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ADVANCED COMPOSITES—
THE STRUCTURES OF THE FUTURE

PLANTEMA MEMORIAL LECTURE

PRESENTED BY

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ADVANCED COMPOSITES—THE STRUCTURES OF THE FUTURE

ABSTRACT

The U.S. Government has for the past two decades sponsored considerable research and development effort to exploit the very attractive structural efficiency achievable through the use of advanced composite structures. The NASA part of this effort has produced a wide variety of structural components that are now in flight service in military and commercial aircraft. This flight service is being accumulated at the rate of about 300,000 component hours per year and thus far, has been generally satisfactory. In most cases 20 to 25% of the structural weight has been saved. The flight service programs are accompanied by more detailed research studies that evaluate environmental effects, develop analytical procedures useful in design, quantify fatigue and damage tolerance and demonstrate practical manufacturing and repair techniques.

The objective of this work is to develop the technology that leads to the early, progressive use of composite structures in production commercial transport aircraft. The 1977 Plantema Lecture seems an appropriate means for introducing composites to ICAF wherein they are likely to be the focus for future meetings.

INTRODUCTION

Advanced composites offer promise of substantial weight savings relative to current metallic structures. Further, the number of parts required to build a composite component may be dramatically smaller than the number of parts needed to construct the same component of metal alloy. This can lead to significant labor savings sometimes offsetting the somewhat higher price of the present composite materials. These features, together with the inherent resistance to corrosion offered by composites, make composites very attractive candidate materials for future aircraft structures.

The National Aeronautics and Space Administration has conducted research during the past decade to demonstrate the viability and desirability of using composites, principally for civil aircraft. Similar research under sponsorship of the Department of Defense has been in progress to exploit these materials for military aircraft systems.

The purpose of this paper is to outline some of the NASA programs, to cite the structural improvements demonstrated, and to report on complementary research to create the technology[1] needed to design composite structures with confidence.

FLIGHT SERVICE PROGRAMS

An early emphasis of the NASA effort to demonstrate advanced composite structures was to sponsor the development, construction, and qualification of structural components to fit existing aircraft and then to place them in routine service operations on such aircraft. Most of the components have been for commercial aircraft for several reasons. First, this focus provided experience for the manufacturers and operators to build their confidence in the safety, reliability, maintainability, and cost effectiveness in composite structures. Second, civil aircraft have high utilization rates and offer exposure to essentially all environmental conditions likely to be encountered anywhere in the world. Special emphasis was placed on accumulating inspection data for all the flight experience gained through these operations.

As will be seen in succeeding sections, the early applications featured selective reinforcement of otherwise metallic structures, with advanced composites. These were placed in military service. Later, small secondary components were developed and placed on commercial aircraft in routine service. Currently, larger secondary components are being developed with the intent that the manufacturers may use these components in future production and to choose such components for the next generation of commercial aircraft.

Thus, experience and confidence have been built progressively with minimum risk and at moderate cost. The first-hand experience of the major U.S. commercial aircraft manufacturers and representative operators around the world should provide convincing evidence that composite structures are the logical choice for future airframes.

Typical Flight Service Program

The flight service programs implemented to date have generally followed the outline in Figure 1. Typically, the development, detail design, fabrication, and ground test phases have taken about 3 years to complete. Tests to evaluate environmental effects on the material properties begin during the fabrication phase and generally continue throughout the flight service phase. Flight service is usually scheduled to continue for 5 years or more. When numbers of components permit, selected components are removed from service for tests. If all goes well, as is expected, the remaining components will be left in service after the scheduled flight service evaluation. Thus, the minimum total time required for a representative program is 8 years.

During the course of such a program, a great many specific tasks are performed. During advanced development, the general concept is developed, preliminary tools are developed and some subcomponent tests are performed. During detail design, the structural design is refined, the tooling is verified and more components tests are conducted to evaluate specific details. During fabrication, the final tooling is constructed, manufacturing processes are perfected and all necessary ground test and flight service components are fabricated. The ground test phase features numerous static, fatigue and vibration tests, and the results are submitted to the Federal Aviation Administration to secure its approval and certification. Environmental effects are evaluated on material samples exposed in laboratory-controlled chambers and in outdoor racks at representative airline terminals and at the Langley Research Center. Some of the laboratory tests are conducted under cyclic temperature and humidity. The final phase is, of course, the flight service, which is monitored by periodic visual and ultrasonic inspections and removal of selected components for testing. The results are summarized and disseminated through a systemmatic annual reporting system.

Flight Service Status

The current status of this flight service program is given in Figure 2, in which each of the components has been listed in chronological sequence according to the date when the first component of each type entered service. Each component program is discussed in detail in later sections of this paper. Note that the first component, a selectively reinforced CH-54B helicopter tail cone entered service in March of 1972 and is still in service. The spoilers for the B737 feature the largest number of replicate components. Twenty-seven ship sets of four each have been in domestic and foreign airline service, beginning in July of 1973. In all, 154 components have been installed on 46 individual aircraft and are being operated by 14 commercial airlines, the U.S. Army, and the U.S. Air Force.

CH-54B Helicopter Tail Cone - This helicopter, shown in Figure 3, is the U.S. Army's Flying Crane and is used for a variety of Army tasks to deploy large and heavy cargoes. The initial design developed an undesirable resonant bending of the aft fuselage for certain combinations of payload weight and sling length. The situation was improved by stiffening the tail cone with heavier aluminum-alloy skins. This stiffening created a substantial increase in weight of this component.

The Sikorsky Aircraft Division designed and fabricated an alternate tail cone under a joint NASA-U.S. Army contract (Ref. 1 and 2). This design featured unidirectional boron-epoxy strips bonded to twelve stringers on the top and bottom skins of the tail cone. The strips extend over the 5-meter length of the tail cone, shown shaded in the figure. Although the composite accounts for only 8 percent of the total (180 kg) mass of the tail cone, the desired stiffness was provided with a 14-percent weight saving.

The boron-epoxy-reinforced construction was evaluated in a variety of static strength, stiffness and fatigue tests to verify adequacy of the design. One full-size component was fabricated and installed during production of a CH-54B and was delivered to the U.S. Army in March 1972 for normal service use.

This helicopter has now been in regular service at Ft. Eustis, Virginia for over 5 years. Sikorsky inspects the composite-reinforced parts of this helicopter regularly, and because Ft. Eustis is only a few miles from the Langley Research Center, NASA personnel occasionally participate in this inspection. Service has been satisfactory, but because of energy-saving restrictions, this machine accumulates flight time at a rather low rate.

L-1011 Fairing Panels - The Lochkeed-California Company is working under NASA contract to gain at least 5 years of flight experience on 18 fairing panels made of Kevlar 49/epoxy (Ref. 3-5). These fairings have been installed in pairs on L-1011 aircraft operated by Trans World, Eastern, and Air Canada Airlines. The panels are located as shown in Figure 4 and they are portrayed prior to installation in Figures 4 and 5. The fairings are of three distinctly different types.

The wing-to-body fairing (Fig. 4) measures approximately 1500 by 1700 mm and is slightly curved. The panel is constructed with a Nomex honeycomb core and Kevlar 49/epoxy faces. Its 7-kg mass is 25 percent less than that of the fiberglass panel it replaces. Each panel is sprayed with aluminum to discharge electrical potentials from the fairing to adjacent structure. The ground test panel failed under simulated pressure loading at 125 percent of the desired ultimate pressure.

A similarly constructed triangular-shaped panel (Fig. 5), measures approximately 2 meters long by 750 mm wide. This panel is installed near the center engine where some elevated temperature is experienced. Thus, a higher-temperature curing epoxy was chosen to construct this panel. The 2.3 kg mass of this panel is 30 percent less than that of its fiberglass counterpart.

