero, the incremental stiffness matrix can become singular during the iterative solution of the nonlinear equations. This problem was circumvented by employing an incremental displacement technique (ref. 2).

Through-width delaminations were studied with the finite-element analysis. As a check of the analysis, measured and calculated lateral deflections were compared. As shown in figure 2, the analysis reflects the actual behavior of postbuckled, through-width delaminations. The possibility that mode I strain-energy-release rate (G_I) is a simple function of lateral deflection was investigated. Figure 3 shows typical results for several delamination lengths. As one might expect, for a given lateral deflection G_I is larger for the shorter delaminations. Note that G_I is not a simple function of lateral deflection. Contrary to intuition, G_I does not increase monotonically with lateral deflection. In fact, the crack tip actually closes at a deflection, δ , of approximately 1.5 mm. This behavior can be explained by considering the load transfer near the crack tip. After the delaminated region buckles, an increase in remote load (and lateral deflection) causes essentially no change in the load carried by the buckled column (region C in fig. 4). However, the load carried by region A continues to increase with increased applied load. Hence, load must be transferred from A to D. The eccentricity in the load path

causes a moment which tends to close the crack tip. Simultaneously, the lateral deflection of the column causes a moment which tends to open the crack tip. The interaction of these processes causes the trends shown in figure 3.

A "strength of materials" analysis was developed based on the insight gained from the finite-element analysis. A schematic of the idealization is shown in figure 5. Simple equations were developed for each region.

The peeling moment, M_0 , at the debond edge in the delaminated strip is due to lateral deflection, and is simply $\frac{1}{2}P_C\delta$, where P_C is the load in the column. The closing moment is given by

$$M_C = -C_1 \frac{t}{2}(P_A - P_C) \quad (1)$$

where P_A = load carried by region A (fig. 4). The mode I strain-energy-release rate resulting from the moments M_0 and M_C is given by

$$G_I = \frac{C_2}{8EI} \left[P_C\delta - C_1 t (P_A - P_C) \right]^2 \quad (2)$$

The constants C_1 and C_2 are determined from finite-element results. Both constants are independent of the delamination length; C_1 is also independent of the thickness of the buckled column. Hence, G_I can be calculated with the approximate analysis for many different configurations after studying a few configurations with the finite-element analysis. As shown in figure 3, the approximate analysis predicts the effect of lateral deflection on G_I very well. Figure 6 shows the effect load has on G_I for various debond lengths, as calculated by each analysis. Again, the agreement is very good.

This study demonstrated the potential of approximate analyses for calculating parameters that characterize instability-related delamination growth.

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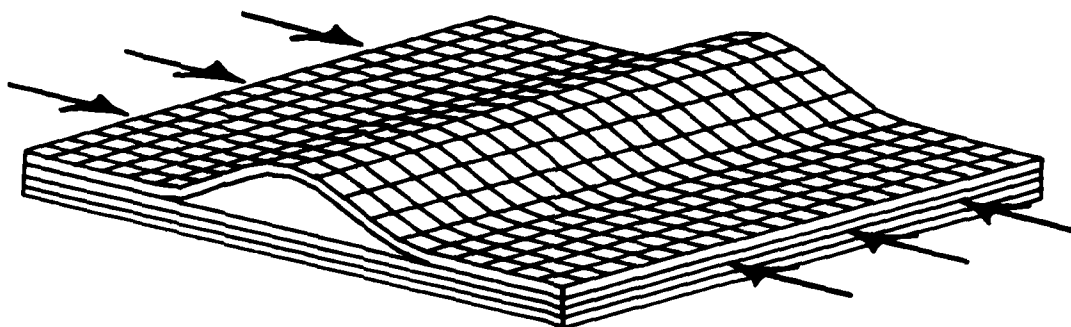


Figure 1.- Local buckling of laminate with through-width delamination.

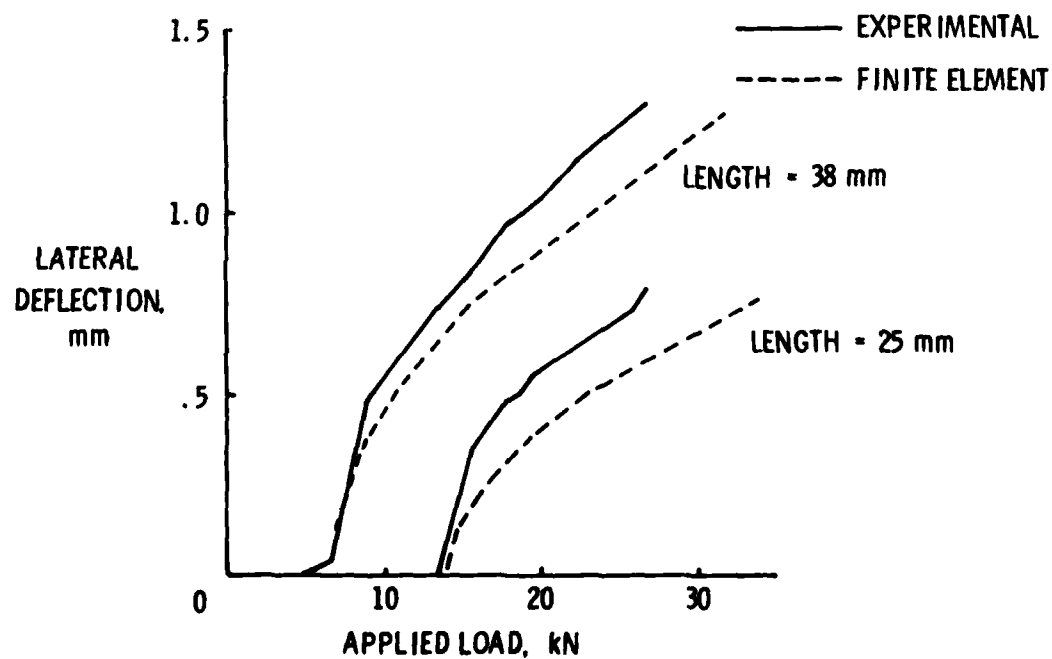


Figure 2.- Comparison of measured and calculated lateral deflections.

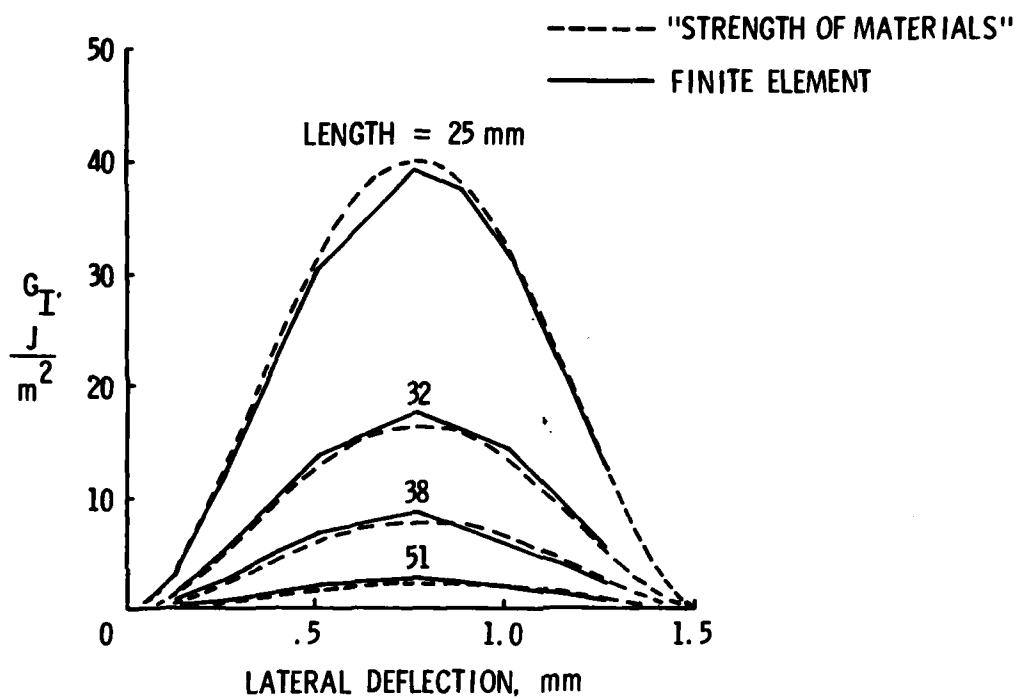


Figure 3.- Effect of lateral deflection on G_I for several delamination lengths.

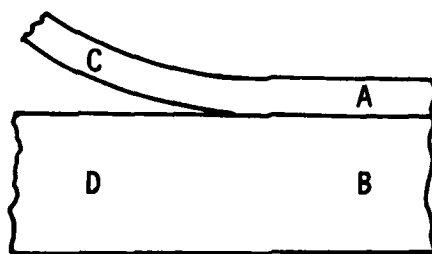


Figure 4.- Schematic of crack tip region.

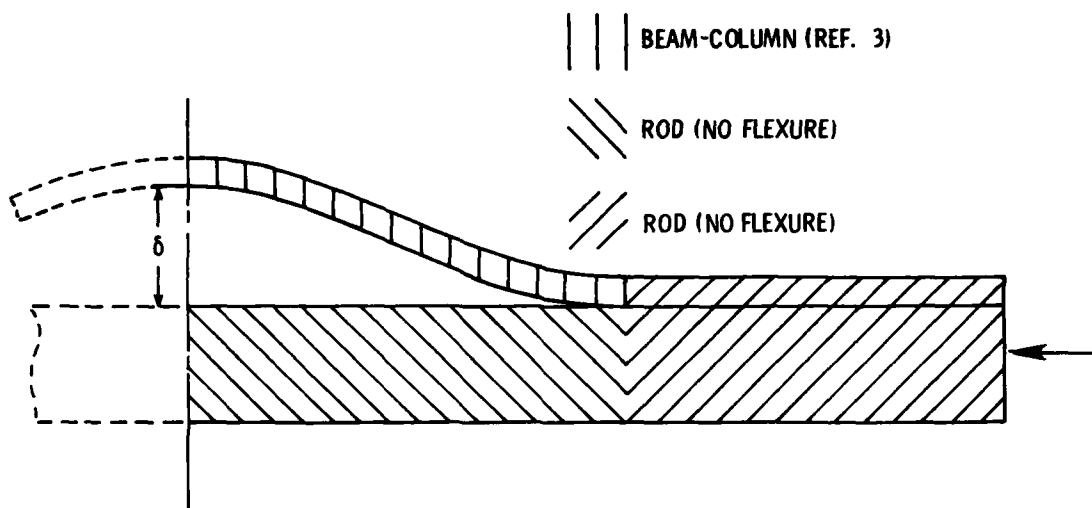


Figure 5.- Idealization of laminate with buckled through-width delamination.

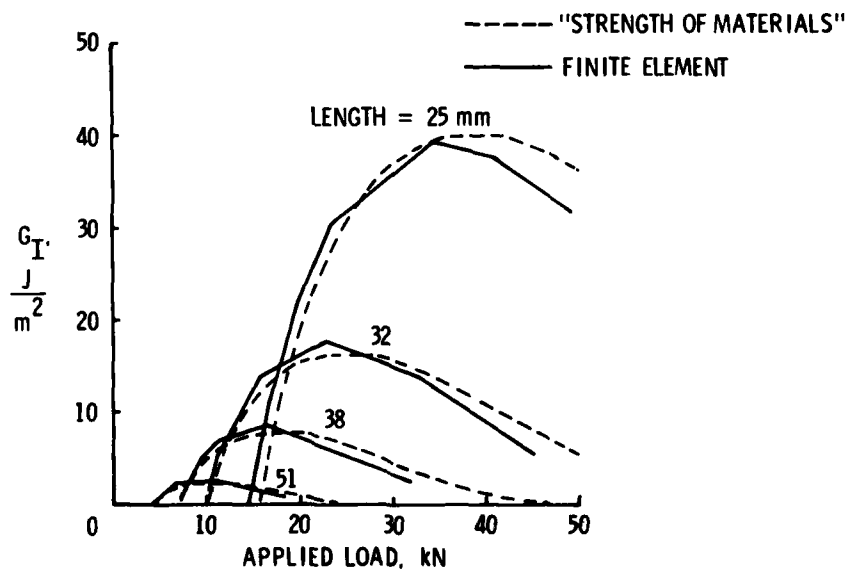


Figure 6.- Effect of load on G_I for several delamination lengths.

AN APPROXIMATE STRESS ANALYSIS FOR DELAMINATION GROWTH IN UNNOTCHED COMPOSITE LAMINATES

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Unnotched composite laminates have been observed to delaminate under fatigue loading. Delamination affects laminate stiffness, residual strength, and fatigue life. Consequently, any realistic fatigue analysis must account for the presence and growth of delaminations. To formulate such a fatigue analysis, a stress analysis must be developed first. Ideally, the stress analysis should be complex enough to reflect the changing stress state as delaminations initiate and grow, yet should be simple enough for practical and economical use.

The stress analysis developed was formulated with the assumptions used in references 1 and 2. First, as shown in figure 1(a), only one laminate cross section, which was assumed to be typical along the laminate length, was analyzed. Because of symmetry, only half of the laminate cross section was analyzed. Each ply was assumed to behave as a membrane with a displacement field of

$$\left. \begin{aligned} u &= \epsilon_0 x + U(y) \\ v &= V(y) \end{aligned} \right\} \quad (1)$$

where ϵ_0 was a prescribed uniform axial strain. Second, as shown in figure 1(b), the plies were assumed to be connected with a thin matrix layer that underwent only shear deformation. Third, the shear forces between plies were assumed to act as body forces on the plies. By using these assumptions, the equilibrium equations for each node in the interior and at the boundaries were developed. These equations were written in terms of nodal displacements with central finite difference operators. The node arrangement for the operators is shown in figure 1(c). Variable nodal spacing operators were used to refine the mesh near the straight edge and delamination fronts. The finite difference expressions resulted in a system of simultaneous algebraic equations in U and V . Once the equations were solved, central difference operators were used to calculate ply strains and stresses.

Figure 2 shows the calculated distribution of σ_x and σ_y stresses near the straight edge in the innermost -30° ply of a $[\pm 30/\pm 30/90/90]_s$ laminate. The laminate was subjected to a unit

axial strain ($\epsilon_0 = 1$). Also shown in figure 2 are similar distributions calculated from a more rigorous, uniform-axial-extension, finite-element analysis (ref. 3). The finite-element analysis allowed gradients in displacements through the ply thickness. The inplane stress distributions from the two analyses were in good agreement.

Figure 3 shows the calculated distribution of interlaminar shear stresses, τ_{xz} and τ_{yz} , near the straight edge at the innermost +30/-30 interfaces. Also shown in figure 3 are similar distributions calculated from the finite-element analysis. Finite difference τ_{xz} distributions agreed well with finite-element analysis distributions. However, τ_{yz} distributions did not agree because the finite-element analysis averaged stresses for the plies above and below the interface to give stress distributions at the interface. This averaging forced τ_{yz} to approach zero near the straight edge. In contrast, the finite difference analysis did not force τ_{yz} to go to zero.

Delamination growth between plies was simulated by setting the shear modulus of the appropriate interface equal to zero in the governing equilibrium equations. By using this simple technique, solutions for various size delaminations, from undamaged to totally delaminated, could be generated in a single computer run. Furthermore, several delaminations could be simultaneously grown in different interfaces.

Figure 4 shows the σ_x stress distribution across the half-width in the innermost -30° ply of a $[\pm 30/\pm 30/90/90]_s$ laminate. The laminate contains a partial delamination in both -30/90 interfaces. The axial stresses remained nonuniform near the straight edge after the delamination had grown into the width. Figure 4 also shows the change in axial stress between the laminated and delaminated regions of the laminate. Increased mesh refinement near the delamination front indicated that the stress distributions were very steep. In realistic laminates which contain delaminations, knowledge of such stress distributions may be needed to predict failure.

The finite difference analysis was also used to calculate axial stiffness. The axial stiffness of an unnotched laminate was calculated by

$$E_{xx} = \frac{1}{N\epsilon_0} \sum_{k=1}^N (\bar{\sigma}_x)_k \quad (2)$$

where $(\bar{\sigma}_x)_k$ was the average axial stress in the k^{th} layer of an N-ply laminate.

The average axial ply stress was given by

$$(\bar{\sigma}_x)_k = \frac{1}{b} \int_0^b (\sigma_x)_k dy \quad (3)$$

where the integral represents the area under the axial stress distribution. An exact expression was derived as

$$(\bar{\sigma}_x)_k = C_{11}\epsilon_o + \frac{C_{12}}{b} v_b + \frac{C_{13}}{b} u_b \quad (4)$$

where C_{11} , C_{12} , and C_{13} are elements of the stiffness matrix and u_b and v_b were displacement functions evaluated at the straight edge. As delamination growth was simulated in the finite difference analysis, u_b and v_b changed. Consequently, laminate stiffness changed. The analysis indicated that stiffness decreased linearly with delamination size. Quasi-static tension tests were performed on four $[\pm 30/\pm 30/90/90]_s$ graphite/epoxy laminates. As shown in figure 5, the delaminated area was measured from X-ray photographs and used to calculate an effective delamination size, a , to compare with the analytical prediction. The stiffness data agreed with the analytical predictions.

To summarize, the analysis developed can easily simulate delamination growth at one or more interfaces between plies. Solutions for various size delaminations, from undamaged to totally delaminated, can be generated in a single run. The corresponding effect of delamination growth on laminate stiffness and stress distribution within individual plies can be determined.

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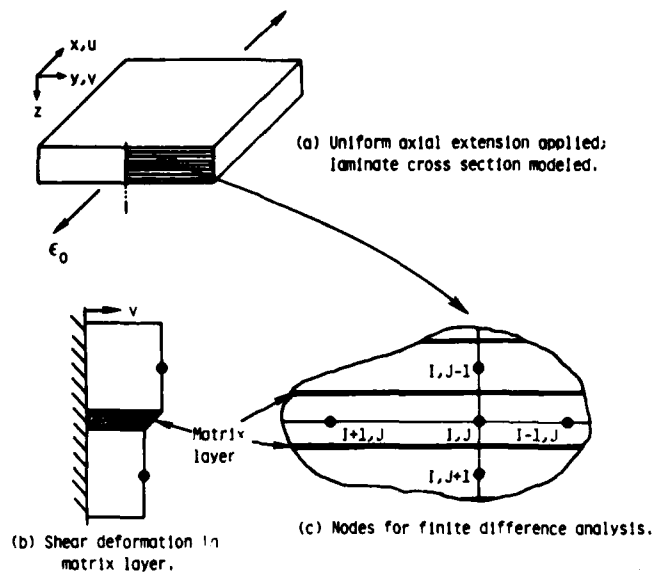


Figure 1.- Finite difference/shear-lag model.

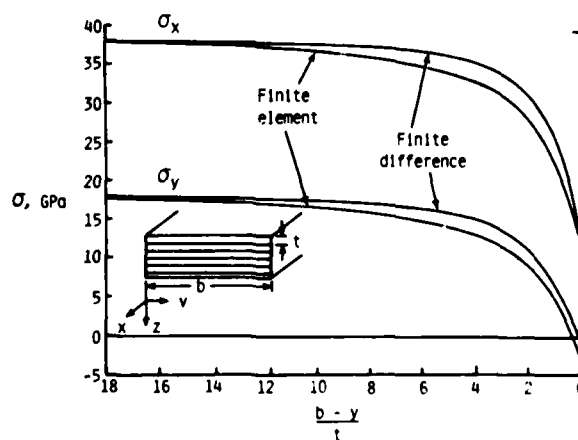


Figure 2.- Through-width σ_x, σ_y distributions in innermost -30° ply near the straight edge.

