Literature Reviews

Composite Materials; Mechanics, Mechanical Properties and Fabrication

Proceedings reviewed by William W. Feng, Lawrence Livermore National Laboratory, Livermore, CA 94550.


This is a collection of papers presented at the 1981 Japan-U.S. Conference on Composite Materials; Mechanics, Mechanical Properties and Fabrication, held at Tokyo and sponsored by the Japan Society for Composite Materials.

Sixty-one papers are collected in this book: 38 were written by Japanese authors, 19 by American authors, and 4 by Japanese and American authors jointly. These papers may be grouped into the following ten categories:

1. Dynamic Behavior and Wave Propagation—Impact resistance, wave propagation in composite materials, and transient wave propagation in viscoelastic laminate composites are presented.
2. Stress Analysis and Mechanical Properties—Both photoelasticity and finite-element methods are used to determine the stress state of woven glass-fiber-reinforced plastic laminates and the local and edge effect of the composite materials under loads. The material properties and elastic, thermal, and compressive creep compliance for various composite materials are presented. Two theoretical papers, based on Eshelby’s equivalent inclusion method, develop the constitutive equations of composite materials and predict the first and second stages of the stress-strain curve of unidirectional short-fiber-reinforced thermoplastics.
3. Fatigue Properties—The mechanics of degradation during cyclic loading of composite laminates, fatigue properties of satin woven glass cloth fiber reinforced plastics under multistep loading conditions, and an analytical approach to predict the accumulation of damage in laminated composites are presented.
4. Fracture and Strength of Composites—The effect of inhomogeneity of composite materials, such as fiber diameter and water content, on the elasto-plastic fracture toughness is presented. The nonlinear fracture analysis of viscoelastic composite materials and a probabilistic approach to the strength of fibrous composites are also given.
5. Metal-Matrix Composites—The following topics are covered in this category:
   - Theory and experiments on tensile strength of fibers coated with a brittle layer.
   - The role of work hardening in the mechanical behavior of metal fiber-metal matrix composites.
   - Some properties of composite metals reinforced with helical fibers.
   - Microstructural analysis of SiCw/Al composites.
   - Compatibility of SiC fibers with aluminum.
   - Continuous-SiC-fiber-reinforced aluminum.
   - Inorganic alumina fibers for reinforcement of metal castings.
   - Effect of thermal cycling on the interface of fiber-reinforced metallic composites.
   - Wear properties of graphite-fiber-reinforced metals.
   - High-temperature strength of Mo-TiC eutectic composite.
   - Reinforcing effect of newly developed steel fiber in the steel fiber mortar.

6. Ceramics and Rubber Composites—This section describes an internal-friction study of elevated-temperature properties of sintered silicon nitrides and the elastic and viscoelastic behavior of carbon-reinforced rubber vulcanizates under large deformation.
7. Thermal and Environmental Effects—Moisture diffusivity, hygrothermally induced boundary layer effects, heat generation and conduction during crack propagation, and surface radiation damage of composites are presented.
8. Strength of Structural Elements—Strengths of reinforced panels with hat-shaped composite stiffener, woven-fabric reinforced plastic, internal pressurized carbon-fiber helical-wound cylinders, and buckling of geodesic cylinders are reported.
9. Design and Applications—The design, analysis, manufacture, and testing of some applications of composite materials to the structural components used in satellites, aircraft, and trucks are presented.
10. Overview of Composites—The state of technology in composite materials and an overview of hybrid composite applications are reported briefly here.

This collection of papers covers all aspects of research areas in composites. It is not a textbook that will be on the shelf of every private library; however, one would benefit from reading a few articles related to one’s field.

Ultrasonic and Mechanical Characterizations of Fatigue States of Graphite Epoxy Composite Laminates

Report reviewed by K. L. Reifsnider, Virginia Polytechnic Institute and State University, Blacksburg, VA 24061.


This report is an extension of earlier work by the principal author and his associates. In the words of the author, this report attempts to "explore the relationship between ultrasonic attenuation and fatigue survivability of graphite fiber epoxy composites fabricated under various processing temperatures and pressures." The principal thesis of the work is that changes in ultrasonic through-transmission attenuation can be correlated with changes in response of composite laminates during flexural fatigue loading, and ultimately to the life of such specimens. An ancillary claim, supported by the earlier work, is that this attenuation in as-fabricated lami-
nates subjected to flexural cyclic fatigue can be correlated with the number of fatigue cycles to failure.