The third fairing (Fig. 5) is approximately 830 mm long, 140 mm wide and has 1 kg mass, 32 percent less than that of the original production fairing it replaces. This component is made of a solid Kevlar 49/epoxy laminate. It is 2.3 mm thick near the middle and is tapered to 0.8 mm near its edges.

Over four years of flight service have been accumulated on each of these panels with no unusual experiences to date (Ref. 4). One ship-set had to be removed and reinstalled on another aircraft because of a fire that severely damaged other parts of the aircraft.

B-737 Spoilers - The largest number of components in a single part of the flight service program is found in the B-737 spoiler program (Ref. 6 and 7). The Boeing Company has fabricated 139 graphite/epoxy spoilers such as those shown in Figure 6. Of these, 108 have been installed in sets of four on 27 aircraft operated by the six airlines listed in the figure. The remaining spoilers have been subjected to a variety of ground tests or are being reserved as spares. The spoilers are about 1300 mm long by 560 mm wide and have 6 kg mass. In the first series, three different graphite/epoxy composite materials were used for the skins, which were adhesively bonded to the production aluminum honeycomb core and aluminum-alloy leading-edge-spar and hinge fittings. The finished spoilers are 35 percent composite and are 17 percent lighter than the production aluminum spoilers they replace.

Three to four years and up to 9000 hours of flight service have been accumulated by some of these spoilers (Ref. 7). In this program, randomly selected spoilers are removed from service annually for intensive inspections and destructive tests to identify whether strength properties have been degraded. Results of these tests are given in Figure 7. The scatter band represents the range of results obtained in tests of as-fabricated spoilers having no service experience. The symbols represent results of tests on components removed from service after the times indicated. No systematic deleterious trend was found in these data. A few isolated cases have been found of exfoliation corrosion of aluminum parts, but this was attributed to insufficient edge sealant. Service reports from the operators have not indicated any unusual problems. A few of the spoilers have suffered significant mechanical damage, but these have been repaired and returned to service.

A second-generation program was also undertaken by Boeing to produce all-composite spoilers (Fig. 8). Metal parts were replaced by glass/epoxy honeycomb cores and molded chopped-graphite hinge fittings, close-out ribs and leading edge spars.

In addition, the cover sheets were made of graphite/polysulfone, a thermoplastic material that is expected to be less costly to manufacture. The higher density of the glass honeycomb was more than offset by the lower density of the molded composite fittings. Thus, these spoilers were 20 percent lighter than the all-aluminum spoilers they replaced.

These advanced spoilers have been removed from service because of unforeseen chemical degradation by hydraulic fluids. Figure 9 shows how the covers delaminated near the actuator fitting. This degradation was detected before any spoiler failed in service. Material property evaluation done after boiling specimens in the hydraulic fluid used in commercial aircraft did not identify this potential failure. Thus, flight service evaluation was a very valuable experience. Based on this experience, polysulfone matrix materials in their present form are not expected to be viable for applications where parts are exposed to hydraulic fluids currently used in commercial aircraft.

<u>C-130 Center Wing Box</u> - The Lockheed-Georgia Company (Ref. 8-11) has constructed three center wing boxes for C-130 Hercules aircraft using unidirectional boron/epoxy strips to reinforce both covers (Fig. 10). One was subjected to fatigue tests to simulate four lifetimes of service loading, to an additional lifetime of fatigue testing to demonstrate crack propagation after cracks and delaminations were artificially intro-

duced and then to a residual strength test (Fig. 11). All these tests and several more on smaller components gave good assurance that the structure would perform satisfactorily in service. The remaining two boxes were installed in U.S. Air Force aircraft for normal tactical airlift service, beginning in October 1974. Special inspections are being conducted at 6-month intervals for 6 years. No service-induced defects have been detected at this writing.

These wing boxes are 11 meters long, 2 meters wide and 0.9 meters deep, to match precisely the standard all-aluminum boxes they replace. The boron strips were applied to the full length of the covers on both the wing planks and on the hat-section stringers. Only 8 percent of the total mass of the box is boron-epoxy, but the boxes so reinforced are 10 percent lighter than the equivalent aluminum alloy structures.

DC-10 Aft Pylon Skin - The Douglas Aircraft Company (Ref. 12) has designed, fabricated, and installed three boron/aluminum skin panels on DC-10 aircraft (Fig. 12). These panels are installed just above the center engine where elevated temperatures and rather high-intensity acoustic loadings were expected. Thus, boron-aluminum was selected for this application. The panels are 1700 mm long and 200 mm wide, and contain 11 plies $(0, \pm\frac{\pi}{4}, \frac{\pi}{2})$ of boron/ aluminum. The panels are 26% lighter than the titanium panels normally installed. Three units have operated satisfactorily since they were placed in airline service beginning in August 1975.

DC-10 Upper Aft Rudder - The Douglas Aircraft Company (Ref. 13) is under contract to design, fabricate, and place in flight service 10 graphite/ epoxy upper aft rudders for DC-10 aircraft. Each unit is 4.1 meters long and 1 meter in chord. The entire unit is made and co-cured in one piece except for the standard aluminum alloy hinges and actuator fittings and the standard glass/epoxy leading and trailing edge members and tip assembly. This rudder has a mass of 26 Kg of which 77 percent is composite material and this mass is 37 percent less than that of the production aluminum component it replaces.

As illustrated in Figure 14, these rudders were fabricated by the so-called "trapped rubber" process. The process began by laying up flat sections of the skins. Preformed spars and ribs were assembled in a tool and were stacked with blocks of silicone rubber that were formed to the proper shape to completely fill all internal cavities. The skins were then laid against these members and an external steel tool was clamped around the complete assembly. Curing was accomplished in an oven at 450K (350°F). The expansion of the rubber at this temperature provided the necessary pressure to assure proper molding and positioning of all parts. After curing, the rubber was removed through holes in the forward spar.

The first of these rudders was installed on a commercial airline DC-10 in June 1976. To date six have been installed and the remainder will be in service by the end of 1977. Visual and ultrasonic inspections will be performed at prescribed intervals during a five-year service period.

Flight Service Summary - The curve in Figure 15 illustrates the cumulative flight service history for the components described in the foregoing sections. By July 1977, these components will have accumulated one million flight hours and by the end of 1981 some of the earliest components will approach 10 years in service with an aggregate of 2-1/2 million component flight hours. Manufacturers and operators will have gained much experience that should inspire confidence in the use of composites for future aircraft.

FUTURE STRUCTURES PROGRAMS

During the past 2 years NASA has been laying out a program known as Aircraft Energy Efficiency (ACEE). Its broad purpose is to conduct research which will lead to significant improvements in fuel economy of civil transport aircraft. Although other disciplines are included, one of the major thrusts of the program is to dramatically reduce the total mass of the aircraft structure. This is being done by expanding the use of composites to larger secondary components and into selected primary structural components. The objective is to develop technology and confidence so that commercial transport manufacturers can commit to production of composite structures in their future aircraft. The technology readiness dates are to make such commitments for secondary structures in the 1980 to 1985 time frame and for primary structures in 1985 to 1990. The technology desired must include verified prediction methods needed for design and cost competitive manufacturing processes. Weight effectiveness is, of course, the fundamental advance that leads to energy conservation.

This program seeks to build the required confidence through airline acceptance and FAA certification, to verify manufacturing costs through construction of significant numbers of components in a production mode, and to develop appropriate durability assessments that should lead to the customary warrantees agreed upon by manufacturers and their clients.