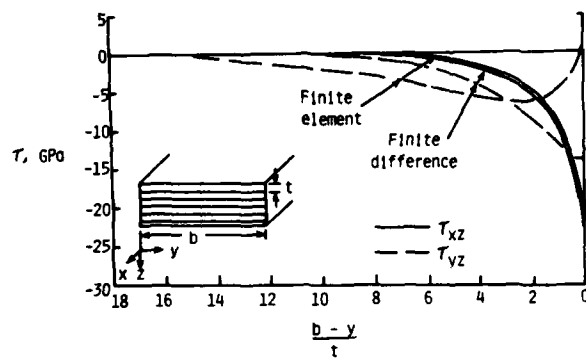


Figure 3.- Through-width interlaminar shear stress distributions at innermost +30/-30 interface near the straight edge.

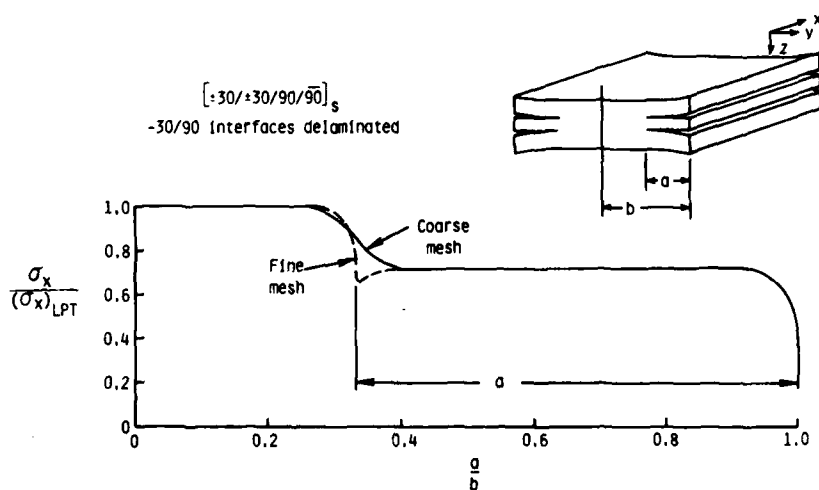


Figure 4.- Through-width σ_x distribution in innermost -30° ply.

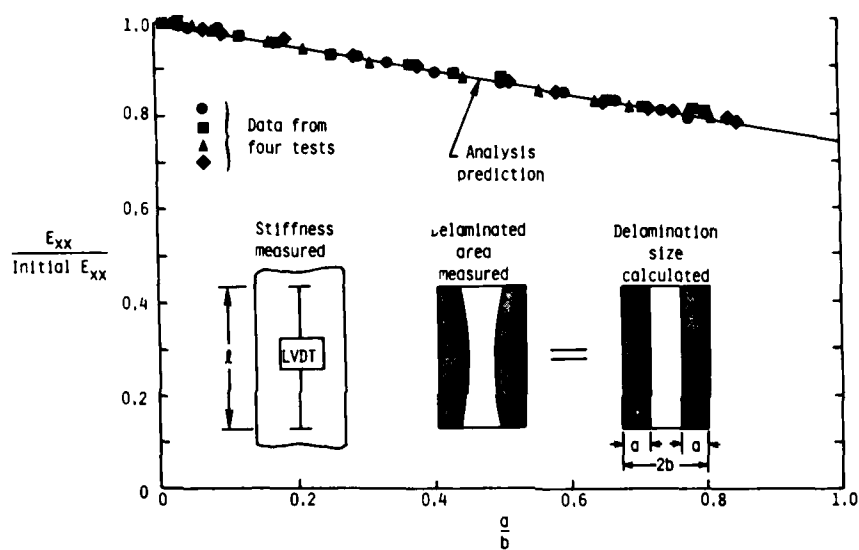


Figure 5.- Stiffness loss as a function of delamination size.

ENVIRONMENTAL EFFECTS ON COMPOSITES

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Environmental effects research at NASA Langley Research Center focuses on understanding the aging of composite materials during service in such widely different applications as aircraft structures, elevated temperature systems, and spacecraft systems. Research on composites for aircraft structures includes flight service exposure of structural components of graphite/epoxy and Kevlar/epoxy composite materials and ground, flight, and laboratory exposure of coupon specimens of these materials. Elevated temperature research on composites includes study of graphite and boron fibers in epoxy, polyimide, and aluminum matrices exposed to simulated supersonic cruise vehicle environments and study of titanium matrix and glass matrix composites. The space environment research on composites is aimed at defining the effects of electron, proton and ultraviolet radiation on the long-term properties of resin matrix composites.

Aircraft Service Effects

The Langley Research Center initiated a flight service evaluation program in 1972 to determine the long-term performance of boron-, Kevlar-, and graphite-fiber reinforced composite materials in flight environments. In the program components are selected for the design, fabrication, test, and certification using composite materials. Early applications of composite materials were for selective reinforcement of military aircraft structures such as the CH-54B tail cone and the C130 center wing box. More recently emphasis has been on evaluating composite material components on commercial transport aircraft. The program currently has 142 composite material articles comprised of six different components in flight service (reference 1). Composite components are inspected periodically by the aircraft operators and the manufacturers to check for damage and/or defects requiring repair. Although minor disbonds, impact damage and corrosion problems have been detected, the flight service components have generally performed outstandingly.

In addition to service evaluation of composite components, the composite spoiler portion of the flight service program includes periodic removal of randomly selected units for detailed inspection.

tion and measurement of moisture gain and residual strength. After five years of service the level of moisture in the graphite/epoxy skins is about 0.70 percent by weight of the composite and the room-temperature strength obtained through structural loading of the spoilers is unchanged.

As an adjunct to the flight service program, Langley Research Center has a ten-year study underway that involves about 17,000 composite material coupon test specimens undergoing environmental testing in ground, flight and laboratory exposure activities (reference 2). These studies are broadening the data base for environmental effects on composites for aircraft structures and focusing on developing accelerated environmental testing procedures. The ground exposure activities are underway at ten exposure sites around the world. The flight exposure activities consist of coupon specimens mounted within vented compartments and on external surfaces of commercial aircraft.

Moisture absorption in ground exposure specimens reaches equilibrium within two to three years with equilibrium moisture levels ranging from about 0.5 percent for T300/5209 graphite/epoxy to about 2 percent by weight of composite for Kevlar/epoxy and T300/2544 graphite/epoxy. The matrix dominated room temperature properties of interlaminar shear and compression strength experienced a ten percent reduction compared to baseline properties after five years of ground exposure.

Moisture absorption in specimens exposed on external surfaces of aircraft show a seasonal variation ranging from about 0.5 percent to 1 percent. Specimens mounted internally to the aircraft do not show a seasonal variation.

Results from laboratory exposure of specimens under controlled conditions show that equilibrium moisture levels are reached after about 25 days exposure at relative humidity levels up to 75 percent and 322 K temperature where substantially longer time is required to achieve the higher equilibrium levels attained on exposure at 95 percent relative humidity and 322 K temperature. The equilibrium moisture level is strongly related to the relative humidity of the exposure environment and ranges from about 0.15% at 40% relative humidity to 1.3% at 95% relative humidity. These data are consistent with data reported in the literature (reference 3).

Laboratory exposure of graphite/epoxy composites to ultraviolet radiation for more than six months has shown that standard aircraft paint will effectively protect the composite materials from surface degradation. Painted specimens lost about one-third as much mass as unpainted specimens during the six month's exposure.

Elevated Temperature Effects

Research on composites at elevated temperature is aimed at defining the maximum use temperature of conventional composite materials in long-term applications at temperatures above those encountered in commercial aircraft. Research is also underway to develop advanced composite materials such as titanium matrix and glass matrix composites which may be suitable for use at 800 to 900 K for 1000 hours. Projected applications of these materials include future aircraft such as supersonic and hypersonic transports and advanced space vehicles.

For the NASA Supersonic Cruise Research Program, a continuing study is characterizing five classes of advanced composite materials for up to 50,000 hours of exposure to simulated supersonic cruise environments (reference 4). Data to this point have produced the following conclusions:

1. The maximum use temperature for each material for a 10,000 hour design life is substantially lower than the limiting use temperature for short term applications.
2. For the resin matrix systems, matrix degradation by oxidation was shown to be the primary cause of mechanical property losses during thermal aging. Absorption of moisture by the epoxy systems caused a significant decrease in short time elevated temperature strength while the graphite/polyimide system was affected to a lesser degree.
3. The boron/aluminum system experienced thermal aging strength decreases resulting from degradation of the boron fiber. Uncoated boron/aluminum is subject to pitting and intergranular corrosion cracking during long-term exposure to an industrial-sea-coast environment.
4. Fatigue strength was found to be dependent on the stress ratio used in testing with higher strength obtained for tension-tension loading. Notched specimens generally had lower strengths; however, some exceptions were noted at elevated temperature and additional studies are warranted. Aluminum matrix specimens tested in fatigue at 560 K experienced severe matrix degradation.

Borsic/Titanium composites research at Langley Research Center has verified good stiffness properties for that material with a potential useful life to 920 K (reference 5). Exposures of Borsic/Titanium specimens to 920 K for 240 hours did not degrade the strength or modulus. However, exposure of Borsic/Titanium at temperatures of 1033 K and higher produced significant reductions in strength and modulus. Electron microprobe and X-ray diffraction analyses of the interface region in specimens exposed at 1144 and 1255 K suggest a two-stage reaction process con-

sisting of (1) the simultaneous interdiffusion of Si, C and Ti resulting in the depletion of the SiC coating and formation of titanium silicides, and (2) significant titanium boride formation. The strength of the composite was degraded before the formation of any identifiable boride compounds indicating that titanium-silicon reactions must also be minimized for good strength retention. SiC/Ti composite material is also under study with performance being analyzed in terms of fiber characteristics, matrix characteristics, and fiber-matrix interaction.

Research on graphite/glass composite materials has shown them to have good resistance to thermal cycling and high temperature oxidation (reference 6). Graphite/glass composites have demonstrated the potential for continuous use at temperatures as high as 810 K in a non-oxidizing atmosphere. Continuous use of graphite/glass composites in an oxidizing atmosphere is limited to 710 K by oxidation of the graphite fibers. Additional advantages of graphite/glass composites compared to resin matrix and metal matrix composites include high specific modulus, high specific strength at elevated temperature, and low thermal expansion.

Space Environmental Effects

The Langley Research Center has a program underway to determine the durability of advanced resin matrix composite materials subjected to long-term exposure to the space environment and to identify the damage mechanisms present during radiation exposure of composite materials (reference 7). Emphasis in the program has been on defining the radiation threshold for damage that affects mechanical properties of 450 K cure graphite/epoxy and graphite/polysulfone materials and examining critical aspects of radiation test methodology for composite materials. Preliminary results indicate that the threshold for radiation damage in composites cured at 450 K is higher than 5×10^9 rads. However, those data were generated under high acceleration rates and the validity of accelerated space radiation exposure testing of composite materials has not been ascertained.

The space environmental effects program at Langley Research Center is being expanded to increase the in-house capability for testing to include: combined radiation exposure of materials to electrons and protons at energy levels up to 1.0 Mev and 2.5 Mev, respectively, in-situ thermal expansion measurement of composite materials, and the study of 394 K-cure composite materials.

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POSTBUCKLING STRENGTH OF STIFFENED FLAT 24-PLY GRAPHITE-EPOXY PANELS LOADED IN COMPRESSION

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INTRODUCTION

Current metal aircraft design practices allow the skin panels of some structural components (e.g., fuselage and stabilizer panels) to buckle under various loading conditions and these structural components are designed to have postbuckling strength. Before advanced-composite structural components can be designed with similar postbuckling response, their strength limits and failure characteristics must be well understood. Most previous work on the postbuckling behavior of composite structures (e.g., Refs. 1-4) has focused on analytical solutions to classical unstiffened orthotropic plate problems. Only a limited amount of data has been published (e.g., Ref. 5) that compares test results with analytical prediction for unstiffened composite plates or that describes the postbuckling behavior of stiffened composite panels loaded in compression. Also, the compression strength of buckling resistant graphite-epoxy panels has been shown (e.g., Refs. 6 and 7) to be sensitive to low-speed impact damage, but only limited data have been published (Ref. 5) that describe the influence of these local effects on the postbuckling response of composite structures. The present paper describes the results of postbuckling tests on stiffened flat 24-ply graphite-epoxy panels loaded in compression, and describes the effects of low-velocity impact damage on the response of these panels.

SPECIMENS, APPARATUS AND TESTS

The specimens tested in this investigation were fabricated from commercially available 450K cure 0.14-mm-thick T300/5208 graphite-epoxy preimpregnated tapes. The tapes were laid up on the appropriate tooling to form flat I-stiffened panels with 24-ply skins and stiffeners with the cross section and stacking sequences shown in Figure 1. The specimens were autoclave cured, ultrasonically inspected, and cut to the desired sizes. The ends of the specimens were potted in an epoxy resin and then ground flat and parallel to permit uniform compression loading. Stiffener spacing was varied to determine the effect of skin postbuckling response on structural performance. All specimens had four equally-spaced stiffeners; three specimens had 10.2-cm stiffener spacing, and two specimens had 17.8-cm stiffener spacing. The panels with 10.2-cm stiffener spacing were 38.1 cm wide and 50.8 cm long, and the panels with 17.8-cm stiffener spacing were 61.0 cm wide and 81.3 cm long. The panel lengths were selected so the initial buckling modes would have at least four longitudinal half waves. The unstiffened side of the specimens were painted white to reflect light so a moire-fringe technique could be used to monitor out-of-plane deformations.

Test specimens were loaded in axial compression using a 4.45-MN capacity hydraulic testing machine. The specimens were flat end tested without any edge supports and with only the epoxy potting material to stabilize the loaded ends. Electrical resistance strain gages were used to monitor strains, and direct-current differential transformers were used to monitor displacements at selected locations. Buckling was determined by the load-strain response of the specimens and by the strain-reversal technique. The strain measurements were complemented by the moire-fringe method which provided a visual definition of out-of-plane deformations.

Both undamaged and low-velocity impact damaged specimens were tested in static compression to determine their postbuckling response. Low-velocity impact damage was introduced in the specimens by 1.27-cm-diameter aluminum spheres propelled by the air gun described in Reference 6. One undamaged specimen with each stiffener spacing was loaded to failure to establish a reference response. Three additional specimens were impacted in two places and then loaded to failure. Each of the damaged specimens was impacted once in the skin midway between two stiffeners and once in the skin at a stiffener attachment flange. Two specimens with 10.2-cm stiffener spacing were impacted at nominal speeds of 67 and 100 m/s, and one specimen with 17.8-cm stiffener spacing was impacted at 67 m/s.

RESULTS AND DISCUSSION

The initial buckling of the undamaged panel with 10.2-cm stiffener spacing occurred at an applied load of 834 kN (longitudinal strain = 0.0052) and the panel failed at 934 kN (longitudinal strain = 0.0061). The buckling mode is shown in Figure 2a by the moire-fringe pattern. The panel failed when the stiffeners separated from the skin in the region of where the skin buckling mode has large out-of-plane deflections (see Fig. 2b). Initial buckling of the undamaged panel with 17.8-cm stiffener spacing occurred at an applied load of 389 kN (longitudinal strain = 0.0022) and the panel failed at 656 kN (longitudinal strain = 0.0039). The initial buckling mode and failure mode of the panel with 17.8-cm stiffener spacing was similar to those of the panel with 10.2-cm stiffener spacing. The maximum amplitude of the skin deformation was much larger at failure for the panel with 17.8-cm stiffener spacing than for the panel with 10.2-cm stiffener spacing, suggesting that the skin-stiffener interface region is highly loaded at failure when the stiffener spacing is wider.

The impact damage considered in this investigation was caused by projectiles with nominal speeds of 67 and 100 m/s. No visible front-surface damage was caused by projectiles with 67 m/s impact speeds, but some visible front-surface cratering was caused by projectiles with 100 m/s impact speeds. Both impact speeds caused visible back-surface damage. Cracks in the skin between stiffeners caused by an impact speed of 65.8 m/s are shown in Figure 3a for the panel with 10.2-cm stiffener spacing, and cracks in the stiffener attachment flange at the skin-stiffener interface caused by an impact speed of 67.7 m/s are shown in Figure 3b. The higher impact speeds caused similar, but more extensive, back-surface damage. Ultrasonic C-scans of the impacted regions indicated that the damage was localized to regions that are approximately 3 to 4 cm in diameter.

Both impact-damaged panels with 10.2-cm stiffener spacing failed before initial skin buckling. The panel impacted between two stiffeners at 65.8 m/s and at the skin-stiffener interface at 67.7 m/s failed at an applied load of 552 kN (longitudinal strain = 0.0036). The panel impacted between two stiffeners at 102.7 m/s and at the skin-stiffener interface at 99.6 m/s failed at an applied load of 463 kN (longitudinal strain = 0.0033). The impact damage between stiffeners propagated before panel failure and was readily detectable in the moire-fringe patterns, while the impact damage in the skin-stiffener interface region was contained in the initial damaged area. The photograph in Figure 4 shows the moire-fringe pattern of the panel impacted at nominal speeds of 67 m/s just before panel failure with the damage in the skin between stiffeners propagated from stiffener to stiffener. Panel failure, however, occurred when the damage at the skin-stiffener interface region suddenly propagated across the entire panel as shown in Figure 5. The panel with 17.8-cm stiffener spacing impacted in the skin between stiffeners at 68.0 m/s and at the skin-stiffener interface at 68.9 m/s buckled at an applied load of 359 kN (longitudinal strain = 0.0024) and failed at an applied load of 530 kN (longitudinal strain = 0.0035). Failure occurred suddenly when the stiffeners separated from the skin and the failure mode propagated through the impact-damaged region in the skin between two stiffeners. These data suggest that impact-damaged compression panels may have some postbuckling strength, but not necessarily as much as corresponding undamaged panels.

CONCLUDING REMARKS

Test results indicate that selected stiffened graphite-epoxy compression panels can be loaded beyond initial skin buckling. The postbuckling response and ultimate loads of these panels are strongly influenced by the separation of the stiffeners from the skin. Low-velocity impact damage can significantly reduce the postbuckling strength of these stiffened compression panels by causing skin-stiffener separation to occur prematurely.

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2. Harris, G. Z.: Buckling and Post-Buckling of Orthotropic Laminated Plates. *AIAA Paper No. 75-813*, May 1975.
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7. Williams, J. G.; Anderson, M. S.; Rhodes, M. D.; Starnes, J. H., Jr.; and Stroud, W. J.: Recent Developments in the Design Testing and Impact-Damage Tolerance of Stiffened Composite Panels. NASA TM 80077, April 1979.