Two aspects of this work make it especially interesting and important. First, it is one of relatively few investigations that deal with attenuation during cyclic loadings. Second, it deals with “fiber-dominated” laminate configurations, specifically with \([0, \pm 45, 0]\) graphite/epoxy (AS/3501-6) laminates. Fabrication details of the specimens tested are summarized in Table 1. While flexural fatigue may not be of primary interest to some readers, it does at least provide a controlled damage development situation for study. The data reported were collected from specimens subjected to cyclic reversed flexural loading of about 0.67 of the static ultimate failure load of the specimens for that type of loading. Failure was defined when the maximum deflection of the specimen reached a predetermined number corresponding to a large stiffness change, usually about 70%.

The ultrasonic attenuation arrangement was a pitch-catch configuration with broadband transmitting and receiving transducers used to measure the changes in the through-the-thickness attenuation as a function of fatigue cycling. Attenuation measurements were made at 4.0 MHz before fatigue testing and intermittently during each test.

Two examples of the results obtained are shown in Figs. 1 and 2. Figure 1 shows the degradation of flexural stiffness as a function of the number of applied cycles of loading. For the log-linear plot presented, the stiffness of the specimen (and therefore its condition) degrades at an ever-increasing rate, and becomes precipitous near the end of the specimen life. The consistency of the data and the magnitude of stiffness changes observed are especially noteworthy. Figure 2 shows the corresponding relative attenuation variations for those specimens. One can see that there is a corresponding increase in the relative attenuation which begins at approximately the same point at which the flexural stiffness began to drop sharply in Fig. 1. One can also see that the fabrication details appear to influence the initial values of attenuation more than the comparative values or changes during fatigue cycling. The authors present a detailed discussion of these relationships and note that specimens with relatively low void volume fraction showed the greatest changes in both flexural stiffness and ultrasonic attenuation.

Table 1—Summary of effects of cure pressure and temperature on void volume fraction, mechanical properties, and ultrasonic attenuation on AS/3501-6 laminates. (Table 1 of Williams et al.)

<table>
<thead>
<tr>
<th>Cure Pressure, MPa</th>
<th>Cure Temperature, °C</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>150</td>
</tr>
<tr>
<td>SPECIMEN DESIGNATION</td>
<td>#1</td>
</tr>
<tr>
<td>0.52</td>
<td>V_v = 2.70%</td>
</tr>
<tr>
<td></td>
<td>(\Delta \sigma^*) = 0.45 neper/cm</td>
</tr>
<tr>
<td></td>
<td>k = 19.60 N/cm</td>
</tr>
<tr>
<td></td>
<td>N = 990 000</td>
</tr>
<tr>
<td></td>
<td>t = 0.108 cm</td>
</tr>
<tr>
<td>0.69</td>
<td>V_v = 2.89%</td>
</tr>
<tr>
<td></td>
<td>(\Delta \sigma^*) = -0.60 neper/cm</td>
</tr>
<tr>
<td></td>
<td>k = 23.70 N/cm</td>
</tr>
<tr>
<td></td>
<td>N = 662 000</td>
</tr>
<tr>
<td></td>
<td>t = 0.102 cm</td>
</tr>
<tr>
<td>0.86</td>
<td>V_v = 2.19%</td>
</tr>
<tr>
<td></td>
<td>(\Delta \sigma^*) = 0.70 neper/cm</td>
</tr>
<tr>
<td></td>
<td>k = 21.10 N/cm</td>
</tr>
<tr>
<td></td>
<td>N = 562 000</td>
</tr>
<tr>
<td></td>
<td>t = 0.090 cm</td>
</tr>
</tbody>
</table>

**Note:** V_v = void volume fraction, \(\Delta \sigma^*\) = relative initial attenuation at 4 MHz, k = initial stiffness, N = number of fatigue cycles to fatigue failure, and t = specimen thickness.
volume fraction had relatively low attenuation at 4.0 MHz. The authors also conclude that the ultrasonic attenuation of the as-fabricated composite specimens is a potential indicator of fatigue life for the situations that they investigated.

This work is useful and constructive, and leaves one curious about the microdamage events that caused the observed changes in attenuation. Continued work in this area certainly appears desirable.