This is admittedly an ambitious undertaking and will be carried out in several phases. As indicated in Figure 16, each of the three major U.S. commercial transport manufacturers will receive contracts to build major secondary components. Douglas will produce more DC-10 upper aft rudders using production tooling, Lockheed-California will build ailerons for the L-1011, and Boeing will build elevators for the B-727. In the second phase, Lockheed and Douglas will build vertical fins for the L-1011 and DC-10, respectively, and Boeing will build the horizontal tail for the B-737. In each case, the components will be designed to match the aluminum alloy structures they might replace, complete ground testing will be performed to secure FAA type certification and realistic manufacturing costs will be developed. In some cases, the structures may also be placed in airline service and at least some of these activities are expected to lead to a production change for future aircraft of the types involved.

In parallel with these efforts all three contractors will be asked to conduct design studies leading to the construction of selected composite primary wing components. Contingent on the results of these studies one manufacturer will be awarded a contract to construct a particular primary wing component, to conduct extensive ground tests and perhaps to construct flight-qualified harware for installation on a transport aircraft.

MILITARY PROGRAMS

Obviously, NASA is not alone in developing composite aircraft structures. Both the U.S. Air Force and the U.S. Navy have conducted similar development programs for years and these have resulted in the use of composites in the production of the components listed in Figure 17. Although military aircraft do not usually see as much flight service in a given unit of calendar time as do civil transports, several other considerations make the challenge greater for military vehicles. Among these are:

(1) More severe loading experiemce

(2) Higher flight speeds involving aerodynamic heating

(3) Somewhat less rigorous maintenance capability

(4) Severe corrosion and humidity environment, especially for carrier-based aircraft

Nevertheless, confidence is high enough to have led to production commitments for a significant number of aircraft types. Improved performance because of lower mass is usually the primary driving force in these decisions, but costs and reliability must also be commensurate with those of competing metallic designs.

The U.S. Army operates a very large number of helicopters, where the use of composites offers significant improvements in weight, but primarily in labor cost for manufacturing. Many studies have been undertaken to develop filament winding techniques for rotor blades, parts of fuselages, and many other parts. Quite frequently, glass fibers are used in the composite with graphite and aramid fibers for additional strength and stiffness.

NASA has collaborated with the Army on several component developments for helicopters. One of these is illustrated in Figure 18 in which a low-cost manufacturing procedure was developed for constructing fuselage structures. The process started with assembly of foamed cores for rings and stringers as shown in the lower left of the figure. Kevlar/epoxy skin with the appropriate orientation, and graphite-epoxy stringers were laid over these framing members as shown and the entire assembly was co-cured in one step. The dramatic reduction in labor cost and number of parts needed is obvious and led to a 15-20 percent lighter structure projected to cost 5 percent less than its conventional aluminum counterpart. Laboratory test results on the component were very encouraging.

COMPANION RESEARCH

Many of the foregoing examples involved structures that were designed primarily for stiffness rather than for strength. This feature, plus the secondary nature of the components, made safety and integrity somewhat less critical design considerations than will necessarily be the case as composites are moved into larger primary structural components. Consequently, systematic basic research is being conducted to produce the design technology needed to produce optimum design in which integrity and safety are prime considerations. Some of these studies are cited in the following paragraphs.

Static Properties of Composite Materials - Because graphite fibers and epoxy matrix material are both rather brittle, the advanced composites made from them also exhibit rather brittle behavior; failure usually occurs at about 1 percent strain. Properties are, of course, strongly effected by orientation of the fibers in the composite. Maximum axial strength is achieved when all fibers are oriented parallel to the axis of loading. However, if a hole or other discontinity is present, such composite first develop axial cracks beginning near the edge of the discontinuity. The ultimate strength is then that which is expected for the remaining net section and the ultimate strain is of the order of .01.

For each different set of ply orientations, the strength properties are essentially those that would be expected from the rule of mixtures, but the strain at failure tends to be of the order of .01. Because of the complex interaction of fibers oriented in various directions and the matrix material that unifies the composite, the failure modes are very different for each set of ply orientations. Similarly, the introduction of holes and other discontinuities in composites of various orientations produces various effects on strength and ultimate strain. The quantitative effects are actively under study with the objective that laminate behavior can be predicted adequately from appropriate manipulation of empirical data obtained from tests of laminae.

Exploratory tests are being conducted by Mikulas, Rhodes, Williams, and Starnes at Langley Research Center to evaluate effects of lower velocity impact damages such as might be produced by runway debris or other accidental means. The results suffest that both tension and compression strengths of thin laminates are 40 to 60 percent less when impacted by 13 mm-diameter (0.5 in.) aluminum balls travelling at a wide range of velocities (Figure 19). This loss of strength is of the same order as that due to fastener holes that are likely to be found in a structure.

These and other related considerations are likely to limit allowable design ultimate strain to the order of .005 in civil aircraft structures for the forseeable future. This limit is roughly equivalent to the strain limits employed in major portions of current aluminum-alloy structures because of various manufacturing compromises (Ref. 14).

Studies have been made (Ref. 15) of the effects of low-velocity impact damage on compressive strength of hat-stiffened panels. The strength of panels designed to resist buckling under modest loading intensity (.0034) were found not to be affected significantly by the impact. However, the strengths of panels designed not to buckle below .008 strain were damaged severely enough to reduce their buckling strain to approximately .004, a value slightly lower than that attained for a panel containing a 13-mm-diameter hole.

Moisture Effects - Those familiar with behavior of epoxy materials are familiar with the fact that these materials inherently absorb moisture from their surroundings. This moisture has the effect of "plasticizing" the polymer, reducing its "glass transition temperature" and also its mechanical properties. To a fair first approximation the amount of moisture absorbed under a given set of environmental conditions is the same regardless of the particular polymer. However, this amount is profoundly dependent upon the environmental conditions and the effect on properties is different for various polymers.

To shed some light on this phenomenon, analytical studies have been performed (Ref. 16) to predict the amount of moisture absorbed for various practical service environments. In the first step, weather bureau information on relative humidity and temperature levels were obtained for a wide variety of locations around the world. A computerized diffusion model was developed to determine the rate at which moisture is absorbed into a composite. Figure 20 shows the prediction for moisture absorbed by a 5208 epoxy composite 12 plies thick and exposed at several cities. Except for cities located in desert areas, the equilibrium level was approximately 1% of the weight of the matrix assuming the material was exposed in the shade. Later refinements included the effects of solar heating and various flight scenarios, including the appropriate temperatures and humidity levels at assumed altitudes. As seen in Figure 21 these assumptions led to estimated equilibrium moisture levels of approximately 0.75%. Obviously, these levels depend upon the thermal absorptivity, α, of the material and its paint or other coating. The effect of this parameter is reflected in the Figure. Generally, the equilibrium level is predicted to be reached within 1 year for composites up to 12 plies thick, but longer times might be required for thicker laminates. As expected, the seasonal variation in moisture content in thin laminates is predicted to be more significant than that in thicker laminates.

As an adjunct to the flight service programs described earlier, racks of small specimens are exposed at the main maintenance facility of each of the airlines flying the components. Figure 22 displays two types of exposure racks -- one (on the left) in which the specimens are exposed passively, and the other (on the right) in which an axial tensile load is applied. The map (center) identifies the several locations at which these specimens are exposed. Selected specimens are returned to NASA-Langley periodically for inspection, for determination of moisture content, and for mechanical test. Representative moisture determinations, shown in Figure 23 agree reasonably well with those predicted for specimens exposed to the sun at non-desert locations. The effects on properties measured at room temperature are shown in Figure 24 and are seen to be quite modest. A few data have been accumulated in elevated temperature tests of specimens containing known amounts of moisture. Although results are not yet statistically significant for a range of materials, the polymers cured at higher temperatures seem to have better resistance to moisture levels expected in subsonic aircraft than do those cured at lower temperatures. Several manufacturers prefer matrices cured at 450K (350°F) for this reason.