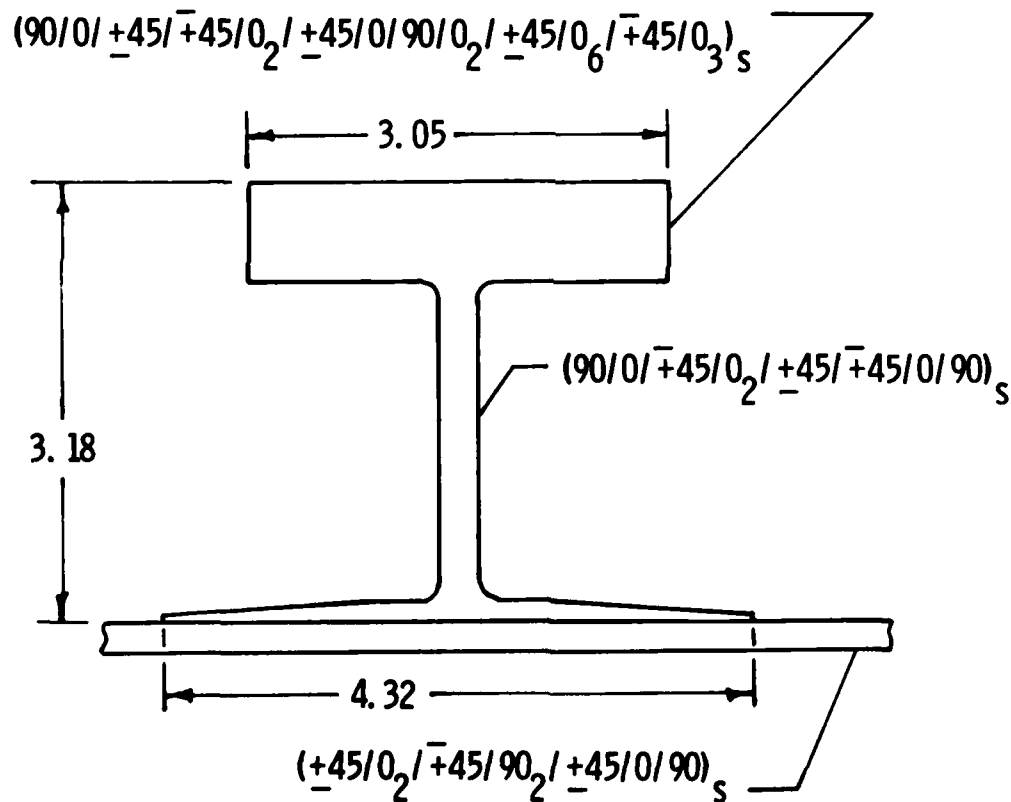
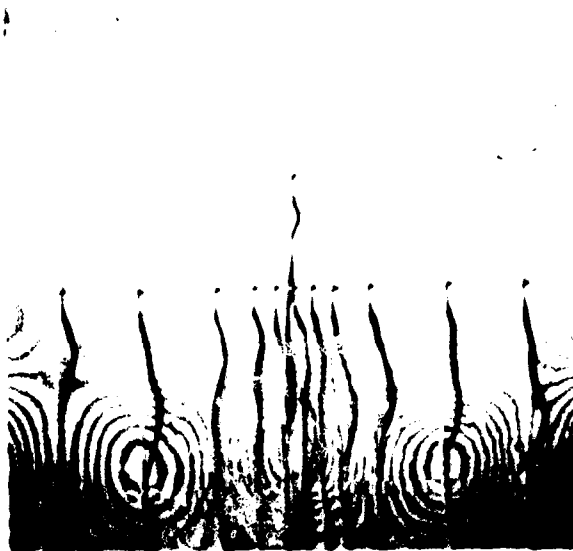


Figure 1. - Stiffener cross section and panel stacking sequences.



(a) Moire-fringe pattern of buckled skin.

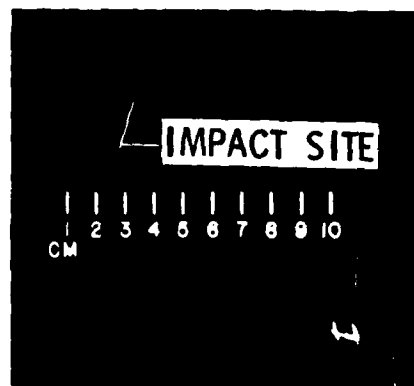


(b) Rear view of failed panel.

Figure 2. - Buckling and failure modes of undamaged panel with 10.2-cm stiffener spacing.



(a) Impact in skin midway between two stiffeners, impact speed equal to 65.8 m/s.



(b) Impact at skin-stiffener interface, impact speed equal to 67.7 m/s.

Figure 3. - Back-surface impact damage in panel with 10.2-cm stiffener damage.

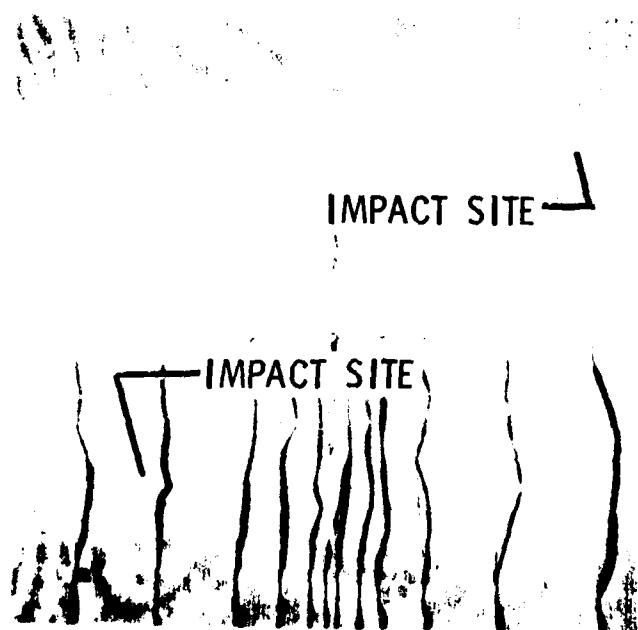
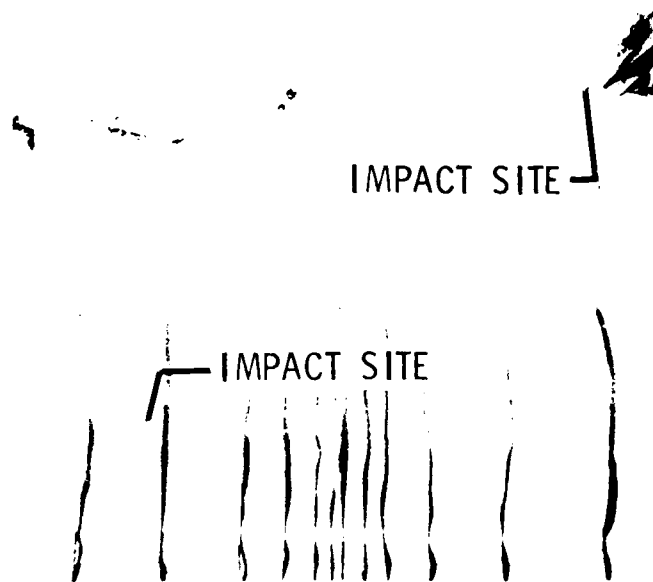
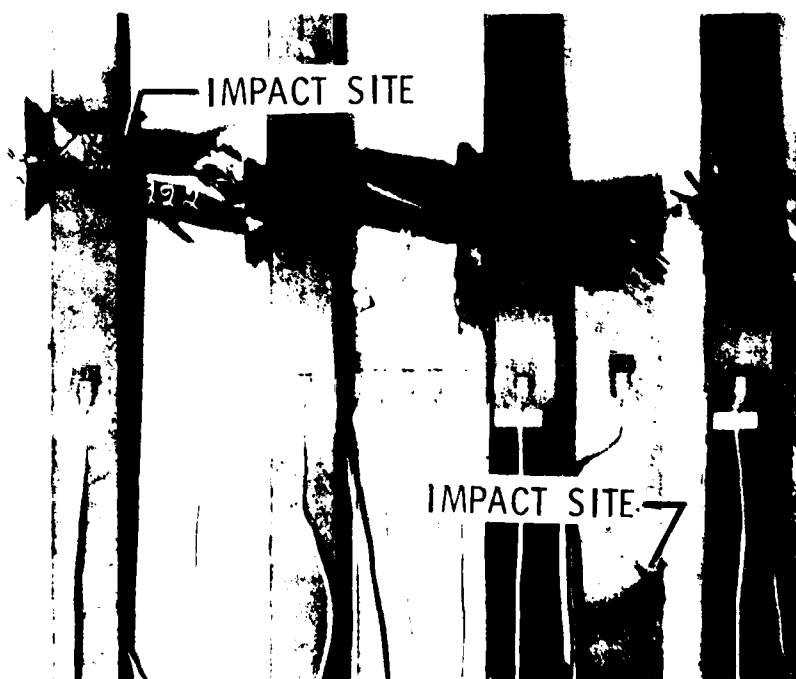


Figure 4. - Moire-fringe pattern of impact-damaged panel with 10.2-cm stiffener spacing. Applied load equal to 538.2 kN.



(a) Moire-fringe pattern of failed panel.



(b) Rear view of failed panel.

Figure 5. - Failure mode of impact-damaged panel with 10.2-cm stiffener spacing.

ENVIRONMENTAL AND HIGH STRAIN RATE EFFECTS
ON COMPOSITES FOR ENGINE APPLICATIONS

C. C. Chamis and G. T. Smith
NASA-Lewis Research Center
Cleveland, Ohio, 44135

The Lewis Research Center is conducting a series of programs intended to investigate and develop the application of composite materials to turbojet engines. A significant part of that effort is directed to establishing the impact resistance, defect growth, and strain rate characteristics of composite materials over the wide range of environmental and load conditions found in commercial turbojet engine operations. Both analytical and empirical efforts are involved. This paper summarizes the status of the major experimental contract programs and attendant in-house and grant activities which emphasize analytical methodology development. The three contract programs are with General Electric (GE) Company, Evendale, Ohio, Boeing Aerospace (BA) Company, Seattle, Washington, and IIT Research Institute (IITRI) Chicago, Illinois. The GE program (NAS3-21017) addresses the hygrothermal effects on the impact resistance of composites. The BA program (NAS3-20405) addresses the hygrothermal effects on defect growth in composites. The IITRI program (NAS3-21016) addresses the strain rate effects on composite mechanical properties.

The two grant programs are with Lehigh University (LU) and with Purdue University (PU). The LU grant program (NSG-3179) is directed towards the development of analytical methods to predict the dynamic stress intensity factor in composites. The PU grant program (NSG-3185) is directed towards the development of analytical models to describe the energy absorbed during impact of composites. The in-house program focused: (1) on the development of simplified equations to estimate the hygrothermal effects on composites, and (2) on the continued implementation of CODSTRAN (COMposite DURability STRuctural ANALYSIS). Progress on all these areas is briefly summarized below.

The comparisons between measured data and predicted results of hygrothermal effects on composite flexural and interlaminar longitudinal and transverse strengths are predicted by the simplified equation shown in Figure 1. The predicted results were obtained from

$$\frac{P_h}{P_o} = \sqrt{\frac{T_{gw} - T}{T_{gd} - T_o}}$$

where p_h is the property with hygrothermal effects, p_0 is the reference property, T_{gw} is the glass transition temperature of the wet composite, T_{gd} is the glass transition temperature of the "dry" composite at reference conditions, T is the temperature at which p_h is needed, and T_0 is the temperature at which the reference property p_0 was measured or known. The equation can be used either at the ply level or at the composites micro-mechanics level using corresponding properties for the resin. The correlation between predicted and measured data (fig. 1) is very reasonable for both longitudinal and transverse flexural and interlaminar (short-beam-shear) strengths. Using the equation at the micromechanics level improves the correlation (solid diamond points) for longitudinal flexural strength. Using the degraded properties in laminate theory and in structural analysis of wedge type impact specimens shows that failure occurs earlier than does in dry, room temperature conditions.

The hygrothermal effects on defect growth in composites are shown in Figure 2 for an interply hybrid angleplied laminate (0/30/0^S/-30/0/30/0^S/-30)_S (s denotes s-glass/resin). This angleplied laminate is suitable for fan blade applications for turbojet engines. The comparisons show that there is a defect sensitivity effect at the room temperature dry and room temperature wet conditions between the smooth (without defect) and 1/8 inch slit. The defect sensitivity is less severe with increasing defect size and with elevated temperature. Data for other non-hybrid angleplied laminates shows less sensitivity than that in Figure 2.

CODSTRAN pilot model analysis results are shown in Figure 3. The 70°F dry conditions show no damage at room conditions for both loadings. However, the specimen exhibits extensive damage at the 2500 pound load and fractured at the 3000 pound load for the indicated hygrothermal conditions. Though CODSTRAN is not yet completely implemented, the results of Figure 3 show that damage growth and fracture in angleplied laminates can be modeled using concepts and procedures embedded in CODSTRAN.

The strain rate effects on transverse and shear moduli of composites are shown in Figure 4. Three different types of transverse moduli are plotted. The strain rate effects on the transverse modulus are very large (up to 400 percent) and these are substantial (less than 70 percent) on the shear modulus for the strain rate investigated. Strain rate effects on the transverse tensile strength of composites are shown in Figure 5. The strain rate effects are also very large, ranging from about three to five times greater than the static values. These very large increases in the transverse tensile strength coupled with an earlier in-house investigation on in situ ply strengths lead to a speculation -- that in situ (in the angleplied laminate) transverse ply failure is a dynamic phenomenon occurring at high-strain rates. If this is indeed the case, then high strain rate transverse tensile strength need be used in predicting angleplied laminate strength based on first ply failure.

Results from the LU grant (not illustrated in this brief summary) show that both Mode I and II dynamic stress intensity factors are about 150 percent greater than the steady state and they occur very early in the impact event. Results from the PU grant indicate that simple mathematical models can be developed to quantify the energy absorbed during point impact. The models developed so far are for cantilever and simply supported beams and account for local indentation including unloading.

Researchers to be contacted for additional information on the above methodology areas are listed under Notes with telephone numbers. Publications with relevant background information are listed under Relevant References.

NOTES

For additional information on the contracts, grants and in-house programs discussed in the text, you can contact:

General Electric Co. Contract NAS3-21017 -- Guy Murphy, Tel. (513) 243-6918.

Boeing Aerospace Co. Contract NAS3-20405 -- Ted Porter, Tel. (206) 655-3090.

IIT Research Institute Contract NAS3-21016 -- Isaak Daniel, Tel. (312) 567-4402.

Lehigh University Grant NSG-3179 -- George Sih.

Purdue University Grant NSG-3185 -- C. T. Sun.

In-house programs and above contracts/grants monitoring -- G. T. Smith, Tel. (216) 433-4000, ext. 5103; R. L. Thompson, Tel. (216) 433-4000, ext. 5103; or C. C. Chamis, Tel. (216) 433-4000, ext. 6831.

RELEVANT REFERENCES

NASA-Lewis Research Center relevant publications:

1. C. C. Chamis, R. F. Lark and J. H. Sinclair: An Integrated Theory for Predicting the Hygrothermomechanical Response of Advanced Composite Structural Components. NASA TM-73812, 1977.
2. C. C. Chamis and T. L. Sullivan: In Situ Ply Strength: An Initial Assessment. NASA TM-73771, 1978.
3. C. C. Chamis and G. T. Smith: CODSTRAN: Composite Durability Structural Analysis. NASA TM-79070, 1978.

4. C. C. Chamis and G. T. Smith: Engine Environmental Effects on Composite Behavior. NASA TM-81508, 1980.

FIGURE 1.

PREDICTED HYGTHERMAL EFFECTS ON COMPOSITES CORRELATE WITH MEASURED DATA

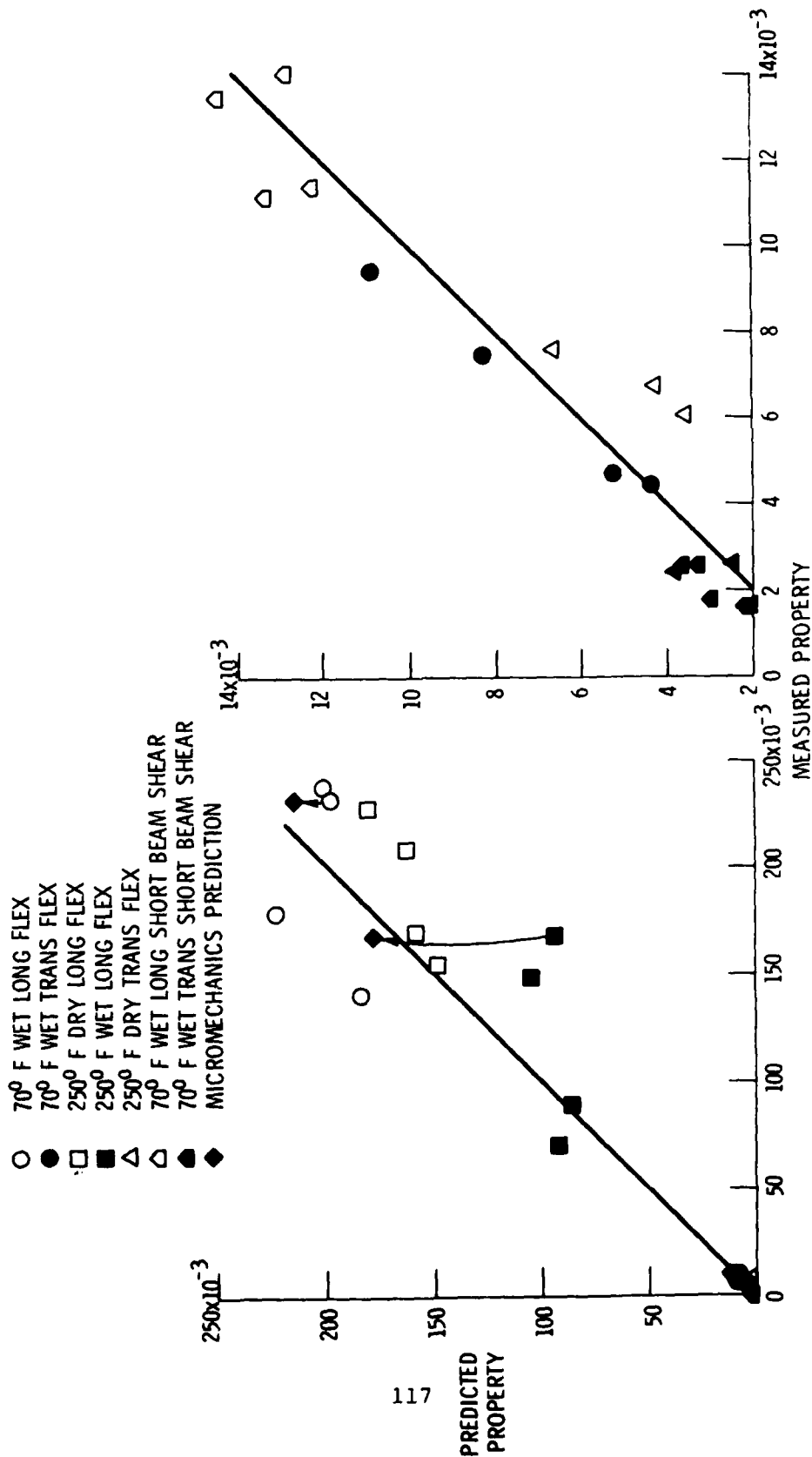
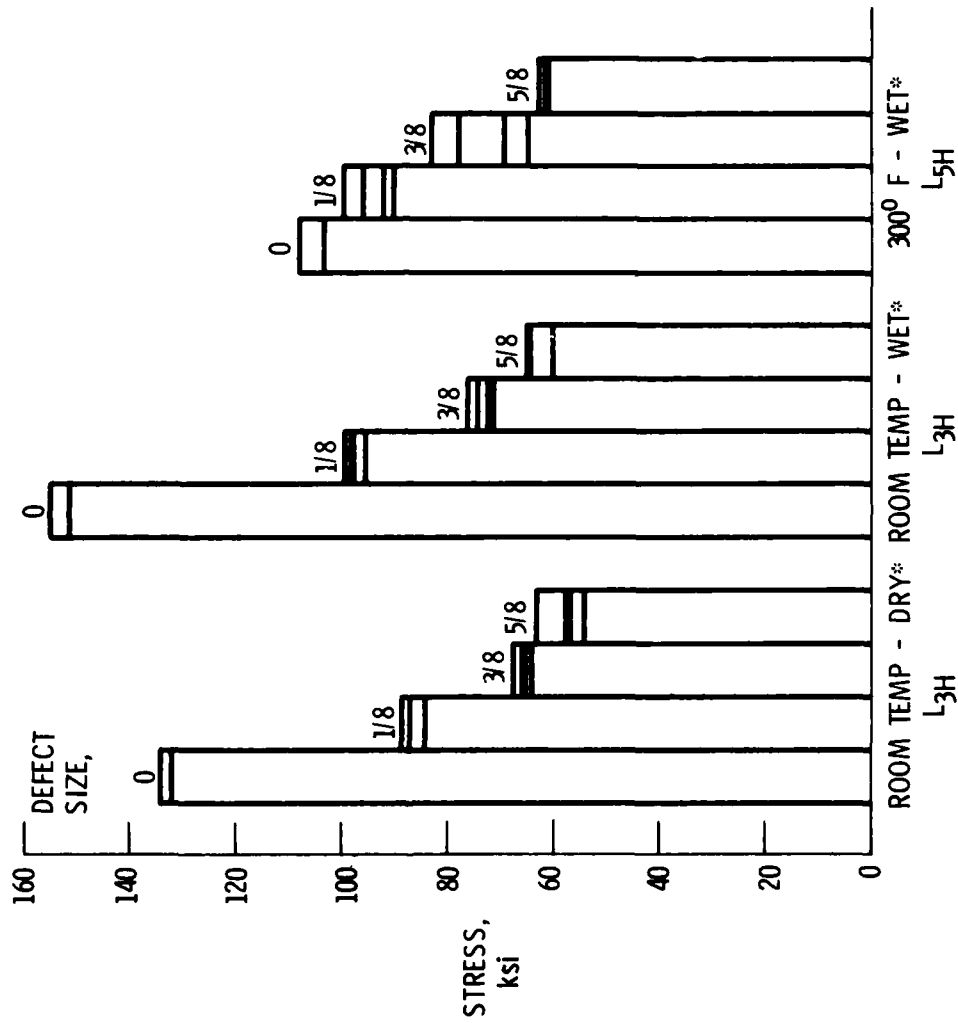


FIGURE 2.