Post-Buckling Fatigue Behavior of Flat, Stiffened Graphite/Epoxy Panels Under Shear Loading

Report reviewed by K. L. Reifsneider, Virginia Polytechnic Institute and State University, Blacksburg, VA 24061.


This report describes an exploratory research project that addresses the performance of integrally stiffened laminated composite shear panel structures of the type that might be used for lightly loaded fuselage assemblies when loaded at quasi-static and cyclic shear flow levels that correspond to post-buckled deformations. The authors make the point that while state-of-the-art metallic fuselage structures are designed to operate well into the post-buckled range, and many years of flight service experience have proven their integrity, there has been a reluctance to design corresponding composite structures that buckle below the limit load. They claim that the reasons for this reluctance include the lack of analytical and experimental data describing post-buckle behavior of laminated anisotropic plates, uncertainties concerning failure modes under repeated loading in the post-buckled range, and, in general, the lack of flight service experience with composite structures of this type. However, it is claimed that very little weight savings can be realized from the use of composite laminates in fuselage structures unless they are permitted to buckle well below the design limit load.

The authors' objective was "to investigate the fatigue behavior of composite shear panels cycled well into the post-buckled range, to provide preliminary design information and a worst-case assessment of problems, and to assess the accuracy of state-of-the-art analysis methods."

Fifteen T300/5208 graphite/epoxy shear panels stiffened with cocured J-section stiffeners (Fig. 3) were fabricated and tested. Three panels were tested at static ultimate failure to assess the accuracy of design and analysis methods. Nine panels were fatigue-tested at three different load levels in the post-buckling range, and residual strength determined. Several panels were also manufactured with simulated skin-stiffener debonding regions and tested to determine the effects of manufacturing or service-induced debonding and subsequent repair on post-buckling fatigue behavior. Shear testing was done with a standard "picture frame" shear loading fixture.

Design criteria for the shear panel were (1) design limit shear flow of 52.5 to 70 kN/m, (2) initial buckling onset at or slightly above 30% of limit shear flow, and (3) withstand design ultimate shear flow (1.5 times limit shear flow).

Preliminary studies by Lockheed indicated that fuselage panels designed for construction with laminated composite materials could carry design shear flows for typical aft fuselage panels in the 35- to 87.5-kN/m range with about 50% weight savings compared to aluminum structures sized to the same loads, if composite panels were allowed to buckle up to a buckling ratio of about 5.2 (a common practice for comparable aluminum structures). Hence, the following design loads were used to size the test panel: 16.625 kN/m for level flight, 175-kN/m initial buckling level, 58.275-kN/m design limit shear flow (3.5 g), and 87.5-kN/m design ultimate shear flow (5.25 g). These design loads result in an ultimate buckling ratio of 5.0.

Overall panel buckling, local panel buckling between stiffeners, and test panel failure were predicted using various finite-element and closed-form analyses. Three potential failure modes were considered: material failure in the skin, stiffener crippling, and debonding of the stiffener from the skin. Skin tensile strain failure was...
calculated using an engineering approach which assumed that the shear panel acted as an incomplete tension field beam with a buckle pattern oriented at an angle of 45° to the stiffeners. Additional shear load beyond the critical level for local buckling \( q_{cr} \) was assumed to be carried as tension load equal to magnitude to \( 2(q - q_{cr}) \) while the compressive loading remained at \( q_{cr} \).

Stiffener crippling levels were calculated from compressive forces acting on the stiffeners assisted by a sinusoidal force distribution along the stiffener caused by the local buckling of the skin. Stiffener stresses were calculated by summing the compressive and bending stresses. Skin-stiffener debonding was predicted by calculating the peeling load per unit length along the interface based on an estimation of the normal component of the tensile load acting in the skin determined from the previously calculated buckle deflection and the assumed buckle geometry.

From these calculations, and from material strength data obtained from earlier testing, it was determined that skin tensile failure corresponding to a minimum allowable strain of 5900 µε occurred at an applied shear load of 146.5 kN/m. These calculations also indicated that stiffener crippling was not a critical failure mode under any conceivable circumstance. J-stiffener pull-off tests were conducted on specimens cut from skin-stiffener elements. A failure load of about 8.75 kN/m was obtained. Hence, this failure mode was expected to be active at a shear flow level of about 116.4 kN/m and became the critical failure mode for the test panel under static shear.