Fatigue Properties - As of this writing, fatigue failures have been so rare in composite structures that one frequently hears the generalization that advanced composites do not suffer fatigue. However, as cited earlier, present experience is almost exclusively with structures that are stiffness-critical and, thus, enjoy comfortable margins on strength or strain. Further, many of the components are essentially free of discontinuities that are common in wing or fuselage structure. Prudence dictates that this generalization be examined carefully when composites are introduced into primary structures where higher structural efficiencies are likely to be required. Current understanding of the basic failure modes is rather limited and systematic quantitative data are rare.

However, the following general comments seem to characterize the current state of knowledge. As seen in Figure 25, the mode of failure is clearly influenced by the particular fiber orientation in the laminate under consideration. Uniaxially alined fibers offer the highest load carrying capability, but are not practical in complex structures except as local reinforcement strips. In contrast to metallic parts which fail by cracking transverse to the maximum tensile force, uniaxial laminates fail in fatigue by forming and propagating axial cracks from the extremities of holes or other discontinuities.

Although the axial static strength is hardly affected by such cracks, a part built of such a laminate would soon possess no transverse tensile or shear strength. The failure mode is essentially identical under compression loading, but a degraded strength could result if the remaining ligaments were free to buckle. A laminate containing fibers alined only in the $\pm \frac{\pi}{4}$ directions generally fails along one or both of the fiber axes. Laminates containing three or more fiber orientations sometimes develop rather complex failure modes involving considerable delamination near a discontinuity, progressive matrix cracking and fiber breakage, and, ultimately, a rather diffused transverse failure. The repeated strain levels that can be sustained for prescribed numbers of cycles depend on the orientations of the laminae and the proportions of each orientation in the laminate (Fig. 26). Stacking sequence can also modify these results for some orientations.

From these considerations, a very large number of fatigue tests would be required to characterize adequately the fatigue capabilities for design purposes. To avoid such extensive testing, systematic studies are underway to seek a more detailed knowledge of the failure mechanism and its sequence. Various means are employed to monitor the failures as they progress.

In early studies (Ref. 17) boron-epoxy sheet specimens with various orientations and containing central holes were subjected to axial loads (Fig. 27). The X-ray photos were taken at regular intervals during tests. These were enlarged and examined to identify the sequence of fiber failures. Representative results are shown in Figures 28 and 29.

Figure 28 is for a specimen with half of its fibers alined with the axis of loading and the other half at $\pm \frac{\pi}{4}$. The axial tension load ranged from zero to approximately 2/3 that required to fail the specimen statically. The symbols indicate that almost no axial fibers failed and that $-\frac{\pi}{4}$ oriented fibers failed along a line parallel to the loading axis. These fibers were,. of course, loaded very highly by shear forces that had previously caused matrix disintegration along the same path.

In contrast, Figure 29 is for a specimen that has 25% of its fibers alined in each direction: axial, transverse, $\pm \frac{\pi}{4}$. In this case, fibers failed in a narrow zone oriented at right angles to the loading axis. Although no transversely oriented fibers failed, they apparently caused constraints which led to numerous failures of the $\pm \frac{\pi}{4}$ fibers, including some near the edge of the specimen. Similar specimens tested under repeated compression loadings led to essentially the same matrix failure sequence, but did not lead to fiber failures within the 10^7 cycles of loading applied.

Thermograms, such as those shown in Figure 30, have been used (Ref. 18) to trace the progress of damage during many of these tests. Such thermograms help the experimenter decide when to make more detailed observations. The four thermal patterns shown are for two stacking sequences and for two laminate orientations. In these examples, the laminates containing transverse fibers develop heated zones oriented transverse to the specimen and those without such fibers develop zones that are oriented more nearly axially, qualitatively consistent with the X-ray studies described earlier. This local temperature rise has been observed to be as high as $100^{o}C$ prior to failure of a boron-epoxy specimen (Ref. 19). Detailed studies have shown that the heated areas are more nearly related to matrix disintegration which precedes fiber failure.

To define the extent of matrix damage, some specimens were sectioned and examined with a scanning electron microscope. A consistent observation is illustrated in Figure 31 which is for a $(0/\pm\frac{\pi}{4}/0)_s$ laminate tested under repeated compression loading. The earliest damage was found to be intralaminar cracking of the matrix as seen in Sections A-A and B-B. Trans-thickness cracks were also found in Section A-A and C-C. The voids seen in these micrographs are apparently due to extensive breakup of matrix material along the path of maximum damage.

Similar research is in progress on graphite/epoxy laminate, in which identification of mechanism is much more difficult because of the much smaller fibers included. X-ray analysis is not practical, because graphite fibers are essentially transparent to the rays.

Studies such as these and analytical developments that accompany them (Ref. 20 and 21) are expected to lead to the development of a rationale for predicting fatigue behavior for any arbitrary laminate. Empirical data obtained at the lamina level are expected to be required. Until such a prediction scheme is developed, designers will be planning to generate laminate data directly for the particular combinations of materials, lamina orientations, stacking sequences, and configurations of interest.

CONCLUDING REMARKS

Advanced composites have been demonstrated to be viable for aircraft structural applications. Their use in such applications should become much more common within the next decade as a result of NASA and military programs that have done much to inspire confidence on the part of manufacturers, operators, and airworthiness authorities. Such composite structures show promise of saving as much as 20 percent of the weight of a metallic aircraft structure and, as manufacturing methods are developed, should become at least cost competitive.

Technologies that need better definition to promote confidently the most optimum structures are: moisture and other environmental effects, impact damage resistance, fatigue and damage propagation resistance, and certification procedures.

Future ICAF meetings will undoubtedly feature many discussions on this new and challenging engineering material.