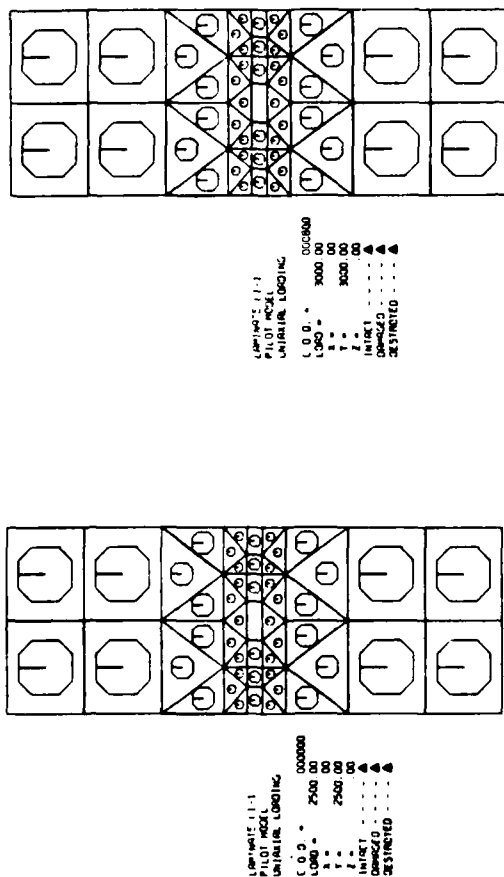
DEFECT SIZE AND HYGTROHERMAL EFFECTS ON COMPOSITE TENSILE STRENGTH
T-300/EPOXY; $[0/30/0^{\circ}/-30/0/30/0^{\circ}/-30]_s$



*WET $\approx$ 1.9% MOISTURE

FIGURE 3.
CODSTRAN PILOT MODEL RESULTS WITH ENVIRONMENTAL EFFECTS

700°F DRY



300°F WET
1.9% MOISTURE

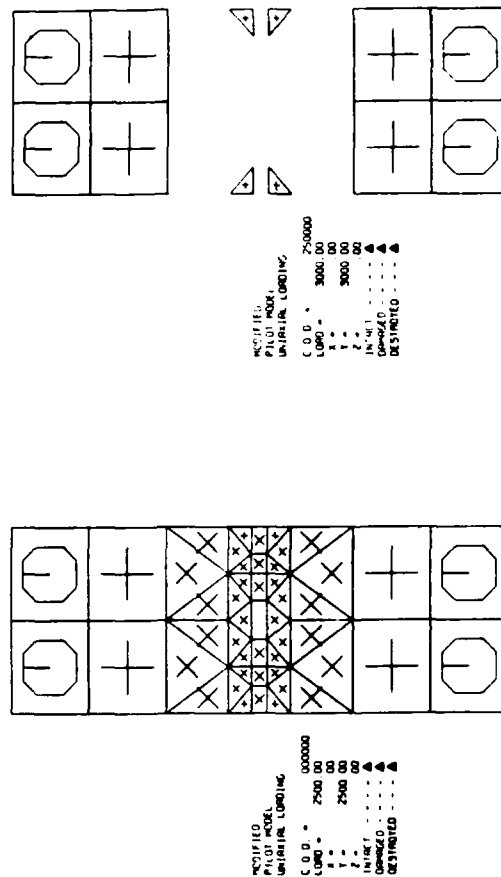


FIGURE 4.

STRAIN RATE EFFECTS ON COMPOSITE TRANSVERSE AND SHEAR MODULI
(AS/EPOXY)

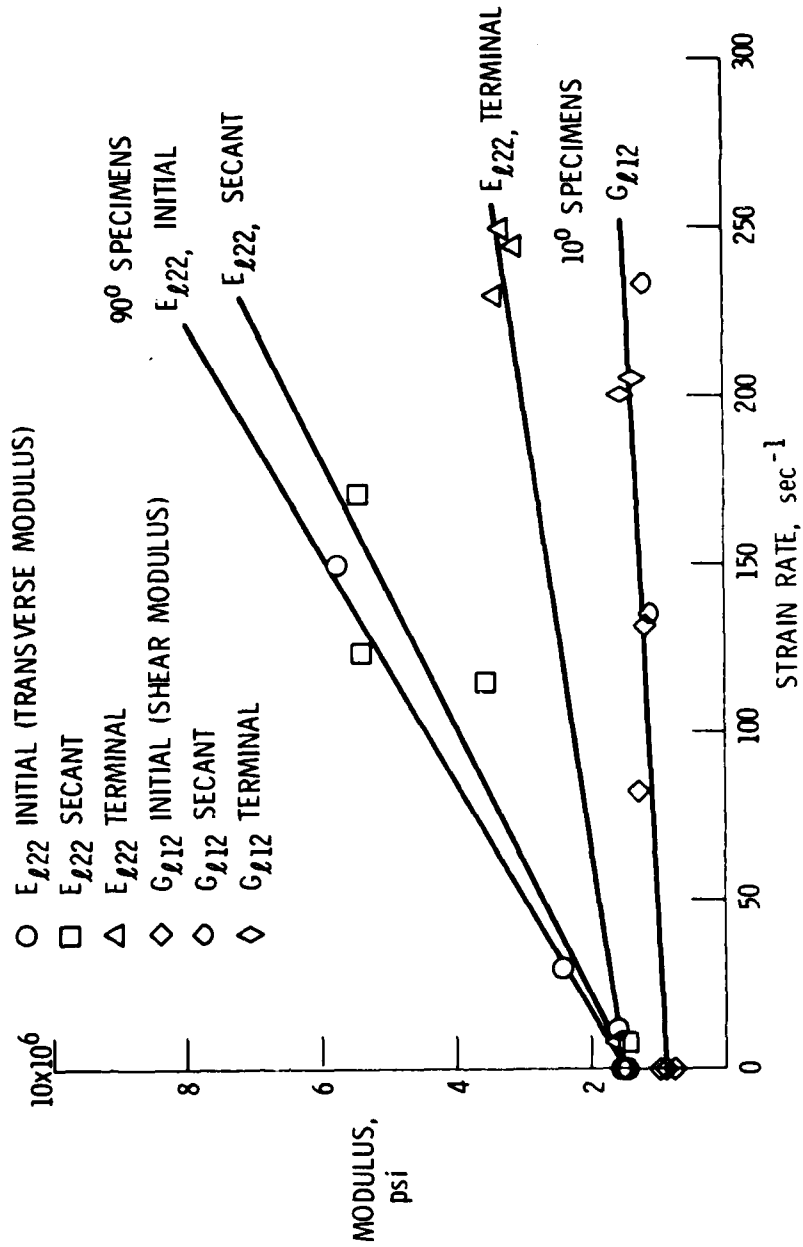
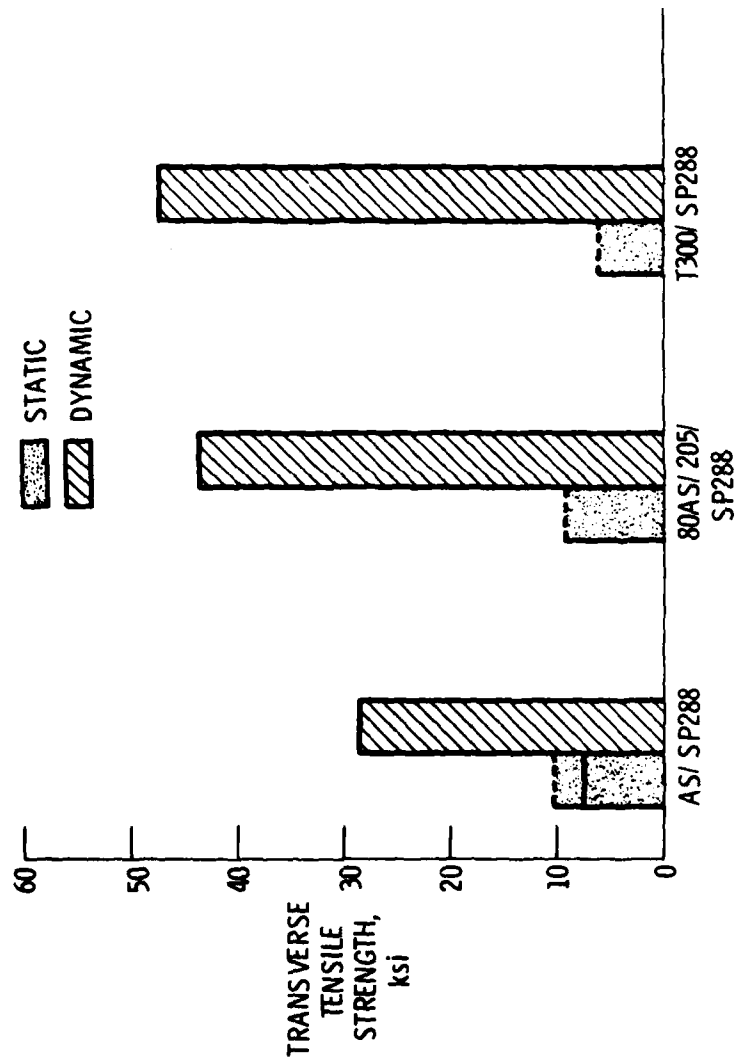


FIGURE 5.

STRAIN RATE HAS VERY LARGE EFFECT ON COMPOSITE TRANSVERSE TENSION STRENGTH

GRAPHITE FIBER/ EPOXY ABOUT 150 in/in/sec



FRACTURE BEHAVIOR OF GRAPHITE/EPOXY COMPOSITES
UNDER COMPLEX IN-PLANE LOADING

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R.W. Thomas, J. Tirosh and I. Wolock

Naval Research Laboratory
Washington, D. C. 20375, U.S.A.

ABSTRACT

Fracture tests were conducted on graphite/epoxy crossply laminates over a broad range of in-plane loads. The tests were conducted on small single-edge-notch test coupons in a unique in-plane loader developed at NRL (Figure 1). The type of loadings produced are shown in Figure 2 - tension or compression, shear, and bending. The tests were conducted over four octants of load space - tension and compression, positive shear, and positive and negative rotation.

The fiber reinforcement was laid up as a crossply; the respective included angles were 30° , 45° , 60° , and 75° .

The failure criterion used in the tests was the initiation of failure. This is defined as the point at which there is a significant increase in the non-recoverable strain energy in the specimen. This increase in the dissipated energy is attributed to the initiation of fracture. The vector sum of the various displacements at this point is referred to as the critical displacement, and the data is presented using this parameter for each of the loading conditions gained.

A typical plot of the data obtained is presented in Figure 3. Each vertical line represents the results of a single test. The reproducibility of the test data is apparent. The complexity of the fracture process in composites is apparent from the range of critical displacements observed as the loading conditions are varied. Failure surfaces can be prepared from this data and a typical surface is shown in Figure 4. Data will be presented showing failure surfaces for the other reinforcement angles studied. Techniques for the efficient utilization of the large amount of data obtained in these studies will be discussed.