Figure 4 shows a comparison of predicted and observed failure loads for each failure mode under quasi-static loading. Skin-stiffener debonding initiated near the ends of the stiffeners adjacent to the loaded corners of the rig before reaching the predicted load for failure corresponding to a minimum allowable strain of 5900 µε (~approximately 1.25 times design limit load), and an intermediate load. Skinstiffener debonding was predicted by calculating the peeling load per unit length along the interface based on an estimation of the normal component of the tensile load acting in the skin determined from the previously calculated buckle deflection and the assumed buckle geometry.

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Cyclic loading was applied at a rate of 2 Hz at levels of 56 kN/m (corresponding to the approximate design limit load), 77.35 kN/m (approximately 1.25 times design limit load), and an intermediate level of 64.75 kN/m for a total of 100,000 cycles or until failure of the panel. Neither initiation of skin-stiffener debonding nor panel failure occurred in 100,000 cycles of loading at the lowest shear flow level. Residual strength tests conducted on specimens that did not fail showed little or no effect before load cycling, suggesting that failure of the panels during cyclic loading occurred immediately upon the initiation of structural damage of the type considered.

The simulated-damage panels were fabricated by inserting a slightly tapered stainless steel shim approximately 5 cm long extending across the full width of the stiffener flange between the stiffener and the panel during fabrication, then removing it after the panel was cured. Mechanical fasteners were installed at the ends of each stiffener to preclude premature separation of the stiffeners from the skin because of skin-stiffener debonded at the stiffener ends. Three panels were fabricated. The third panel was subjected to a "field-type" repair consisting of two metal angles and one metal external doubler joined with mechanical fasteners as indicated in Fig. 5. At a cyclic load level of 56 kN/m (which is 96% of the design limit load), the debond in the panel did not grow to a size that produced panel failure, and the residual strength was relatively unaffected. Quasi-static loading of these three specimens indicated that the fasteners at the end of the stiffeners had a definitely beneficial effect on panel strength. The "field-type" repair applied to the third damaged panel was totally successful in preventing further extension of the debond.

**FIG. 4**—Static strength versus design/predicted failure loads. (Fig. 38 of Ostrom; SI axis added.)

**FIG. 5**—Repair design. (Fig. 33 of Ostrom; SI dimensions added.)
The conclusions of the authors are reproduced below in their words.

The results of this exploratory research project demonstrate that viable post-buckling designs of flat composite shear panels representative of lightly loaded fuselage structure are feasible if reasonable design buckling ratios of 5.0, typically associated with thin-skinned, lightly loaded metallic shear panels, can provide composite structures which meet structural integrity and design life requirements.

The experimental results confirm that skin-stiffener debonding is the critical failure mode for these cocured, integrally stiffened, thin-skinned shear panels. Both static and fatigue test failures initiated at the skin-stiffener interface, as a result of failure of the matrix material. The fatigue test results further indicate the existence of an apparent endurance limit, below which fatigue-initiated debonding does not occur.

The static and fatigue test results also indicate that the load concentrations associated with the use of simple "picture frame" shear loading fixtures caused skin-stiffener debonding to occur at lower nominal load levels (or at fewer load cycles) at the loaded corners of the panels than in the central region of the panels. In addition, the termination of the stiffeners at the edges of the panels without the use of special precautions conventionally used to resist out-of-plane forces at the ends of stiffening members resulted in extension of the debonding. At the same time, the use of mechanical fasteners at the ends of the stiffeners restrained extension of skin stiffener debonding from the stiffener ends, and in higher panel failure loads. It is expected that actual fuselage structure, with a lower degree of load concentration at the intersection of stiffening members, and with provisions at the intersections for resisting the out-of-plane forces, would sustain higher loads (or more load cycles) before initial skin-stiffener debonding occurred, and also would sustain higher loads before final panel failure.

Both the static and the fatigue test results indicate that these structures have appreciable damage tolerance. Again, for panels designed for reasonable buckling ratios, manufacturing or service-induced debonds did not grow under fatigue loadings to the extent that panel failure occurred. Furthermore, panels with existing debonds had residual strengths comparable to their original static strength.

The comparison of predicted and actual test results indicates the need for further development of post-buckling analysis methods suitable for prediction of skin-stiffener debonding.

This report is a well-written and well-organized important addition to the literature on this subject. Post-buckling behavior of composite materials and structures is an area that needs the attention of researchers and engineers and represents a potentially profitable opportunity for exploitation by ingenious (and courageous) designers. The present report appears to support such a premise.