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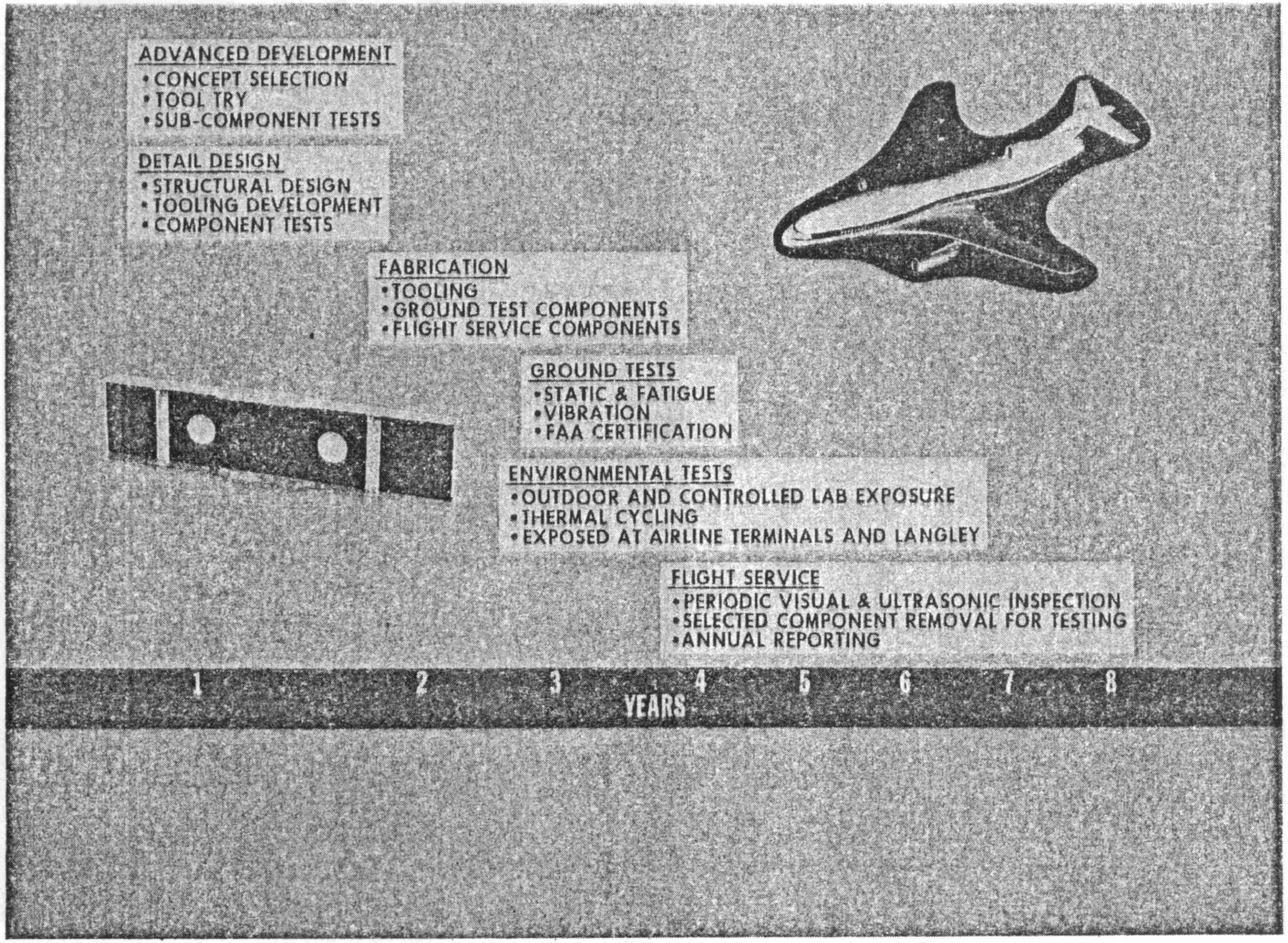

Figure 1.- Typical Flight Service Evaluation Project

A/C AND COMP. % AND PRINCIPAL COMPOSITE	START OF FLIGHT SERVICE	NUMBER PARTICIPATING		CUMULATIVE FLIGHT HOURS			
				APRIL 1, 1977		DECEMBER 31, 1981	
		A/C	COMP	HIGH TIME A/C	TOTAL COMP	HIGH TIME A/C	TOTAL COMP
CH-54B TAIL CONE 8% BORON/EPOXY (REINF)	MARCH 1972	1	1	730	790	1000	1000
L-1011 FAIRING PANELS 100% KEVLAR 49/EPOXY	JANUARY 1973	3	18	10952	153600	23 410	353 000
B-737 SPOILERS 35% GRAPHITE/EPOXY	JULY 1973	27	120	8982	758000	19 734	1 938 000
C-130 CENTER WING BOX 8% BORON/EPOXY (REINF)	OCTOBER 1974	2	2	1825	3505	5176	10 260
DC-10 AFT PYLON SKIN 100% BORON/ALUMINUM	AUGUST 1975	3	3	4540	13050	19 750	59 250
DC-10 UPPER AFT RUDDER 77% GRAPHITE/EPOXY	JUNE 1976	10	18	3250	8003	16 300	155 000
TOTALS		46	154	30,360	937760	85370	2 534 310