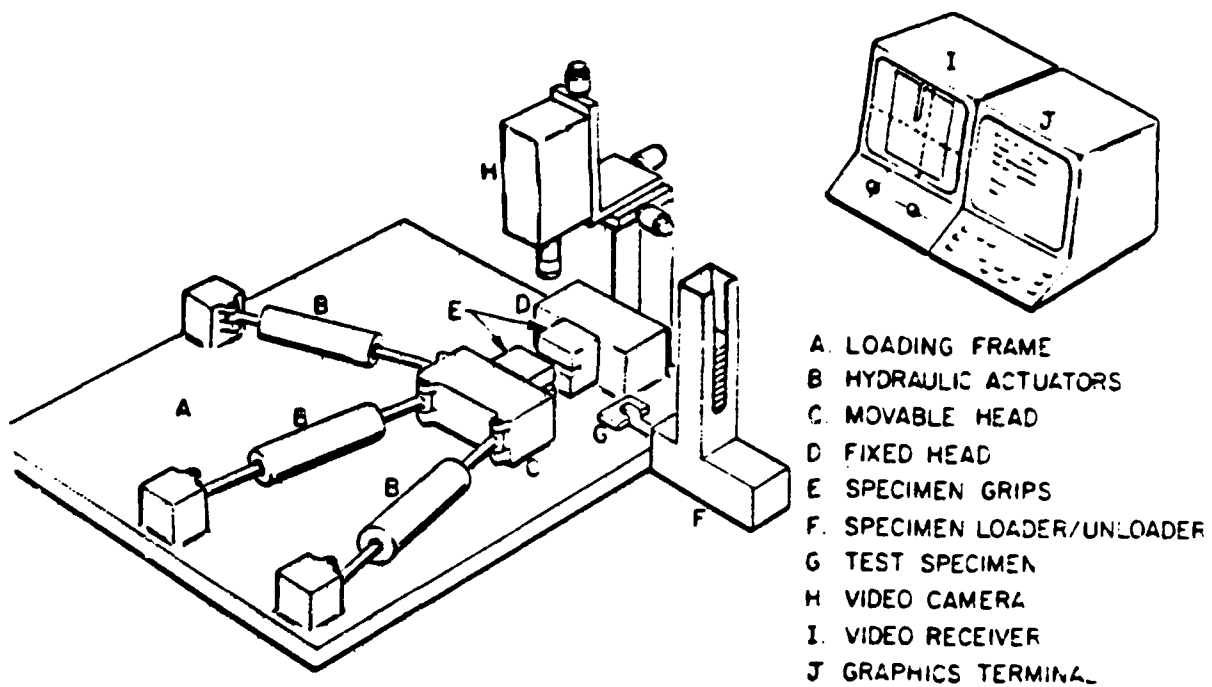


FIGURE 1. IN-PLANE LOADER

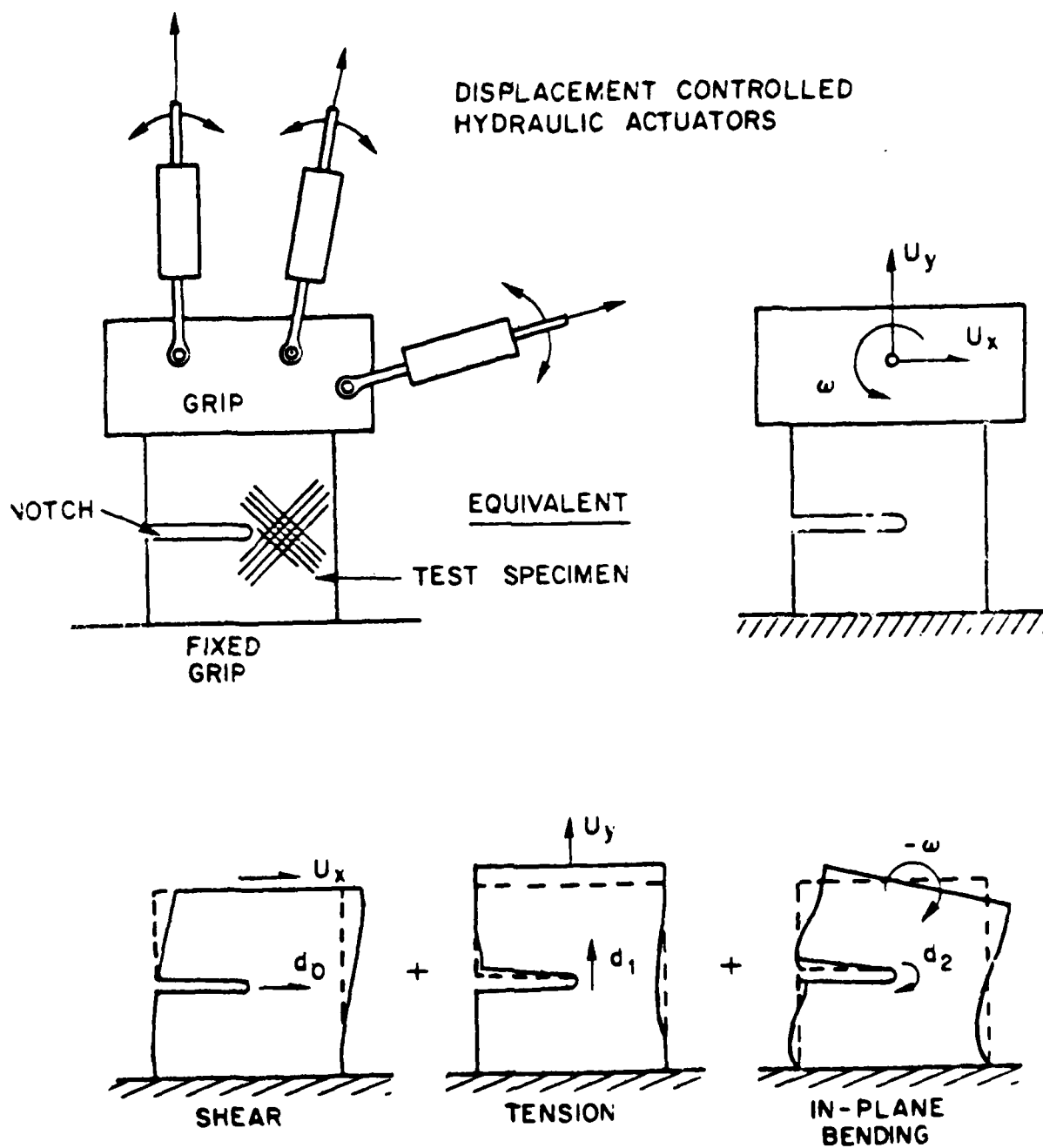


FIGURE 2. LOADING CONDITIONS PRODUCED BY IN-PLANE LOADER

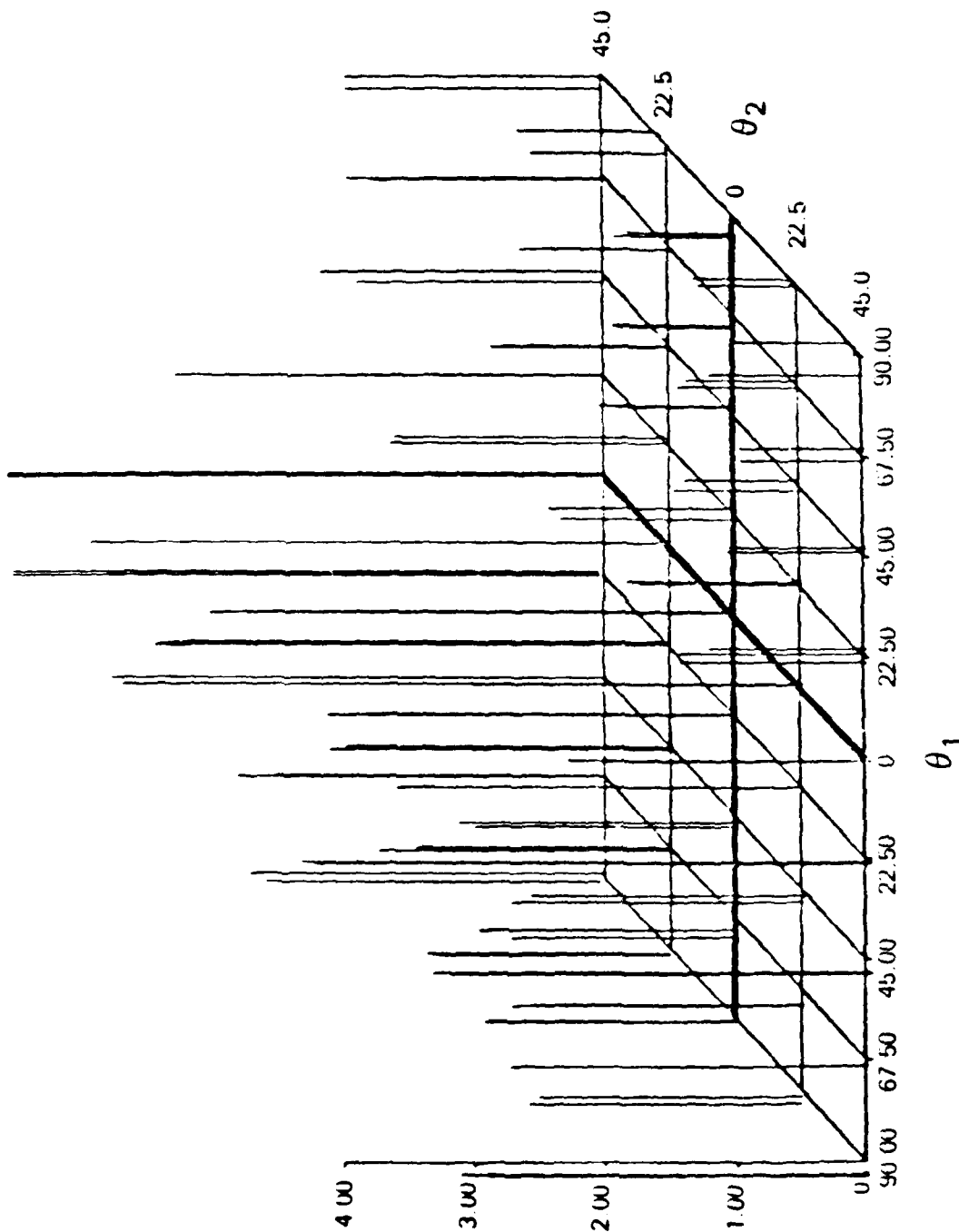
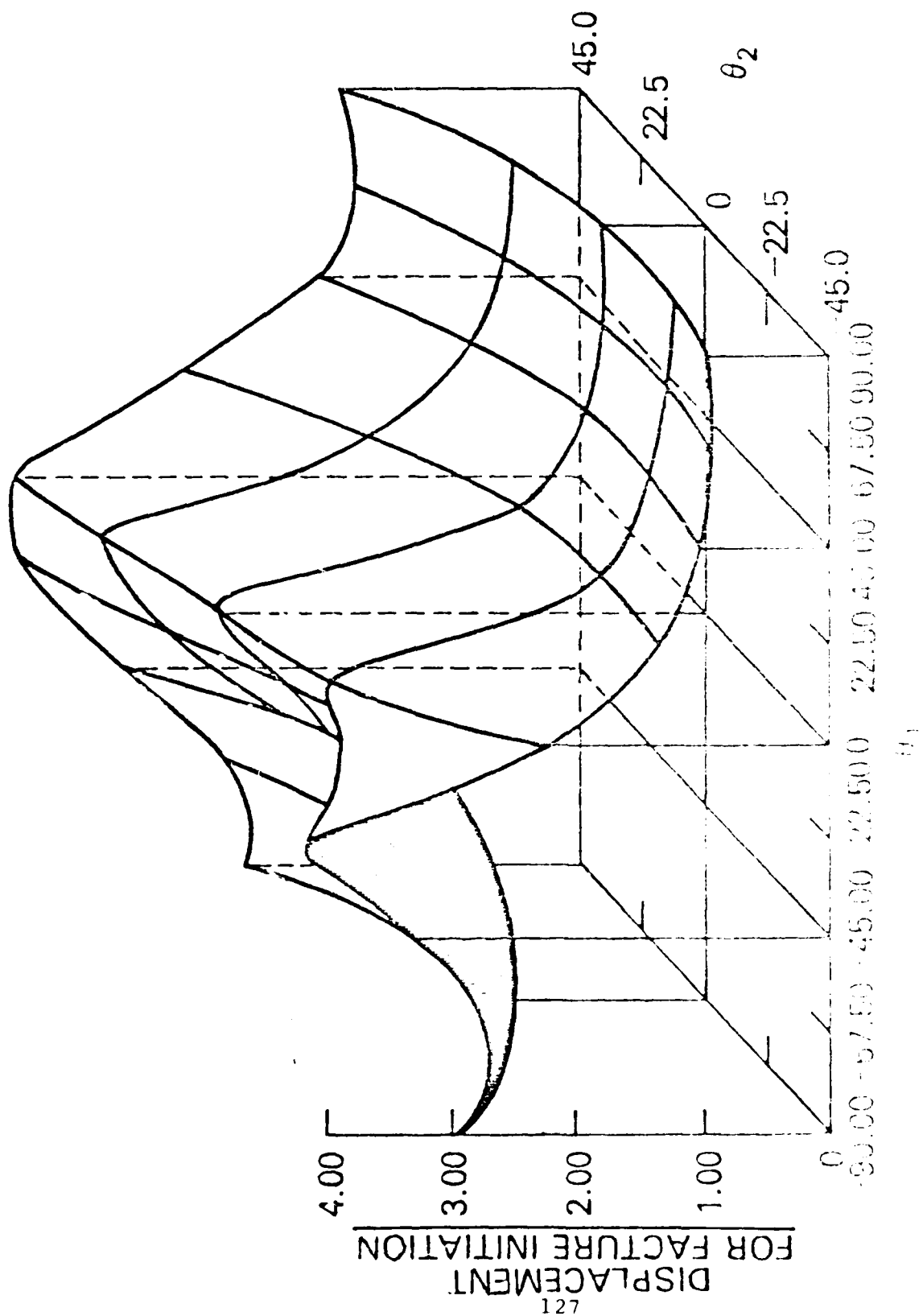


FIGURE 3. FAILURE DATA FOR GRAPHITE/EPOXY COMPOSITE UNDER COMPLEX LOADING FOR FOUR OCTANTS OF LOAD SPACE. VERTICAL LINES REPRESENT DISPLACEMENT FOR FRACTURE INITIATION. POINTS TO RIGHT OF ORIGIN ARE TENSION, TO LEFT ARE COMPRESSION. θ_1 IS RATIO OF TENSION TO SHEAR DISPLACEMENT, θ_2 IS IN-PLANE BENDING DISPLACEMENT.



EVALUATION OF JOINING CONCEPTS IN COMPOSITE STRUCTURES

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Material not received in time for inclusion in the publication.

PRELIMINARY ASSESSMENT OF COMPOSITES FOR
LONG RANGE ARTILLERY PROJECTILES

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Research Center/DRXMR-TM
Watertown, MA 02172

Material not received in time for inclusion in the publication.

PRELIMINARY DESIGN ASSESSMENT OF METAL MATRIX
BRIDGING STRUCTURES

E. Derby
Materials Science Corporation
Blue Bell, PA 19422

Material not received in time for inclusion in the publication.

FATIGUE DAMAGE MECHANISMS
IN METAL MATRIX COMPOSITE LAMINATES

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Civil Engineering Department
University of Utah
Salt Lake City, Utah 84112

and

William Steven Johnson

NASA, Langley Research Center
Hampton, Virginia 23665

The possible relationship between fatigue and shakedown in metal matrix composites was first suggested by Dvorak and Tarn¹ and related to then available experimental data, obtained primarily for unidirectional 6061 Al-B materials. In the present paper, the relationship is examined theoretically and experimentally for both unidirectional and laminated 6061-O Al-B composites.

One unidirectional and two laminated 6061-O Al-B composite plates were tested under various loading conditions in uniaxial tension. Three distinct types of material response to cyclic loading were identified: No evidence of damage at low loads; damage accumulation at moderate to high loads, caused primarily by growth of long fatigue cracks in the matrix parallel to the fibers within off-axis layers; and sudden, localized failure of the fibers at loads exceeding the endurance limit.

Quantitative analysis of the results shows that the onset of internal damage, demonstrated by a gradual reduction in axial elastic modulus with the increasing number of load cycles, depends on the applied stress range and is independent of the mean stress. The stress range at which damage first starts to appear coincides with the shakedown range, both in unidirectional and laminated plates. Since the aluminum matrix is strained elastically after the composite shakes down, it is concluded that the long fatigue cracks in the off-axis layers are caused by cyclic plastic straining of the matrix when the composite is loaded beyond the shakedown range. The magnitude of this range, which is relatively small in certain laminates, can be predicted from theoretical considerations and from measured cyclic strain response of the matrix material.

If the applied stress range causes fatigue damage but is smaller than that required for fatigue failure, there is evidence that

after about 1×10^6 cycles a certain saturation damage state is reached which depends on the applied stresses and remains essentially unchanged with increasing number of cycles. The existence of this damage state can be related to the growth of long matrix cracks in the off-axis layers. As these layers lose elastic stiffness due to fatigue cracking, the internal stresses in the laminate are transferred to the zero-degree layers. The unloading of the off-axis layers eventually leads to arrest of matrix crack growth, providing that the zero-degree layers can support the additional stress transferred from the off-axis plies. If the zero-degree layers become loaded at stress levels which exceed their endurance limit, as measured in tests on unidirectional specimens, the laminate fails.

It is possible to calculate the local stresses in individual plies during the damage evolution process, and arrive at predictions of the stiffness reduction of a laminate in the saturation damage state, or, alternatively, of the laminate endurance limit if the saturation state cannot be reached under a given applied stress.

Observed stiffness reductions in the saturation damage state ranged from about 8% in unidirectional specimens, to 50% in quasi-isotropic laminates. Fiber breaks prior to failure have been found only in specimens tested at loads approaching their endurance limits. These fiber breaks appear to be a part of the final failure process rather than of the mechanism leading to the saturation damage state.

ACKNOWLEDGEMENT

This work has been supported in part by the U.S. Army Research Office.

REFERENCES

1. G. J. Dvorak and J. Q. Tarn, "Fatigue and Shakedown in Metal Matrix Composites," Special Technical Publication 569, ASTM, 1975, pp. 145-168.
2. G. J. Dvorak and S. W. Johnson, "Fatigue of Metal Matrix Composites," International Journal of Fracture, to appear.

APPENDIX A
Listings of Unpresented Programs

MATERIALS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES

INHOUSE

ADVANCED COMPOSITES
WORK UNIT DIRECTIVE (WUD) NUMBER 45
77 April - 84 April

WUD Leader: J. M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autocon: 785-6685

Objective: The objective of the current thrust under this work is to develop and demonstrate concepts of damage resistance as applied to fiber reinforced composite laminates. Short term objectives (1-3 yrs) include the following:

- (a) Development of failure mode models with emphasis on delamination and matrix cracking.
- (b) Assess the role of matrix toughness in composite failure processes.
- (c) To develop concepts of interface/interphase strengthening.

CONTRACTS

IMPROVED MATERIALS FOR COMPOSITES AND ADHESIVES
F33615-78-C-5102
78 May 16 - 81 August 21

Project Engineer: J. M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autocon: 785-6685

Principal Investigator: R. L. Conner
University of Dayton Research Institute
300 College Park Avenue
Dayton, OH 45469
(513) 229-3016

Objective: Synthesis, formulation, processing, specimen fabrication, characterization, and evaluation of polymeric base and other non-metallic materials shall be performed to create, investigate and validate new concepts for aerospace vehicles.

FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES

IN-HOUSE

STRUCTURAL INTEGRITY RESEARCH FOR ENGINES AND AIRFRAMES

JON*: 2307N101

77 January 1 - 82 March 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBE
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Objective: To resolve theoretical questions and develop damage tolerance and life analysis methods which can be used to satisfy the requirements of MIL-STD-1530A for advanced composite and metallic airframe structures. The specific objectives in the composites area are:

- (a) assess the state of the art in fracture mechanics of composite materials;
- (b) develop procedures for analyzing composite materials static strength and fatigue life data;
- (c) assess the effect of fabrication variability and percentage of zero degree plies in a composite on the shape of the S-N curve and data scatter statistics;
- (d) demonstrate experimentally that a state of damage approach is viable for predicting the life of composites under block and random spectrum loading;
- (e) explore various nondestructive inspection methods for accurately documenting damage in resin matrix composites; and
- (f) demonstrate softening strip concepts for improving the damage tolerance of composite structures.

APPLICATION OF PHOTOMECHANICS TO EXPERIMENTAL STRUCTURAL ANALYSIS (STRUCTURAL INTEGRITY RESEARCH FOR ENGINES AND AIRFRAMES)

JON: 2307N101

1980 June 1 - 82 March 30

Project Engineer: Gene E. Maddux
Air Force Wright Aeronautical Laboratories
AFWAL/FIBE
Wright-Patterson AFB, Ohio 45433
(513) 255-5159 Autovon 785-5159

Objective: To develop and apply laser based metrology techniques (such as,

*JON is an internal Laboratory number assigned to the work unit.

holographic interferometry and speckle photography) to solve experimental stress analysis problems. Specific applications are being made in the following composite structures areas:

- (a) detection and characterization of failure modes;
- (b) verification of analytically predicted behavior of new test specimens for determining interlaminar tension allowables and shear properties; and
- (c) resonant frequency and mode shape determination for complex structures.

SONIC FATIGUE DESIGN OF ADVANCED STRUCTURES

JON: 24010146

79 November 11 - 82 November 12

Project Engineer: Howard F. Wolfe
Air Force Wright Aeronautical Laboratories
AFWAL/FIBE
Wright-Patterson AFB, Ohio 45433
(513) 255-5753 Autovon 785-5753

Objective: Develop test techniques and sonic fatigue design data for graphite/epoxy skin stringer composite structures.

TESTING OF CYLINDRICAL COMPOSITE PANELS

JON: 24010208

79 January 1 - 81 December 15

Project Engineer: Dr. N. S. Khot
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-5651 Autovon 785-5651

Objective: To investigate the buckling behavior of graphite/epoxy cylindrical panels subjected to compressive loading, and compare the test results with theoretical predictions. This study will give information on the effect of fiber orientation, layup and boundary conditions on the buckling and postbuckling behavior.

AEROELASTIC STUDY OF SWEPT WINGS WITH ANISOTROPIC BEHAVIOR

JON: 24010239

79 September 3 - 82 June 30

Project Engineer: Michael H. Shirk
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRC
Wright-Patterson AFB, Ohio 45433
(513) 255-6832 Autovon 785-6832

Objective: To provide analytical and experimental data on the aeroelastic behavior of wings constructed with advanced composite materials. The studies will include parameter variations, such as aspect ratio, sweep, and ply orientation. Experiments will include load deflection, influence coefficient measurement, ground vibration and wind tunnel testing.

ANALYSIS AND OPTIMIZATION OF AEROSPACE STRUCTURES

JON: 24010244

80 March 10 - 83 March 30

Project Engineer: Dr. V. B. Venkayya
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-4893 Autovon 785-4893

Objective: The key to the successful design of lighter and more reliable airframe structures is the ability to accurately predict structural response and to make rapid sensitivity analysis with parametric changes. The sensitivity analysis in turn is the important element in the evolution of dependable and cost effective structures. The objective of the effort is to develop computational tools for rapid analysis and optimization of metallic and composite aerospace structures.

STRUCTURAL TESTING OF COMPOSITE PANELS

JON: 24010246

80 April 28 - 83 June 30

Project Engineer: Lt Philip J. Conrad
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-4893 Autovon 785-4893

Objective: To develop experimental methods and to conduct tests to determine the buckling and postbuckling strength of stiffened and unstiffened composite panels.

PRELIMINARY DESIGN OF AIRCRAFT STRUCTURES

JON: 24010338

78 December 1 - 81 December 1

Project Engineer: Dr. R. S. Sandhu
Air Force Wright Aeronautical Laboratories
AFWAL/FIBC
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-4864

Objective: The overall objectives are (a) to explore structural deficiencies and related remedial measures, and (b) to develop and validate innovative design concepts. The specific objectives germane to mechanics of composites are:

(a) develop finite element methods for use in analysis and design of composite test specimens and structures;

(b) develop an optimum off-axis tension test specimen design;

(c) develop, analyze and validate a new specimen design for obtaining flatwise tension data for composite laminates; and

(d) assess the effects of jet fuel on the mechanical response of resin matrix composites.

COMPOSITE TEST METHODS (COMPRESSIVE TEST FIXTURE EVALUATION)

JON: 24010344

79 January 1 - 81 January 1

Project Engineer: Rick Rolfes
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCC
Wright-Patterson AFB, Ohio 45433
(513) 255-6658 Autovon 785-6658

Objective: To evaluate various compressive test fixtures currently in use by industry, together with an in-house prototype design. Efforts will focus on (a) elimination of the predominate brooming and buckling failure mode associated with present test fixtures, (b) 0° compressive strengths analogous to 0° tensile strengths, and (c) a reduction in costs of test specimen fabrication.

HYDRODYNAMIC RAM ASSESSMENT OF INTEGRAL SKIN/SPAR DESIGNS

JON: 24010349

80 March 24 - 81 January 1

Project Engineer: Dale E. Nelson
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Objective: To study the effect of hydrodynamic ram caused by ballistic penetration on advanced structures and to evaluate the relative susceptibility of several integral composite skin/spar concepts. This will provide designers with information necessary to allow transition of composite technology to operational aircraft.