Figure 2.- NASA Flight Service Summary

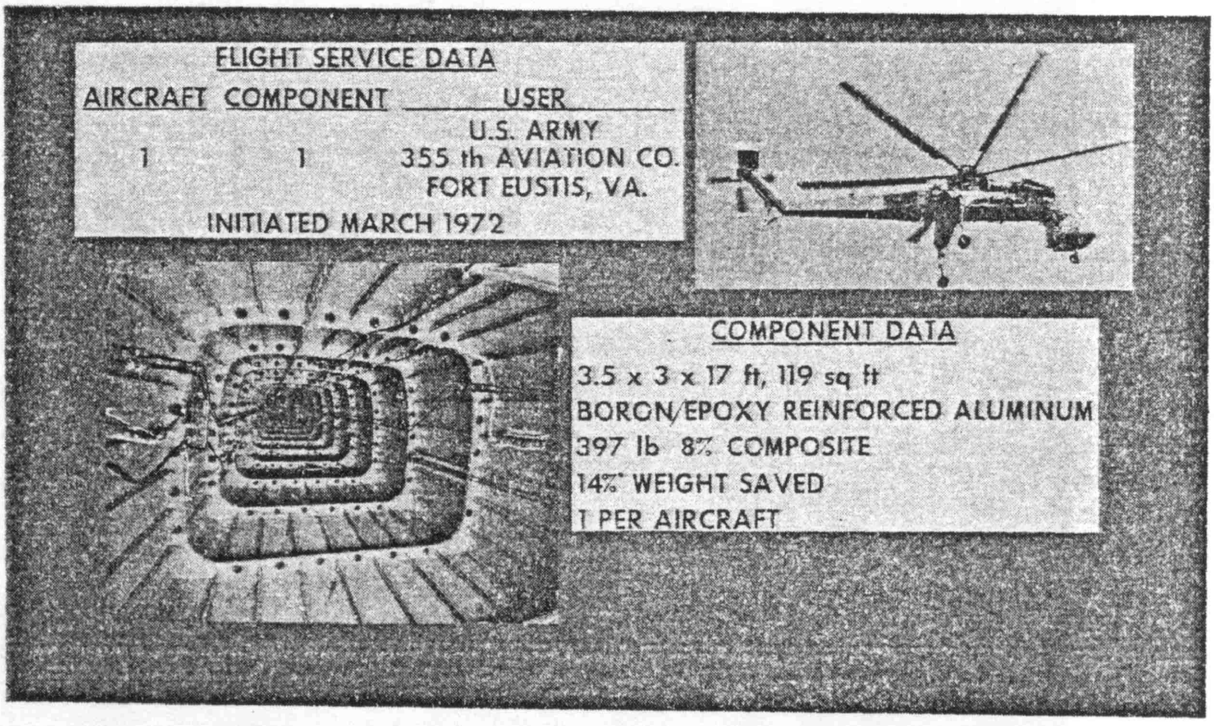

Figure 3.- CH-54B Tail Cone

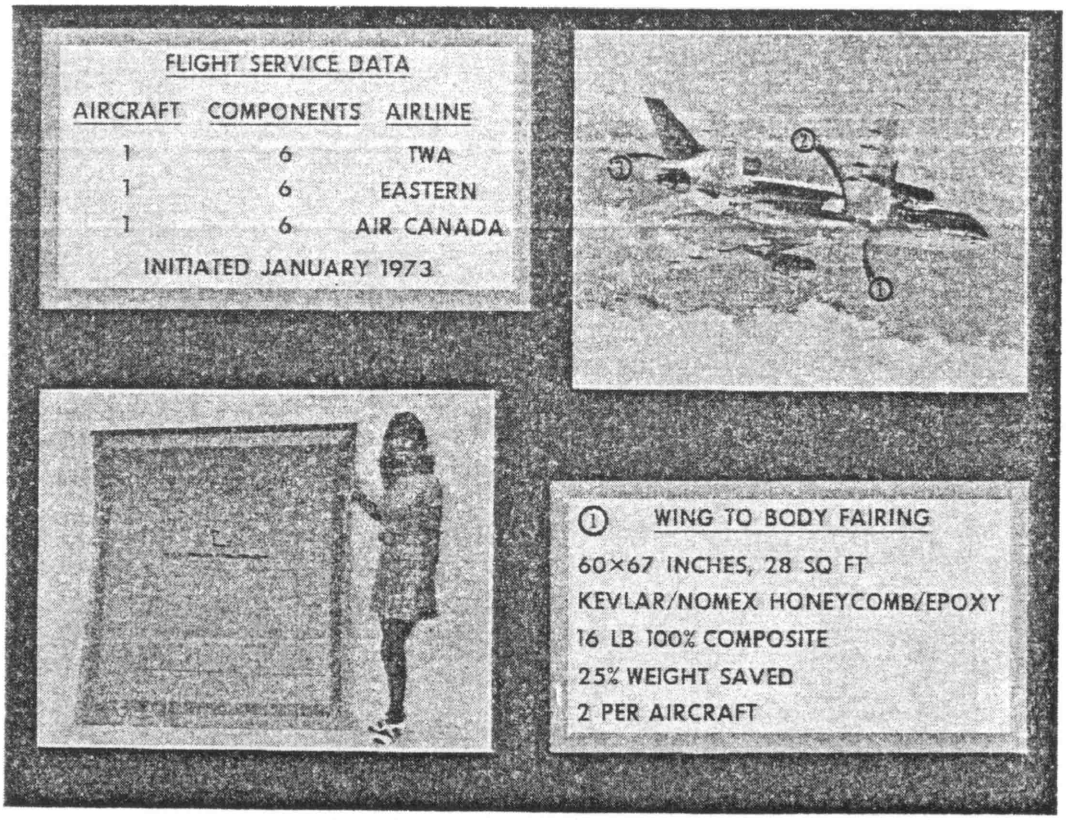

Figure 4.- L-1011 Fairing Panels, I

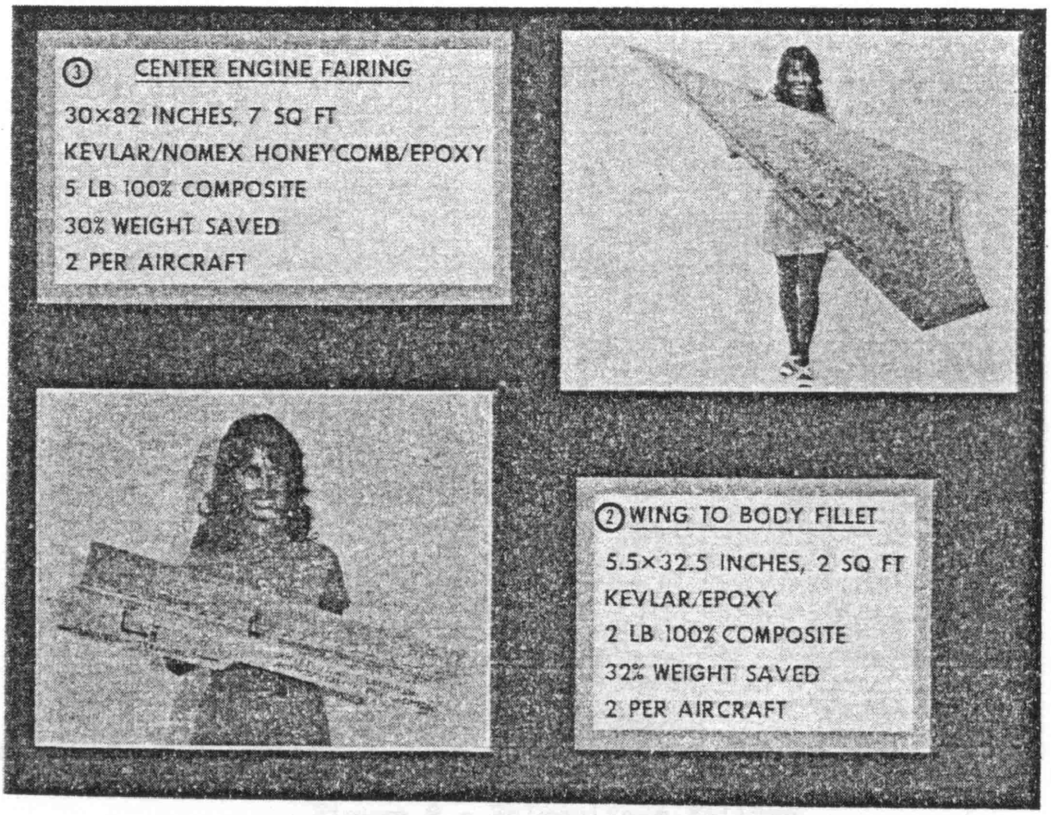

Figure 5.- L-1011 Fairing Panels, II

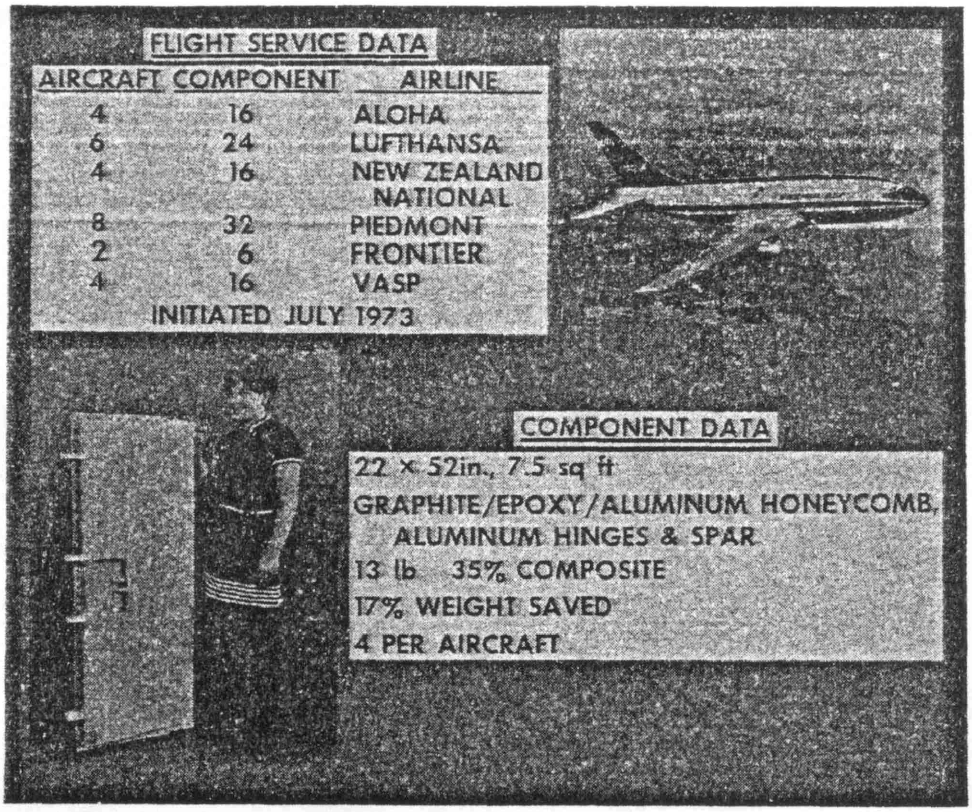

Figure 6.- B-737 Spoilers

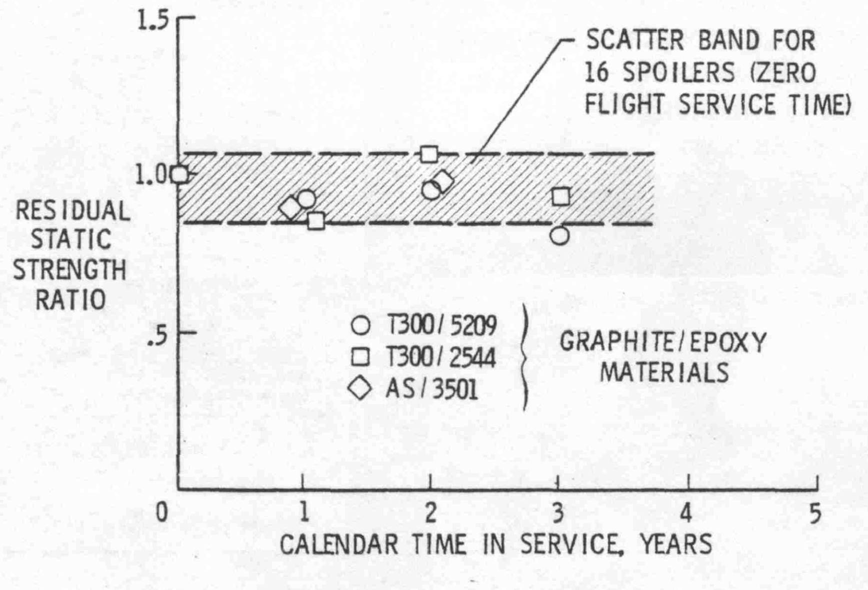

Figure 7.- Residual Static Strength of Graphite/Epoxy Spoilers After Removal from Flight Service

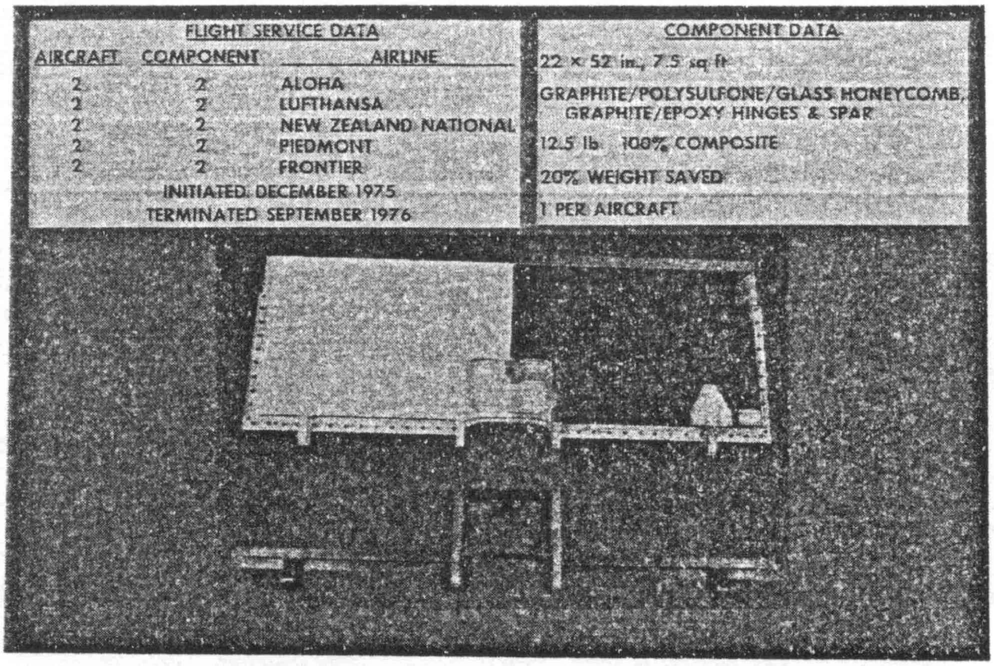

Figure 8.- B-737 Advanced Spoilers

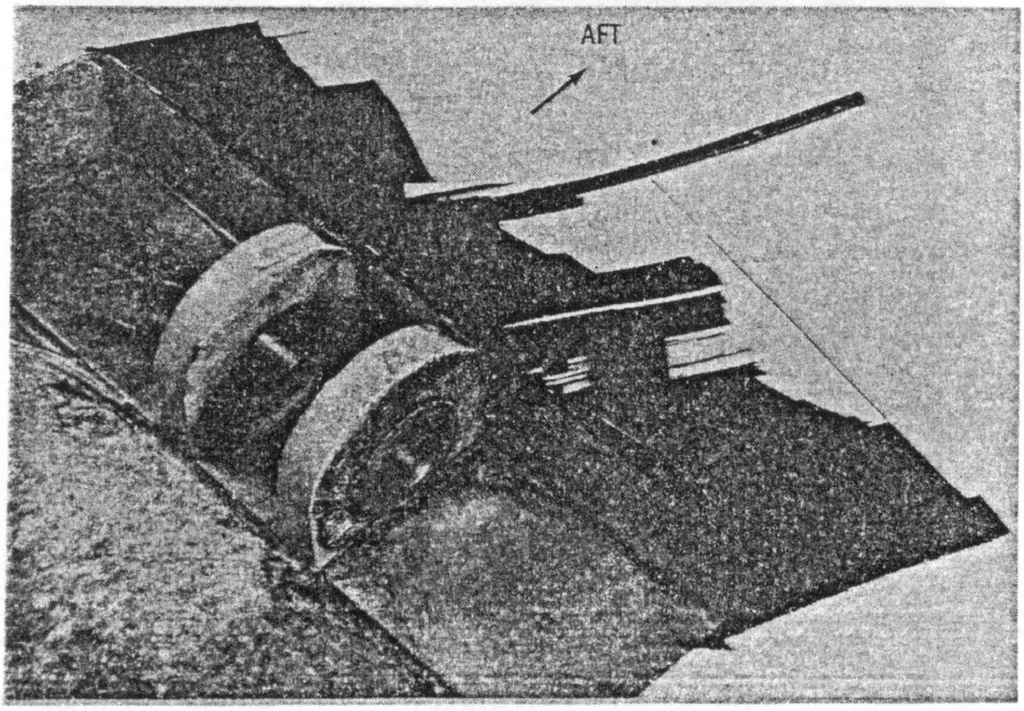

Figure 9.- Polysulfone Failure

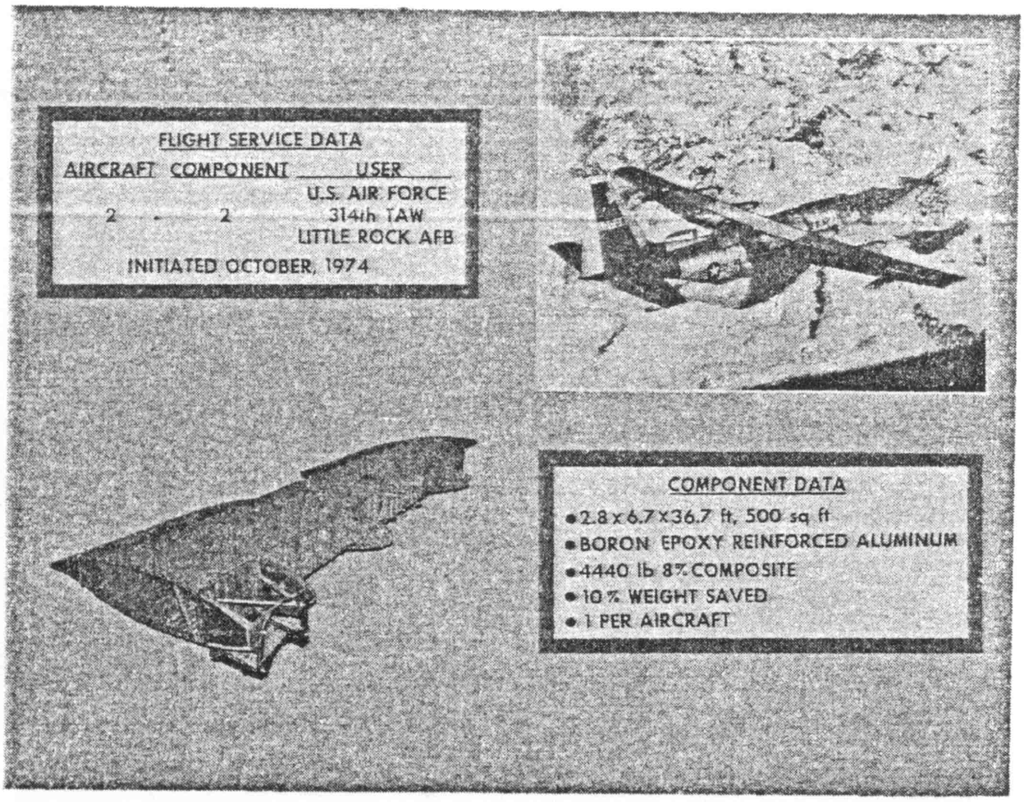

Figure 10.- C-130 Center Wing Box