ASSESSMENT OF CORROSION CONTROL PROTECTIVE COATINGS

JON: 24010350

80 April 28 - 85 May 1

Project Engineer: S. D. Thompson
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Objective: To determine the susceptibility of graphite/epoxy-aluminum joints to corrosion when protective coatings, that have undergone fatigue loading, are used. The knowledge gained will be used to determine if present corrosion control systems actually prevent corrosion and if not, how they could be modified to prevent corrosion from occurring.

GRANTS

CONDUCTION HEAT TRANSFER ANALYSIS IN COMPOSITE MATERIALS

Grant AFOSR 78-3640

JON: 2307N112

78 July 1 - 81 September 30

Project Engineer: Lt Sheryl K. Bryan
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-4893 Autovon 785-4893

Principal Investigator: Dr. L. S. Han
Ohio State University Research Foundation
1314 Kinnear Road
Columbus, Ohio 43212
(614) 422-6349

Objective: To investigate a class of heat conduction problems in fiber-matrix composite materials for which the proximity effects of the embedded fibers are significant and to strengthen the modelling approach by establishing bounds of accuracy through comparisons with exact data of analysis.

MEMORANDUM OF AGREEMENT

SPECTRUM LOAD/ENVIRONMENT INTERACTION EFFECTS IN ADVANCED FIBER REINFORCED LAMINATES

JON: 2307N106

76 October 1 - 81 September 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. Edward M. Wu, L-421
Lawrence Livermore Laboratory
Livermore, California 94550
(415) 422-6937

Objective: To develop (a) understanding of and methods for predicting creep and spectrum fatigue behavior of composites under various environmental conditions, and (b) procedures for accelerated durability testing.

CONTRACTS

TEST SYSTEM FOR CONDUCTING BIAXIAL TESTS OF COMPOSITE LAMINATES
Contract F33615-77-C-3014 JON: 2307N103
77 September 19 - 82 September 20

Project Engineer: T. N. Bernstein
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-4893 Autovon 785-4893

Principal Investigator: Dr. Isaac M. Daniel
IIT Research Institute
10 West 35th Street
Chicago, Illinois 60616
(312) 567-4000

Objective: To develop, design and fabricate a biaxial test machine capable of applying, without constraints, in-plane loads, singly and in any combination, to laminated tubular composite specimens.

EFFECTS OF VARIANCES AND MANUFACTURING TOLERANCES ON THE DESIGN STRENGTH AND LIFE OF MECHANICALLY FASTENED COMPOSITE JOINTS
Contract F33615-77-C-3140 JON: 24010110
78 February 15 - 81 April 15

Project Engineer: Capt. Robert L. Gallo
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Sam P. Garbo
McDonnell Company
P. O. Box 516
St. Louis, Missouri 63166
(314) 232-3356

Objective: To develop improved analytical methods and failure criteria which account for design variables and manufacturing anomalies in the prediction of failure load, mode, location, and fatigue life of bolted composite joints.

ADVANCED RESIDUAL STRENGTH DEGRADATION RATE MODELING FOR ADVANCED COMPOSITE STRUCTURES

Contract F33615-77-C-3084

JON: 24010117

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: D. E. Pettit
Lockheed-California Company
Rye Canyon Research Laboratory
Dept. 74-71, Bldg. 204, P/2
P. O. Box 551
Burbank, California
(213) 847-6121 ext. 131 291

Objective: To develop procedures and the required supporting data needed to predict (a) the growth of damage zones as a function of fatigue loading, (b) the residual strength as a function of the size and shape of the fatigue induced damage zones, (c) the Mechanisms of fatigue induced damage formation, and (d) the threshold levels of damage.

DESIGN SPECTRUM DEVELOPMENT AND GUIDELINES HANDBOOK FOR COMPOSITES

Contract F33615-78-C-3218

JON: 24010125

78 September 1 - 81 March 31

Project Engineer: John M. Potter
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. Robert Badaliane
McDonnell Douglas Corp.

P. O. Box 516
St. Louis, Missouri 63166
(314) 232-3356

Objective: To demonstrate existence of and quantify effects of load history variation on fatigue life of composite structures. This study will determine the load spectrum requirements for full scale durability testing of composite structures.

ENVIRONMENTAL TRACKING OF F-15 HORIZONTAL STABILATOR
Contract F33615-79-C-3210 JON: 24010132
79 June 15 - 83 October 1

Project Engineer: Carl L. Rupert
Air Force Wright Aeronautical Laboratories
AFWAL/FIBED
Wright-Patterson AFB, Ohio 45433
(513) 255-5753 Autovon 785-5753

Principal Investigator: Thomas V. Hinkle
McDonnell Douglas Corporation
P. O. Box 516
St. Louis, Missouri 63166
(314) 232-3356

Objective: To evaluate the effects of additional exposure to a service environment on the F-15 boron-epoxy stabilator.

ENHANCED X-RAY STEREOSCOPIC NDE OF COMPOSITE MATERIALS
Contract F33615-79-C-3220 JON: 24010133
78 September 18 - 80 November 15

Project Engineer: Dr. George P. Sendekyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Ward D. Rummel
Martin Marietta Aerospace
Denver Division
P. O. Box
Denver, Colorado 80201
(303) 977-4403

Objective: To explore the application of opaque penetrant enhanced, three-dimensional x-ray radiography to damage documentation in resin-matrix composite materials.

SPECIAL FASTENER DEVELOPMENT FOR COMPOSITE STRUCTURES
Contract F33615-80-C-3223 JON: 24010144
79 November 19 - 81 November 30

Project Engineer: Capt. Robert L. Gallo
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Robert T. Cole
Lockheed-Georgia Company
86 S. Cobb Drive
Marietta, Georgia 30063
(404) 424-3085

Objective: To develop fasteners that will improve the durability of bolted joints in composite structures.

DAMAGE PROGRESSION IN GRAPHITE-EPOXY BY A DEPLYING TECHNIQUE
Contract F33615-80-C-3224 JON: 24010148
79 November 17 - 81 September 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Sam Freeman
Lockheed-Georgia Company
86 S. Cobb Drive
Marietta, Georgia 30063
(404) 424-4730

Objective: To document the state of damage as a function of applied load in simple bolted joints in composites. Damage will be documented by using acoustic emission monitoring, penetrant enhanced x-ray radiography, and deplying.

FATIGUE/IMPACT STUDIES IN LAMINATED COMPOSITES
Contract JON: 24010152
80 May 12 - 83 December 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. Avva V. Sharma
Mechanical Engineering Department
North Carolina Agricultural & Technical State Univ.
Greensboro, North Carolina 27411
(919) 379-7620

Objective: To systematically document the fatigue induced damage
accumulation process in impact damaged structural laminates.

VALIDATION OF AEROELASTIC TAILORING BY STATIC AEROELASTIC AND FLUTTER TESTS
Contract F33615-77-C-3105 JON: 24010214
77 December 5 - 81 November 4

Project Engineer: Michael H. Shirk
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRC
Wright-Patterson AFB, Ohio 45431
(513) 255-6832 Autovon 785-6832

Principal Investigator: William Rogers
General Dynamics/Fort Worth Division
Fort Worth, Texas 76101
(817) 732-4811 ext 2320

Objective: To generate wind tunnel test data using static aeroelastic and flutter models to: (1) evaluate current analytical procedures used to predict aeroelastic tailoring benefits, (2) develop aeroelastic and flutter model scaling and fabrication techniques and (3) demonstrate performance benefits attainable through aeroelastic tailoring, e. g. reduced drag at maneuver conditions. A rigid model, three aeroelastic models, and two flutter models will be designed and tested. All models will be of the wing-body type, will utilize the same body of revolution and will also employ the AFTI-16 wing planform. The aeroelastic wings will have large design variations to provide the data needed to properly evaluate the aeroelastic tailoring design methods. The wings to be designed are: (1) a rigid undeformed wing designed to the jig shape, (2) an aeroelastically tailored wing, (3) an aeroelastically tailored wing designed to twist in the opposite direction of (2) and (4) a non-tailored wing. The two flutter wing designs will be representative of the two tailored aeroelastic models.

DESIGN METHODOLOGY FOR BONDED-BOLTED COMPOSITE JOINTS
Contract F33615-79-C-3212 JON: 24010228
79 August 1 - 81 July 1

Project Engineer: Lt. Philip Conrad
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-4893 Autovon 785-4893

Principal Investigator: L. J. Hart-Smith
McDonnell Douglas Corp.
Douglas Aircraft Company
3855 Lakewood Blvd.
Long Beach, California 90846
(213) 593-4079

Objective: To develop an efficient analysis procedure for predicting the structural behavior of bonded-bolted composite joints, identify the important variables that contribute to the strength of the joint, and develop a joint design methodology that can be coded into a joint design computer program.

DOD/NASA ADVANCED COMPOSITES DESIGN GUIDE
Contract F33615-78-C-3203 JON: 24010324
78 March 1 - 81 March 1

Project Engineer: Dale E. Nelson
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Principal Investigator: G. Howard Arvin
Rockwell International Corp.
LA Aircraft Division
5701 W. Imperial Highway
Los Angeles, California 90009
(213) 670-9151 ext 1666

Objective: To develop a new, updated version of the "Advanced Composites Design Guide." The New Version will incorporate new data and analysis techniques. The guide will be reorganized and condensed to make it a more useful document to designers.

INTEGRAL COMPOSITE SKIN/SPAR DESIGN STUDIES
Contract F33615-78-C-3209 JON: 24010328
78 September 1 - 81 December 1

Project Engineer: Dale E. Nelson
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Principal Investigator: Carlos Cacho-Negrete
Grumman Aerospace Corporation
Bethpage, L. I., New York 11714
(516) 575-2648

Objective: To obtain extensile design information on three advanced concepts for integral skin/spar construction. This information can then be used to incorporate these designs into future aircraft.

RESPONSE OF COMPOSITES TO FRAGMENT IMPACTS
JON: 24020230
78 October 1 - 82 October 30

Project Engineer: Lt. M. Kempster
Air Force Wright Aeronautical Laboratories
AFWAL/FIES
Wright-Patterson AFB, Ohio 45433
(513) 255-6302 Autocon 785-6302

Principal Investigator: W. Vikstad
Aberdeen Proving Grounds, Maryland

Objective: To develop empirical penetration equations for composite materials subjected to fragmenting missile impacts.

FATIGUE SPECTRUM SENSITIVITY STUDY FOR ADVANCED COMPOSITE MATERIALS
Contract F33615-75-C-5236 JON: 69CW0124
75 June 27 - 80 December 31

Project Engineer: Dr. Edvins Demuts
Air Force Wright Aeronautical Laboratories
AFWAL/FIBAC
Wright-Patterson AFB, Ohio 45433
(513) 255-6639 Autocon 785-6639

Principal Investigator: L. L. Jeans
Northrop Corp. 3853/82
3901 West Broadway
Hawthorne, California 90250
(213) 970-2134

Objective: To experimentally determine the sensitivity of the fatigue properties of advanced composite materials to the loading and environmental content of fighter aircraft fatigue spectra. To develop procedures and guidelines for deriving realistic accelerated/truncated fatigue spectrum simulations.

COMPOSITE WING/FUSELAGE PROGRAM

Contract F33615-79-C-3203

JON: 69CW0152

79 July 1 - 84 July 30

Project Engineer: Neal V. Loving
Air Force Wright Aeronautical Laboratories
AFWAL/FIBAC
Wright-Patterson AFB, Ohio 45433
(513) 255-6639 Autovon 785-6639

Principal Investigator: J. Eves, Program Manager
Northrop Corp., Aircraft Division
3901 West Broadway
Hawthorne, California 90250

Objective: To develop composites structural design technology and durability qualification methodology for advanced composite aircraft.

NAVAL AIR DEVELOPMENT CENTER

Point of Contact:

Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

NAVAL AIR DEVELOPMENT CENTER
WARMINSTER, PA 18974

INHOUSE

COMPOSITE IMPACT RESISTANCE
74 March - Continuing

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Objective: Ascertain the impact response of generic composite structural elements and identify the physical mechanisms associated with impact damage and the critical parameters governing impact response.

HYBRID COMPOSITE FRACTURE CHARACTERIZATION
80 September - 81 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Objective: Characterize the strength, mechanical properties, and failure characteristics of woven and intimately mixed hybrid composite laminates.

MOISTURE/IMPACT INTERACTION
79 January 1 - 81 September 30

Project Engineer: Mr. R. E. Trabocco
Naval Air Development Center
Warminster, PA 18974
(215) 441-2808 Autovon 441-2808

Objective: Examination of the effect of possible interaction of environmental factors, specifically moisture and low temperature, with low velocity impact loading on subsequent compressive buckling behavior and extent of damage.

DATA BASE-ORGANIC MATRIX COMPOSITES

79 January 1 - 81 September 30

Project Engineer: Mr. R. E. Trabocco
Naval Air Development Center
Warminster, PA 18974
(215) 441-2808 Autovon 441-2808

Objective: Establish a data base for promising emerging reinforced organic matrix composites based on natural and artificial exposure effects on physical and mechanical properties.

RT CURED PATCHES FOR C/EPOXY

79 October 1 - 81 September 30

Project Engineer: Mr. R. E. Trabocco
Naval Air Development Center
Warminster, PA 18974
(215) 441-2808 Autovon 441-2808

Objective: To develop polymer systems for repair that have higher service temperature capability than the cure temperature.

DEFECT SIGNIFICANCE IN COMPOSITE MATERIALS

76 November 1 - 80 September 30

Project Engineer: Dr. William R. Scott
Naval Air Development Center
ACSTD/6063
Warminster, PA 18974
(215) 441-3232 Autovon 441-2543

Objective: Characterize the nature and detectability of critical defects in composite materials through coordinated studies of mechanical properties and nondestructive testing. Recent studies have emphasized modeling of wave propagation in anisotropic laminates.

ULTRASONIC ATTENUATION MEASUREMENTS FOR NDT AND CHARACTERIZATION OF GRAPHITE/EPOXY LAMINATES

1 October 1980 - Continuing

Project Engineer: Dr. William R. Scott
Naval Air Development Center
ACSTD/6063
Warminster, PA 18974
(215) 441-3232 Autovon 441-3232

Objective: To investigate the relationship between frequency-dependent ultrasonic attenuation and material condition (e.g. % water absorption, void content, state of cure) for AS/3501-6 graphite/epoxy laminates. Further correlation with material properties such as strength and shear modulus is also anticipated.

CONTRACTS

DEVELOPMENT OF COMPRESSION FATIGUE LIFE PREDICTION METHODOLOGY AND DATA BASE FOR COMPOSITE STRUCTURES

N62269-79-C-0214

79 September 8 - 82 January 31

Project Engineer: Edward Kautz
Naval Air Development Center
ACSTD/60434
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: Dr. Robert Badaliance
McDonnell Douglas Corporation
Saint Louis, MO 63166
(314) 232-3356

Objective: To develop analytical compression fatigue life prediction methods and statistically sound experimental data based on fatigue testing specimens representing generic bolted composite joints.

DEFINITION AND MODELING OF CRITICAL FLAWS IN GRAPHITE FIBER REINFORCED RESIN MATRIX COMPOSITE MATERIALS

N62269-79-C-0209

77 October 1 - 80 July 30

Project Engineer: Dr. William R. Scott
Naval Air Development Center
ACSTD/6063
Warminster, PA 18974
(215) 441-3232 Autovon 441-2543

Principal Investigator: Dr. B. Walter Rosen
Materials Sciences Corporation
Blue Bell Office Campus
Merion Towle House
Blue Bell, PA 19422
(215) 542-8400

Objective: To pursue an iterative program of mathematical modeling, mechanical testing, and nondestructive testing in order to develop a methodology for detecting and assessing the criticality of defects in graphite fiber reinforced epoxy.

CONCEPTUAL DESIGN STUDY OF AIRCRAFT LANDING
GEAR UTILIZING ADVANCED MATERIAL SYSTEMS
N62269-80-C-0220
80 January 1 - 81 June 30

Project Engineer: Edward Kautz
Naval Air Development Center
ACSTD/60434
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: William B. Haynes
Grumman Aerospace Corporation
Bethpage, NY 11714
(516) 575-8525

Objective: To identify and evaluate the application of advanced materials to aircraft landing gear components which will yield substantial advantages in terms of reliability, maintainability, weight, and cost as compared to conventional metal designs. Organic and metal matrix composites, advanced metallic systems, and hybrid material systems will be considered.

FIBER REINFORCED ADVANCED TITANIUM LANDING GEAR STUDY
N62269-80-C-0284
80 October 1 - 81 September 30

Project Engineer: Edward Kautz
Naval Air Development Center
ACSTD/60434
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: V. E. Wilson
Rockwell International
P.O. Box 92098
Los Angeles, CA 90009
(213) 647-3531

Objective: To evaluate the potential for utilization of fiber reinforced advanced titanium in landing gear applications. A section of a typical landing gear outer cylinder will be designed, fabricated and tested.

ADVANCED COMPOSITE AFT FUSELAGE STUDY

N62269-78-C-0502

78 September 30 - 81 June 30

Project Engineer: T. E. Hess
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2041 Autovon 441-2041

Principal Investigator: E. Dhonau
Vought Corporation
Dallas, TX 75222
(214) 266-3639

Objective: The design and development of advanced composite fuselage types of construction which can satisfy the configuration and structural loading requirements of advanced aircraft systems and to identify, design, and fabricate critical subcomponents for structural test.

ADVANCED COMPOSITE CENTER FUSELAGE STUDY

N62269-78-C-0017

78 September 30 - 81 March 30

Project Engineer: T. E. Hess
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2041 Autovon 441-2041

Principal Investigator: N. Corvelli
Grumman Aerospace Corporation
Bethpage, NY 11714
(516) 575-2754

Objective: The design and development of advanced composite fuselage types of construction which can satisfy the configuration and structural loading requirements of advanced aircraft systems and to identify, design, and fabricate critical subcomponents for structural test.

THE INFLUENCE OF THERMAL EXPOSURE ON STRESSED AND
UNSTRESSED COATED GRAPHITE/EPOXY LAMINATES

N62269-79-C-0240

79 February 27 - 81 September 30

Project Engineer: Mr. Ronald E. Trabocco
Naval Air Development Center
ACSTD/60631
Warminster, PA 18974
(215) 441-2808 Autovon 441-2808

Principal Investigator: Mr. Robert Anderson
Hercules, Inc.
Magna, Utah 84044
(804) 250-5911, extension 2977

Objective: To determine the effect of one sided thermal exposures on stressed and unstressed, coated, thick (48 plies) graphite/epoxy laminates. Coating changes, NDI and residual static strength will be monitored for various times at temperature.

SPECTRUM FATIGUE TESTING OF GRAPHITE/EPOXY BOLTED JOINTS

N62269-77-C-0340

77 September 15 - 80 October 1

Project Engineer: Maurice S. Rosenfeld
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: J. F. Haskins
General Dynamics/CONVAIR
5001 Kearney Villa Road
San Diego, CA 92138
(714) 277-8900, extension 2088

Objective: To investigate the compression fatigue behavior of a mechanically fastened step joint between a graphite/epoxy laminate and an aluminum alloy fitting. The joint will be protected by the standard Navy corrosion control procedure and the tests will be performed for a flight-by-flight spectrum typical of Naval aircraft. The specimens will be exposed to various temperature and humidity conditions prior to and during the tests.

HIGH STRAIN COMPOSITE WING FOR PATROL
TYPE AIRCRAFT - CONCEPT VALIDATION
N62269-80-C-0129
80 September - 81 July

Project Engineer: Mark Libeskind
Naval Air Development Center
ACSTD/60434
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: J. Bruno
Grumman Aerospace Corp.
Bethpage, NY 11714
(516) 575-2648

Objective: Evaluate various high strain design concepts previously developed for patrol type aircraft through a progressive series of coupon and element tests. Maximum allowable design strain level for each concept shall be determined. Strain concentration around fastener holes, fatigue and environmental effects, damage tolerance and repairability for each concept will be determined.

HIGH STRAIN COMPOSITE WING FOR FIGHTER/ATTACK
TYPE AIRCRAFT - CONCEPT VALIDATION
N62269-80-C-0130
80 September - 81 July

Project Engineer: Mark Libeskind
Naval Air Development Center
ACSTD/60434
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: T. Hinkle
McDonnell Aircraft Co.
Box 516
St. Louis, MO 63166
(314) 232-3356

Objective: Evaluate various high strain design concepts previously developed for fighter/attack type aircraft through a progressive series of coupon and element tests. Maximum allowable design strain level for each concept shall be determined. Strain concentration to improve damage tolerance, survivability and repairability over current composite designs. Structural efficiency, manufacturing feasibility and quality assurance requirements will be determined.

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHMENTS FOR TAIL STRUCTURES

N62269-80-C-0286

80 September - 81 July

Project Engineer: Mark Libeskind
Naval Air Development Center
ACSTD/60434
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: B. Butler
Northrop Corporation
Aircraft Group
Hawthorne, CA 90250
(213) 970-3579

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft tail structures as an alternative to high-load transfer adhesive bonded titanium step joints. To improve damage tolerance, survivability and repairability over current composite designs. Structural efficiency, manufacturing feasibility and quality assurance requirements will be determined.

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHMENTS FOR WING STRUCTURES

N62269-80-C-0285

80 September - 81 July

Project Engineer: Mark Libeskind
Naval Air Development Center
ACSTD/60434
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: S. Garbo
McDonnell Aircraft Co.
P.O. Box 516
St. Louis, MO 63166
(314) 233-2016

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft wing structures as an alternative to high-load transfer adhesive bonded titanium step joints. Strain concentration around fastener holes, fatigue and environmental effects, damage tolerance and repairability for each concept will be determined.

COMPOSITE IMPACT DAMAGE SUSCEPTIBILITY

N62269-79-C-0274

79 September - 80 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. R. L. Ramkumar
Northrop Corporation
Aircraft Group
Hawthorne, CA 90250
(213) 970-5075

Objective: Conduct a combined theoretical/experimental program to identify and quantify impact damage mechanisms, to develop and verify design relationships capable of predicting incipient damage, and to provide design-useful procedures for estimating damage magnitude to composite laminate skins of monocoque wing-type structures subject to low velocity, hard object impact.

ENVIRONMENTAL EFFECTS ON COMPOSITE DAMAGE CRITICALITY

N62269-79-C-0259

79 September - 81 March

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. R. L. Ramkumar
Northrop Corporation
Aircraft Group
Hawthorne, CA 90250
(213) 970-5075

Objective: Experimentally investigate damage growth mechanisms and residual strength degradation of composite laminates as a result of impact-type damage, and to determine if there are additional performance degradations due to possible interaction between damage and environmental factors.

FRACTURE MECHANICS OF DELAMINATION INITIATION AND GROWTH

N62269-79-C-0270

79 September - 81 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. A. S. D. Wang
Drexel University
Philadelphia, PA 19104
(215) 895-2297

Objective: Delineate the intricacy in the interlaminar flaw behavior of composite laminates within the context of classical linear fracture mechanics. Four classes of problems, all involving delamination growth and failures, will be studied both experimentally and theoretically.

DAMAGE DEVELOPMENT MECHANISMS IN NOTCHED COMPOSITE LAMINATES

UNDER COMPRESSIVE FATIGUE LOADING

N62269-79-C-0261

79 September - 81 March

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. K. L. Reifsnider
Virginia Polytechnic Institute and
State University
Blacksburg, VA 24061
(703) 961-5316

Objective: Experimentally trace the sequence of damage development in graphite/epoxy composite laminates with center holes by cyclic compressive loading and to establish the influence of such damage on residual strength under uniform environmental conditions.

COMPOSITE SPECTRUM FATIGUE ANALYSIS

N62269-80-C-0265

80 August - 81 August

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. M. M. Ratwani
Northrop Corporation
Aircraft Group
Hawthorne, CA 90250
(213) 970-5285

Objective: Develop analytical techniques for predicting compressive fatigue life and residual strength for flight-by-flight spectrum loading.

POLYMER MATRIX FATIGUE PROPERTIES

N62269-80-C-0278

80 September - 82 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: David E. Walrath
University of Wyoming
Laramie, WY 82001
(307) 766-2177

Objective: Characterize and compare the fatigue properties of various matrix materials and correlate the resin fatigue properties to composite laminate fatigue behavior.

NAVAL AIR SYSTEMS COMMAND

INHOUSE

DETERIORATION OF LAMINATING RESINS

80 October 1 - 81 September 30

Project Engineer: Dr. J. Augl
Naval Surface Weapons Center
White Oak, Silver Spring, MD 20910
(204) 394-2262 Autovon 290-2261

Objective: To investigate the mechanisms of composite degradation under storage and service environments and to develop quantitative analytical procedures to predict such degradations and to verify these predictions experimentally.

HIGH PERFORMANCE COMPOSITES & ADHESIVES FOR V/STOL AIRCRAFT (DLF)

80 October 1 - 81 September 30

Project Engineer: Dr. Luther Lockhart
Naval Research Laboratory
Washington, DC 20375
(202) 767-2336

Objective: To provide guidance for selection of high performance adhesive and composite systems for application to V/STOL aircraft.

CONTRACTS

THERMOPLASTIC MATRIX COMPOSITES

80 October 1 - 81 September 30

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-7543 Autovon 222-7543

Principal Investigator: Mr. E. House
Boeing Aerospace Corporation
Seattle, WA 98124
(206) 655-3081

Objective: To evaluate the engineering properties of various thermoplastic resins and known reinforcements for aircraft structural application.

HOLES AND FASTENERS FOR ADVANCED COMPOSITES
80 October 25 - 81 July 30

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-7543 Autovon 222-7543

Principal Investigator: (to be designated)
McDonnell-Douglas Corp.
St. Louis, MO 63166
(314) 232-9501

Objective: To develop optimum hole preparation and fastener installation techniques.

COMPRESSION FATIGUE OF COMPOSITES
80 October 1 - 81 September 30

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-7543 Autovon 222-7543

Principal Investigators: Mr. G. Grimes
Northrop Corp.
3901 W. Broadway
Hawthorne, CA 90250
(213) 970-5075

Mr. Don Adams
University of Wyoming
Laramie, WY 82071
(307) 766-2371

Objective: To investigate the compression fatigue properties of graphite fiber epoxies, particularly under moist conditions.

METALLIC COATINGS FOR ADVANCED COMPOSITES
80 October 1 - 81 September 31

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-7543 Autovon 222-7543

Principal Investigator: Mr. C. Staebler
Grumman Aerospace Corp.
Bethpage, L.I., NY 11714
(516) 575-2244

Objective: To explore and evaluate the trade-offs associated with the use of metallic coatings on graphite epoxy composites.

AGING OF ORGANIC MATERIALS
80 October 1 - 81 September 31

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-7543 Autovon 222-7543

Principal Investigator: Mr. Z. Sanjana
Westinghouse Research & Development
Center
Pittsburgh, PA 15235
(412) 256-7218

Objective: To assess laboratory tests and a simple, low cost method for determining how long organic based materials (prepregs, adhesives, sealants) are suitable for use.

NAVAL AIR SYSTEMS COMMAND
WASHINGTON, D.C. 20361

SUMMARY OF PROGRAMS IN MECHANICS OF COMPOSITES

INHOUSE

FATIGUE OF COMPOSITES UNDER COMPLEX LOADS
79 October - Continuing

Project Engineer: Dr. P. W. Mast
Naval Research Laboratory
Washington, D.C. 20375
(202) 767-2165 Autovon 297-2165

Objective: Develop a capability for predicting the structural response and initiation of failure in composite laminates after complex cyclic loading.

CONTRACTS

DELAMINATION FAILURE CRITERIA FOR COMPOSITE STRUCTURES
80 August - 1981 August

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. R. Wilkins
General Dynamics
Fort Worth, Texas 76101
(817) 732-4811 Ext. 4631

Objective: Conduct experimental studies to develop a delamination failure criteria which can be combined with structural analysis to predict debonding in composite structures.

DELAMINATION IN COMPOSITE STEPPED LAP JOINTS
80 August - 1981 August

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. M. M. Ratwani
Northrop Corporation
Hawthorne, CA. 90250
(213) 970-5285

Objective: Formulate an analytical model to predict delamination in a laminated composite material bonded to a metallic adherend in a stepped lap joint configuration.

COMPRESSION FATIGUE FAILURE MECHANISMS

79 October - 1981 March

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. B. W. Rosen
Material Science Corporation
Blue Bell, PA. 19422
(215) 542-8400

Objective: Investigate critical failure mechanisms in notched composites under compression fatigue and establish an analytical methodology for prediction of failure initiation and damage propagation.

FATIGUE INDUCED DAMAGE IN COMPOSITE LAMINATES

78 October - 1981 October

Project Engineer: Dr. N. Perrone
Office of Naval Research
Washington, D.C. 22217
(202) 696-4307 Autovon 226-4307
and
Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. Z. Hashin
University of Pennsylvania
Philadelphia, PA. 19104

Objective: Develop analytical models to describe the response of composite laminates to cyclic loading, include problems of reduction of elastic moduli due to initiation of microcracking.

ACCEPTANCE CRITERIA FOR GRAPHITE/EPOXY STRUCTURES

79 July - 1981 July

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Mr. R. Riley
McDonnell Aircraft Company
St. Louis, MO. 63166
(314) 232-0232

Objective: Conduct an experimental investigation to determine the effects of various void levels in graphite/epoxy composites on structural response under combined compression and shear loading.

LOW VELOCITY IMPACT OF HYBRID COMPOSITE LAMINATES
1980 June - 1981 June

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. J. Renton
Vought Corporation
Dallas, Texas 75266
(214) 266-2436

Objective: Conduct an analytical and experimental investigation of the damage tolerance of graphite/Kevlar hybrid composites laminates subjected to low velocity impact.

NAVAL RESEARCH LABORATORY

Point of Contact:

Irvin Wolock
Mechanics of Materials Branch (6383)
Naval Research Laboratory
Washington, D. C. 20375

NAVAL RESEARCH LABORATORY

INHOUSE

FAILURE CRITERIA FOR COMPOSITES

70 July 1 - 81 September 30

Project Engineer: Dr. Phillip W. Mast (6383)
Mechanics of Materials Branch
Naval Research Laboratory
Washington, D. C. 20375
(202) 767-2165 Autovon 297-2165

Objective: To develop failure criteria for composites under static in-plane loading, to determine the effects of various environments, and to demonstrate the validity of these criteria in subcomponent tests.

FATIGUE OF COMPOSITES

79 October 1 - 81 September 30

Project Engineer: Dr. Phillip W. Mast (6383)
Mechanics of Materials Branch
Naval Research Laboratory
Washington, D. C. 20375
(202) 767-2165 Autovon 297-2165

Objective: To determine the effect of cyclic loading on the static structural response of composites measured under a broad range of static in-plane loadings.

ENGINEERING MODELING OF COMPOSITE MATERIALS

79 October 1 - 81 September 30

Project Engineer: Dr. Y. Rajapakse (6370)
Composite Materials Branch
Naval Research Laboratory
Washington, D. C. 20375
(202) 767-2264 Autovon 297-2264

Objective: To develop efficient methods for calculating moisture absorption in graphite/epoxy composites under transient conditions.

NASA LANGLEY RESEARCH CENTER

INHOUSE

FRACTURE OF LARGE LAMINATES
78 October 1 - 81 March 31

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Objective: To determine the effect of extensive damage on the tensile strengths of large composite laminate panels, and to correlate with fracture properties of small laminates made of taped and woven composites.

FATIGUE AND FRACTURE PROPERTIES AFTER MULTIPLE IMPACTS
78 October 1 - 81 June 30

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Objective: To determine the residual fatigue lives and strengths of coupons subjected to repeated scattered low-velocity impacts below visible-damage energies, and to determine the important impact parameters that correlate the data.

FRACTURE OF LAMINATED COUPONS
78 October 1 - 81 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3192 FTS 928-3192

Objective: To develop a methodology to predict residual strengths of damaged composite laminates using, as starting points, lamina properties or possibly the properties of the fibers and matrix. To determine the parameters that lead to tough composites.

DAMAGE TOLERANT COMPOSITE STRUCTURES

74 June 1 - 83 May 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3192 FTS 928-3192

Objective: To measure the ability of buffer strips and bonded stringers to increase the residual tension strength of damaged panels, and to develop an analysis to predict residual strength in terms of panel configuration and damage size.

FATIGUE OF BOLTED JOINTS

76 October 1 - 85 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2318 FTS 928-2318

Objective: To identify the bolted parameters that govern fatigue of joints, to incorporate these parameters in simple fatigue models, and to develop efficient life prediction procedures using these models.

FATIGUE OF BONDED JOINTS

76 October 1 - 85 September 30

Project Engineers: R. A. Everett, Jr.
Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Objective: To identify the parameters that govern fatigue of bonded joints, to relate these parameters to the failure mechanisms, and to develop life-prediction procedures.

RELAXATION OF BOLT CLAMPUP

80 January 1 - 81 December 31

Project Engineer: Dr. K. N. Shivakumar
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3178 FTS 928-3178

Objective: To develop a viscoelastic stress analysis to predict the relaxation of bolt clampup in composite joints.

PREDICTION OF FATIGUE LIFE OF NOTCHED COMPOSITE LAMINATES
73 June 1 - 81 September 30

Project Engineers: Dr. T. Kevin O'Brien
Dr. George L. Roderick
John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3191 FTS 928-3191

Objective: To develop a method to design fatigue resistant composite laminates. The method addresses three areas: failure mechanisms are identified; analyses to predict inplane and interlaminar damage growth are developed; and inplane and interlaminar data bases are developed to evaluate the methodology.

PREDICTION OF STIFFNESS LOSS, RESIDUAL STRENGTH, AND FATIGUE LIFE OF UNNOTCHED LAMINATES
80 June 1 - 81 October 31

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
U.S. Army R&T Laboratories (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3191 FTS 928-3191

Objective: To predict the stiffness loss, residual strength, and fatigue life of realistic unnotched laminates using baseline data from simple laminates.

THE EFFECTS OF REALISTIC FLIGHT ENVIRONMENTS ON FATIGUE OF COMPOSITE MATERIALS
72 June 1 - 83 May 31

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3191 FTS 928-3191

Objective: To determine the effects of realistic environments on the fatigue behavior of composite materials. Flight environments of conventional and supersonic aircraft transports and the Space Shuttle are being investigated. Tests are either accelerated or conducted in real time. Temperatures and load spectra are simulated for transport or Space Shuttle environments.

PREDICTION OF INSTABILITY-RELATED DELAMINATION GROWTH
79 January 2 - 83 December 31

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3191 FTS 928-3191

Objective: To predict rate of instability-related delamination growth. Approximate stress analyses will be developed based on understanding gained from rigorous analyses. Experiments will be performed to obtain a data base for use by the analysis in making predictions and for verifying and improving the analysis.

TENSION PROPERTY CHARACTERIZATION OF GRAPHITE/POLYIMIDE LAMINATES
79 June 1 - 80 November 1

Project Engineer: Andrew J. Chapman
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2869 FTS 928-2869

Objective: To experimentally determine the tension properties of C6000/PMR-15 and other graphite/polyimide materials in various laminate orientations at temperatures from 117 K (-250°F) to 589 K (600°F).

FLIGHT SERVICE EVALUATION OF COMPOSITE COMPONENTS ON COMMERCIAL AND MILITARY AIRCRAFT
72 March 1 - 90 December 31

Project Engineer: H. Benson Dexter
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2869 FTS 928-2869

Objective: To evaluate the long-term durability of composite components installed on commercial and military transport and helicopter aircraft. Over 200 components constructed of boron, graphite, and kevlar composites will be evaluated after extended service. Components include graphite/epoxy rudders, spoilers, tail rotors, vertical stabilizers, kevlar/epoxy fairings, doors and ramp skins, boron-reinforced aluminum center wing boxes and tail cone, and boron/aluminum aft pylon skins.

STIFFENED SANDWICH PANELS SHEAR PANELS
DODD, L - 3714 - 2848

Contracting Officer: L. J. Dodd
Mail Stop 145A
U.S. Army Research Center
Hampden, Virginia 23665
DOD 928-2848 ETS 928-2848

Abstract: This study will determine the shear strength of hat-stiffened, and blade-stiffened graphite polyimide shear panels. The shear tests will be performed at room temperature and 180 F (800 F).

STIFFENED SANDWICH PANELS SHEAR PANELS
DODD, L - 3714 - 2848

Contracting Officer: L. J. Dodd
Mail Stop 145A
U.S. Army Research Center (AVRADCOM)
Hampden, Virginia 23665
DOD 928-2848 ETS 928-2848

Abstract: This study will determine the static strength of kevlar and graphite epoxy composites under static shear and torsional loading. This study will establish a basis for demonstrating the use of this composite material beyond the point of initial shear failure.

STIFFENED SANDWICH PANELS CRASHWORTHY STRUCTURE
DODD, L - 3714 - 2848

Contracting Officer: L. J. Dodd
Mail Stop 145A
U.S. Army Research Center (AVRADCOM)
Hampden, Virginia 23665
DOD 928-2848 ETS 928-2848

Abstract: This study will determine the energy absorption of glass, kevlar, and graphite epoxy composite material and crashworthy composite structure subject to static and impact loading test conditions.

PRELIMINARY BOLTED JOINT DATA
78 July 1 - 81 June 30

Project Engineer: Gregory R. Wisch
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2818

Objective: To determine bolted joint strength and failure modes for advanced graphite/polyimide laminates from 110 K to 589 K, as well as the effect of joint geometry and temperature on joint strength and failure mode.

THE EVALUATION OF GRAPHITE/POLYIMIDE BOLTED JOINT PANELS
79 June 15 - 81 March 31

Project Engineer: Jane A. Hagaman
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2810

Objective: To evaluate the shear behavior of an optimized graphite/polyimide panel at room and elevated temperatures using a diagonal tension test method, and to correlate the behavior with analytical predictions.

MECHANICAL PROPERTY TEST METHODS FOR COMPOSITES
78 June 1 - 81 June 30

Project Engineer: Dr. Ronald K. Clark
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3386

Objective: To develop technology to support the establishment of industry standard test methods for reliable tension, compression, and inplane shear for advanced composite materials which will reduce the coefficient of variation of property data to less than 10 percent. Included will be schemes to test moisture-conditioned specimens at cryogenic and elevated temperatures with minimum time from conditioning to start of mechanical test.

EFFECT OF IMPACT ON MECHANICAL PROPERTIES OF GRAPHITE/EPOXY COMPOSITE MATERIAL

80 October 1 - 82 September 30

Project Engineer: Dr. Ronald K. Clark
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3386 FTS 928-3386

Objective: To determine the effect of impact on strength of Gr/Ep composite materials and to improve their resistance to impact loads.

ENVIRONMENTAL EFFECTS ON METAL MATRIX COMPOSITES

78 January 1 - 82 December 31

Project Engineer: W. D. Brewer
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2434 FTS 928-2434

Objective: To determine the effects of exposure to fabrication and various service environments on titanium and aluminum matrix composites, to identify the controlling mechanisms for material property changes, and to develop techniques and materials to control these changes to yield optimum composite properties for selected high-temperature aerospace applications.

RADIATION EFFECTS ON MATERIALS FOR STRUCTURAL COMPOSITES

79 July 1 - 84 June 30

Project Engineer: Dr. Edward R. Long, Jr.
Mail Stop 396
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3892 FTS 928-3892

Objective: To determine and correlate the effects of particulate radiation exposure on the properties and chemical structure of materials for structural composites and to develop procedures for accelerated laboratory simulation of long-term missions in a space radiation environment.

EFFECTS OF THERMAL CYCLING ON RESIDUAL PROPERTIES OF GRAPHITE/
POLYIMIDE COMPOSITES

77 October 1 - 81 September 30

Project Engineer: Dr. S. S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2434 FTS 928-2434

Objective: To determine the effects of cycling Gr/Pi from 117 K to 589 K on residual mechanical properties, at room temperature and 589 K, and on moisture sorption properties.

ALLOY AND THERMAL EXPOSURE EFFECTS ON METAL MATRIX COMPOSITES

77 January 1 - 80 December 31

Project Engineer: George C. Olsen
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2434 FTS 928-2434

Objective: To determine the matrix alloy constituent effects of B/Al and SiC/Al composites and to determine the long-duration thermal exposure effects on mechanical properties and microstructure.

DEVELOPMENT OF PRECISION ALIGNMENT FIXTURE FOR TENSILE TESTING

78 September 1 - 81 March 31

Project Engineer: Dr. Donald R. Rummeler
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2956 FTS 928-2956

Objective: To determine the effect of precision alignment on the mean and variance of the tensile strength of composite materials.

POSTBUCKLING AND CRIPPLING OF COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS

79 March 1 - 81 September 30

Project Engineer: Marshall Rouse
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2841 FTS 928-2841

Objective: To study the postbuckling and crippling of compression-loaded composite components and to determine the limitations of postbuckling design concepts to structural applications.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH CUTOUTS
77 October 1 - 81 September 30

Project Engineer: Dr. Martin M. Mikulas, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2551 FTS 928-2551

Objective: To study the effects of cutouts on the compression strength of composite structural components and to identify the failure modes that govern the behavior of compression-loaded components with cutouts.

DESIGN TECHNOLOGY FOR STIFFENED CURVED COMPOSITE PANELS
79 October 1 - 81 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2551 FTS 928-2551

Objective: To develop verified design technology for generic advanced-composite stiffened curved panels.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH LOW-VELOCITY IMPACT DAMAGE
76 October 1 - 81 September 30

Project Engineer: Marvin D. Rhodes
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3596 FTS 928-3596

Objective: To study the effects of low-velocity impact damage on the compression strength of composite structural components and to identify the failure modes that govern the behavior of compression-loaded components subjected to low-velocity impact damage.

DAMAGE TOLERANT DESIGN TECHNOLOGY FOR COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
78 October 1 - 81 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3524 FTS 928-3524

Objective: To develop structural design concepts for containing and resisting damage in compression-loaded composite structural components.

CONTRACTS

INCREMENTAL ANALYSIS OF IMPACT DAMAGE

NAS1-15888

79 August 3 - 81 January 31

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Principal Investigator: Edward A. Humphreys
Materials Sciences Corporation
Blue Bell Office Campus
Merion Towle House
Blue Bell, Pennsylvania 19422
(215) 542-8400

Objective: To develop an incremental damage analysis that predicts the extent of fiber breaks and matrix delaminations as a projectile transfers energy to a laminate in discrete steps. At each step, failure criteria determine the advance of damage and thus establish the configuration for the next increment of deformation.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES

NSG-1606

79 July 1 - 81 October 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3192 FTS 928-3192

Principal Investigator: Dr. Jonathan Awerbuch
Department of Mechanical Engineering
Drexel University
Philadelphia, Pennsylvania 19104
(215) 895-2291

Objective: To explore the fracture characteristics of graphite/polyimide composites at elevated temperatures using laminates with slits.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES

NSG-1297

74 October 16 - 81 October 15

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3192 FTS 928-3192

Principal Investigator: Dr. James G. Goree
Department of Mechanical Engineering
Clemson University
Clemson, South Carolina 29631
(803) 656-3291

Objective: To develop analyses that predict strength of buffer strip panels using models that treat the fiber and matrix as discrete elements.

EVALUATION OF THE DURABILITY AND DAMAGE TOLERANCE OF CTOL COMPOSITE STRUCTURES

NAS1-15107

77 October 12 - 80 October 11

Project Engineer: Edward P. Phillips
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Principal Investigator: Daniel J. Hoffman
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 241-3443

Objective: To perform selected analysis, fabrication, and testing tasks in the general area of durability and damage tolerance of graphite/epoxy composites, laminates, and structures. (To date, tasks have included a wing panel design study; a test program to determine the effect of built-in interlaminar defects on compression-compression fatigue life; and fabrication of damage tolerance test specimens--unstiffened panels containing glass or kevlar buffer strips, and stiffened panels without buffer strips.)

ENHANCED FRACTURE TOUGHNESS (OF GRAPHITE/EPOXY COMPOSITES)
NGR 23-005-528
78 November 1 - 81 June 30

Project Engineer: Dr. W. B. Fichter
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2093 FTS 928-2093

Principal Investigator: Dr. David K. Felbeck
Department of Mechanical Engineering
University of Michigan
Ann Arbor, MI 48109
(313) 764-3325

Objective: To increase the fracture toughness of typical multi-layered graphite/epoxy composite materials by examination and appropriate modification of the interlaminar load-transfer process in damaged laminates.

THREE-DIMENSIONAL STRESSES NEAR HOLES
NSG-1449
77 September 1 - 81 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2318 FTS 928-2318

Principal Investigator: Dr. I. S. Raju
Mail Stop 188E
Joint Institute for Advancement of Flight
Sciences
George Washington University at NASA
Langley Research Center
Hampton, Virginia 23665
(804) 827-3178 FTS 928-3178

Objective: To develop efficient three-dimensional finite-element methods that enable quantitative analysis of interlaminar stresses near laminate free edges.

DEVELOPMENT OF AN ORTHOTROPIC HOLE ELEMENT

NAS1-15890

79 July 9 - 80 September 24

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2318 FTS 928-2318

Principal Investigator: J. W. Markham
Lockheed-Georgia Company
86 South Cobb Drive
Marietta, Georgia 30063
(404) 424-3083

Objective: To develop a special finite element to represent the region around each fastener hole in a multi-fastener composite joint subjected to inplane axial and shear loads.

FRACTURE TESTING OF GRAPHITE/POLYIMIDE

NAS1-15080, Task 4; and NSG-1571

78 March 27 - 81 February 28

Project Engineer: R. A. Everett, Jr.
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U.S. Army R&T Laboratories (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Principal Investigator: Dr. Don H. Morris
Department of Engineering Science and
Mechanics
Virginia Polytechnic Institute and State
University
Blacksburg, Virginia 24061
(703) 961-5726

Objective: To establish the fracture characteristics of graphite/polyimide laminates containing round holes at cryogenic, room, and elevated temperatures.

BIAXIAL FATIGUE OF NOTCHED COMPOSITE LAMINATES

NSG-1289

79 January 1 - 80 December 31

Project Engineer: Dr. George L. Roderick
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NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3012 FTS 928-3012

Principal Investigators: Dr. D. L. Jones
Dr. H. L. Liebowitz
School of Engineering and Applied Science
George Washington University
Washington, DC 20052
(202) 676-6929

Objective: To determine the effects of biaxial fatigue loads on damage accumulation of notched composite laminates. Sheet material specimens are used in the experimental portion of the investigation.

EXPERIMENTAL STUDY OF FATIGUE DEGRADATION OF COMPRESSIVELY-LOADED COMPOSITE LAMINATES

NAS1-15956

79 September 15 - 81 March 15

Project Engineer: John D. Whitcomb
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Principal Investigator: Dr. R. L. Ramkumar
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Northrop Corporation
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3901 West Broadway
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(213) 970-5075

Objective: To identify the dominant mechanisms of fatigue degradation in compressively-loaded composite laminates.

A STUDY OF STIFFNESS, RESIDUAL STRENGTH, AND FATIGUE LIFE RELATIONSHIPS FOR COMPOSITE LAMINATES

NAS1-16406

80 October 1 - 81 September 30

Project Engineer: Dr. T. Kevin O'Brien
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Principal Investigators: Dr. James T. Ryder
Lockheed-California Company
Burbank, California 91520
Dr. Frank W. Crossman
Lockheed Research Laboratory
Palo Alto, California 94304

Objective: To develop quantitative relationships between laminate stiffness, residual strength, and fatigue life for unnotched laminates.

THERMOMECHANICAL RESPONSE OF GR/PI COMPOSITES

NAS1-15841

79 October 1 - 80 December 31

Project Engineer: Dr. John G. Davis, Jr.
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Hampton, Virginia 23665
(804) 827-2125 FTS 928-2125

Principal Investigator: E. A. Derby
Material Sciences Corporation
Blue Bell Office Campus
Merion Towle House
Blue Bell, Pennsylvania 19422
(215) 542-8400

Objective: To develop the capability to predict the response of graphite/polyimide composites under thermal and mechanical loading. Nonlinear and viscoelastic matrix behavior will be incorporated in laminate stress analysis.

DESIGN ALLOWABLES CHARACTERIZATION OF C6000/LARC-160 GRAPHITE/
POLYIMIDE

NAS1-15183

80 October 1 - 82 January 1

Project Engineer: Andrew J. Chapman
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Hampton, Virginia 23665
(804) 827-2869 FTS 928-2869

Principal Investigator: H. Q. Norris
Rockwell International Corporation
Space Division
Seal Beach, California 90740
(231) 594-3289

Objective: Experimentally determine mechanical properties of
graphite/polyimide laminates for use in designing
aerospace structures for service at temperatures
from 117 K (-250°F) to 589 K (600°F).

DESIGN ALLOWABLES CHARACTERIZATION OF C6000/PMR-15 GRAPHITE/
POLYIMIDE

NAS1-15644

80 October 1 - 81 September 1

Project Engineer: Andrew J. Chapman
Mail Stop 188A
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Hampton, Virginia 23665
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Principal Investigator: D. E. Skoumal
Boeing Aerospace Company
P. O. Box 3999
Seattle, Washington 98124
(206) 773-8016

Objective: Experimentally determine mechanical properties of
graphite/polyimide laminates for use in designing
aerospace structures for service at temperatures
from 117 K (-250°F) to 589 K (600°F).

TIME-TEMPERATURE-STRESS CAPABILITIES OF COMPOSITE MATERIALS FOR
ADVANCED SUPERSONIC TECHNOLOGY APPLICATIONS

NAS1-12308

73 June 1 - 84 September 30

Project Engineer: Bland A. Stein
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NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3354 FTS 928-3354

Principal Investigator: J. F. Haskins
Mail Zone 43-6320
General Dynamics
P. O. Box 80847
San Diego, California 92138
(714) 891-8900, ext. 2088

Objective: To establish the time-temperature-stress characteristics and capabilities of five classes of high temperature composite materials (Gr/Ep, B/Ep, K/Ep, B/Pi, and B/Al) subjected to a simulated supersonic cruise flight environment for up to 50,000 hours.

RADIATION EXPOSURE OF COMPOSITE MATERIALS

NAS1-15606

79 February 27 - 81 January 31

Project Engineer: Wayne S. Slep
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Hampton, Virginia 23665
(804) 827-3041 FTS 928-3041

Principal Investigator: Lawrence B. Fogdall
Boeing Aerospace company
P. O. Box 3999
Seattle, Washington 98124
(206) 773-6711

Objective: To determine the effects of simulated space radiation on the mechanical and chemical properties of composite materials. This study will provide data for establishing the long-term space durability of current composites. Particular attention will be directed toward combined proton and electron effects to determine whether synergistic interactions occur.

GRAPHITE FIBER REINFORCED GLASS MATRIX COMPOSITES

NAS1-14346

76 March 2 - 81 March 16

Project Engineer: Dennis L. Dicus
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665

Principal Investigator: Dr. Earl M. Prew
United Technologies Research Center
East Hartford, Connecticut 06108
(203) 727-7237

Objective: To develop a composite material with useful structural properties at temperatures up to 800 K. The resistance of this material to oxidation, long-term high temperature exposure, thermal cycling, and space radiation is being studied.

ENVIRONMENTAL EXPOSURE EFFECTS ON COMPOSITE MATERIALS FOR COMMERCIAL AIRCRAFT

NAS1-15148

77 November 1 - 88 November 30

Project Engineer: Dr. Ronald K. Clark
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3386 FTS 928-3386

Principal Investigator: Daniel J. Hoffman
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 241 3443

Objective: To provide technology in the areas of characterization methods and environmental effects on Gr/Ep composite materials, including development of accelerated test and analysis methods to predict long-term performance of advanced resin-matrix composite materials within 20 percent of real-time aircraft service exposure results.

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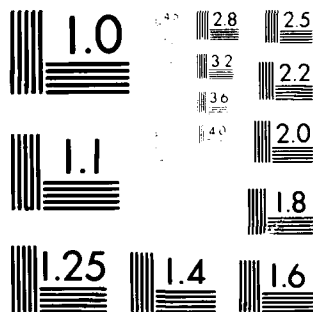
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FIBER-REINFORCED TITANIUM MATERIALS AND PROCESS INTERACTIONS

NAS1-16403

80 September 15 - 81 December 31

Project Engineer: W. D. Brewer
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Principal Investigator: Gordon S. Doble
T/M 2127
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Materials Technology
2355 Euclid Avenue
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(216) 383-2127

Objective: To develop, through innovative materials systems and processing techniques, fiber-reinforced titanium composites that are stable after high-temperature processing and that have sufficiently good properties for long-term service in high-performance aircraft structures applications.

EFFECTS OF HIGH-ENERGY RADIATION ON THE MECHANICAL PROPERTIES OF GRAPHITE FIBER REINFORCED EPOXY RESINS

NSG-1562

79 October 1 - 81 December 31

Project Engineer: Dr. Edward R. Long, Jr.
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Hampton, Virginia 23665
(804) 827-3892 FTS 928-3892

Principal Investigators: Dr. Jasper D. Memory
Dr. Raymond E. Fornes
Departments of Physics and Textiles
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(919) 737-2503/737-3231

Objective: To investigate the effects of high-energy radiation on graphite fiber composites by study of composite curing effects, radiation exposure rates, mechanical fracture surfaces, and electron spin resonance properties.

LSST HOOP/COLUMN ANTENNA: MATERIALS TASK

NAS1-15763

79 April 1 - 83 March 31

Project Engineer: George C. Olsen
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(804) 827-2434 FTS 928-2434

Principal Investigator: Marvin Sullivan
Harris Corporation
P. O. Box 37
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(305) 727-5813

Objective: To develop tension stabilizing cables with a high degree of dimensional stability for use on a 100-m diameter space deployable antenna; to develop lightweight, thermally stable composite materials for structural members and joints.

ENVIRONMENTAL EFFECTS ON ADVANCED COMPOSITES

NCCI-10

79 August 1 - 80 November 30

Project Engineer: Dr. Darrel R. Tenney
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(804) 827-2434 FTS 928-2434

Principal Investigators: Dr. Jalaiah Unnam
Dr. Charles R. Houske
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-5652

Objective: To develop silicon carbide fiber reinforced titanium matrix composites with rule of mixture properties for elevated temperature applications.

STRUCTURAL TEST SPECIMENS USING FIBER-REINFORCED COMPOSITE MATERIALS

NAS1-12675

73 September 6 - 81 January 6

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
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(804) 827-3524 FTS 928-3524

Principal Investigator: Cliff Kam
Douglas Aircraft Company
3855 Lakewood Boulevard
Long Beach, California 90846
(213) 593-5332

Objective: To design, fabricate, and test composite compression components for structural applications; to develop fabrication procedures for stiffened panels; and to evaluate damage tolerant materials.

EVALUATION OF THE DURABILITY AND DAMAGE TOLERANCE OF COMPOSITE STRUCTURES SUITABLE FOR COMMERCIAL TRANSPORT AIRCRAFT

NAS1-15107

77 October 1 - 81 September 30

Project Engineer: Marvin D. Rhodes
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(804) 827-3596 FTS 928-3596

Principal Investigator: John E. McCarty
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 433-1430

Objective: To design, fabricate, and test generic composite structural components for commercial aircraft applications that are durable and damage tolerant.

LOW-SPEED IMPACT DAMAGE ON COMPOSITE MATERIALS
NSG-1483

78 January 15 - 81 January 15

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
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Hampton, Virginia 23665
(804) 827-2551 FTS 928-2551

Principal Investigators: Dr. Wolfgang G. Knauss
Dr. Charles D. Babcock
California Institute of Technology
Pasadena, California 91125
(213) 795-6811, ext. 1524/1528

Objective: To study the effects of low-speed impact damage in composite structural components using high-speed motion pictures and to develop an analytical procedure for the propagation of the resulting impact damage.

ADVANCED COMPOSITE STRUCTURAL DESIGN TECHNOLOGY FOR COMMERCIAL
TRANSPORT AIRCRAFT
NAS1-15949

79 September 24 - 82 September 24

Project Engineer: Dr. James H. Starnes, Jr.
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(804) 827-2551 FTS 928-2551

Principal Investigator: John N. Dickson
Lockheed-Georgia Company
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(404) 424-3085

Objective: To design, analyze, fabricate, and test generic advanced-composite structural components for transport aircraft applications in order to develop verified design technology.

AN INVESTIGATION OF AN IMPROVED THEORY FOR THE ANALYSIS OF NATURAL
AND MANMADE LAYERED STRUCTURES

NSG-1401

79 September 16 - 81 March 15

Project Engineer: Dr. Manuel Stein
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Hampton, Virginia 23665
(804) 827-2813 FTS 928-2813

Principal Investigator: Dr. Paul Seide
University of Southern California
Los Angeles, California 90007
(213) 741-2948

Objective: An improved finite element has been developed which
will enable complex problems to be modeled much more
efficiently and accurately than previously. Explora-
tion of the capabilities of the new finite element
code is being conducted.

NAVAL RESEARCH LABORATORY

INHOUSE

FAILURE CRITERIA FOR COMPOSITES

70 July 1 - 81 September 30

Project Engineer: Dr. Phillip W. Mast (6383)
Mechanics of Materials Branch
Naval Research Laboratory
Washington, D. C. 20375
(202) 767-2165 Autovon 297-2165

Objective: To develop failure criteria for composites under static in-plane loading, to determine the effects of various environments, and to demonstrate the validity of these criteria in subcomponent tests.

FATIGUE OF COMPOSITES

79 October 1 - 81 September 30

Project Engineer: Dr. Phillip W. Mast (6383)
Mechanics of Materials Branch
Naval Research Laboratory
Washington, D. C. 20375
(202) 767-2165 Autovon 297-2165

Objective: To determine the effect of cyclic loading on the static structural response of composites measured under a broad range of static in-plane loadings.

ENGINEERING MODELING OF COMPOSITE MATERIALS

79 October 1 - 81 September 30

Project Engineer: Dr. Y. Rajapakse (6370)
Composite Materials Branch
Naval Research Laboratory
Washington, D. C. 20375
(202) 767-2264 Autovon 297-2264

Objective: To develop efficient methods for calculating moisture absorption in graphite/epoxy composites under transient conditions.

APPENDIX B
Attendance List

MECHANICS OF COMPOSITES REVIEW
Bergamo Conference Center
Dayton, Ohio
28-30 October 1980

List of Attendees

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