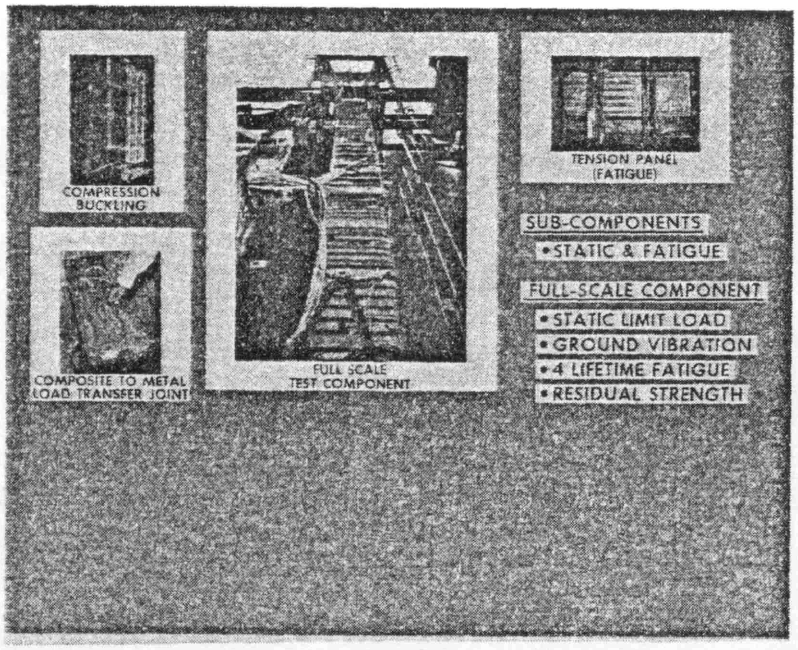

Figure 11.- C-130 Composite Reinforced Center Wing Box Test Program

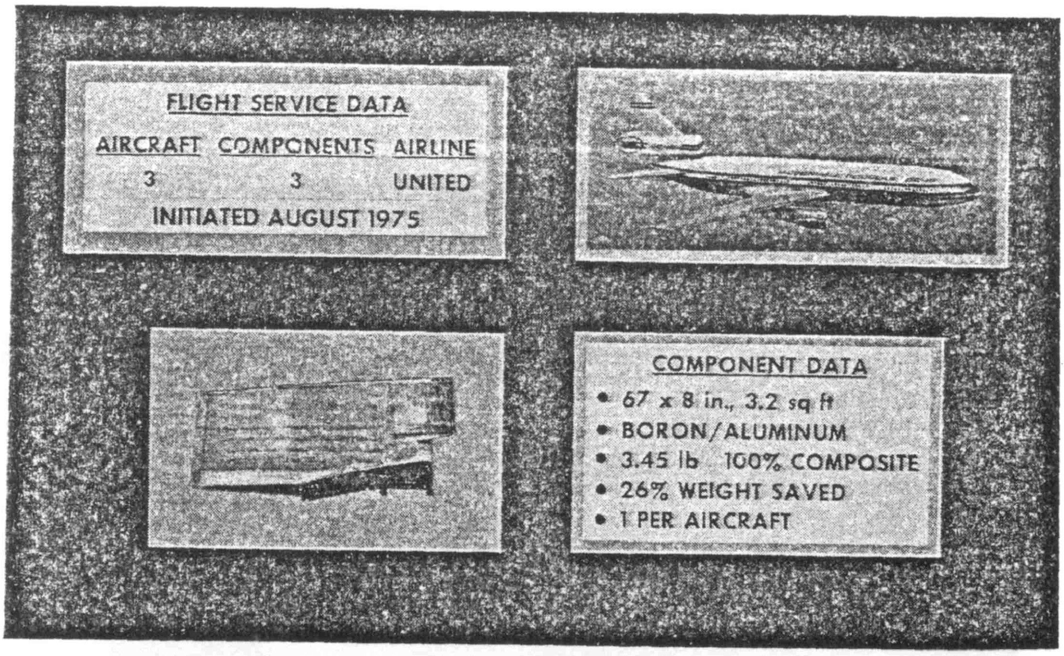

Figure 12.- DC-10 Aft Pylon Skin

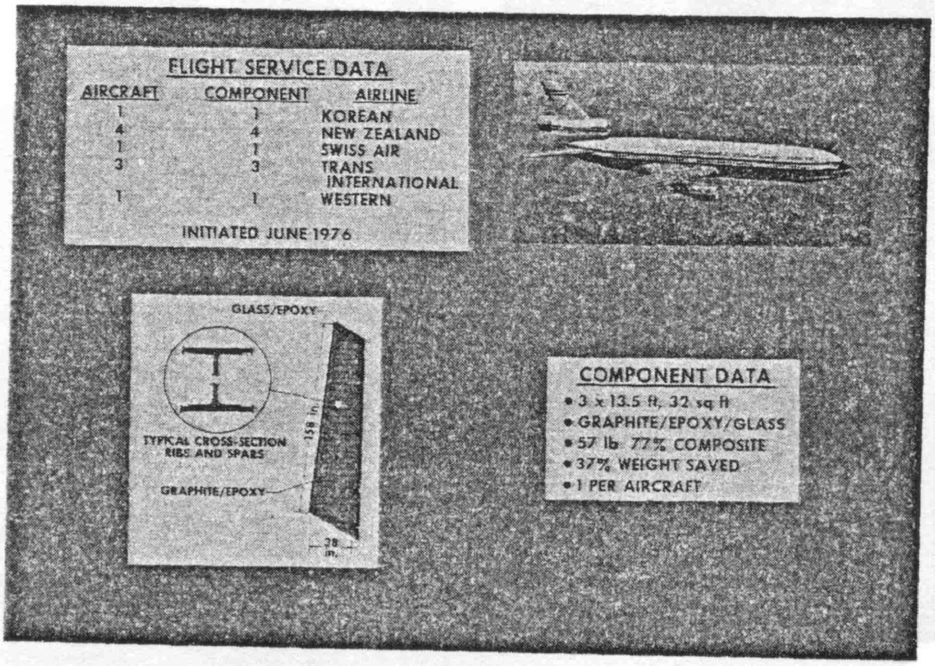

Figure 13.- DC-10 Upper Aft Rudder

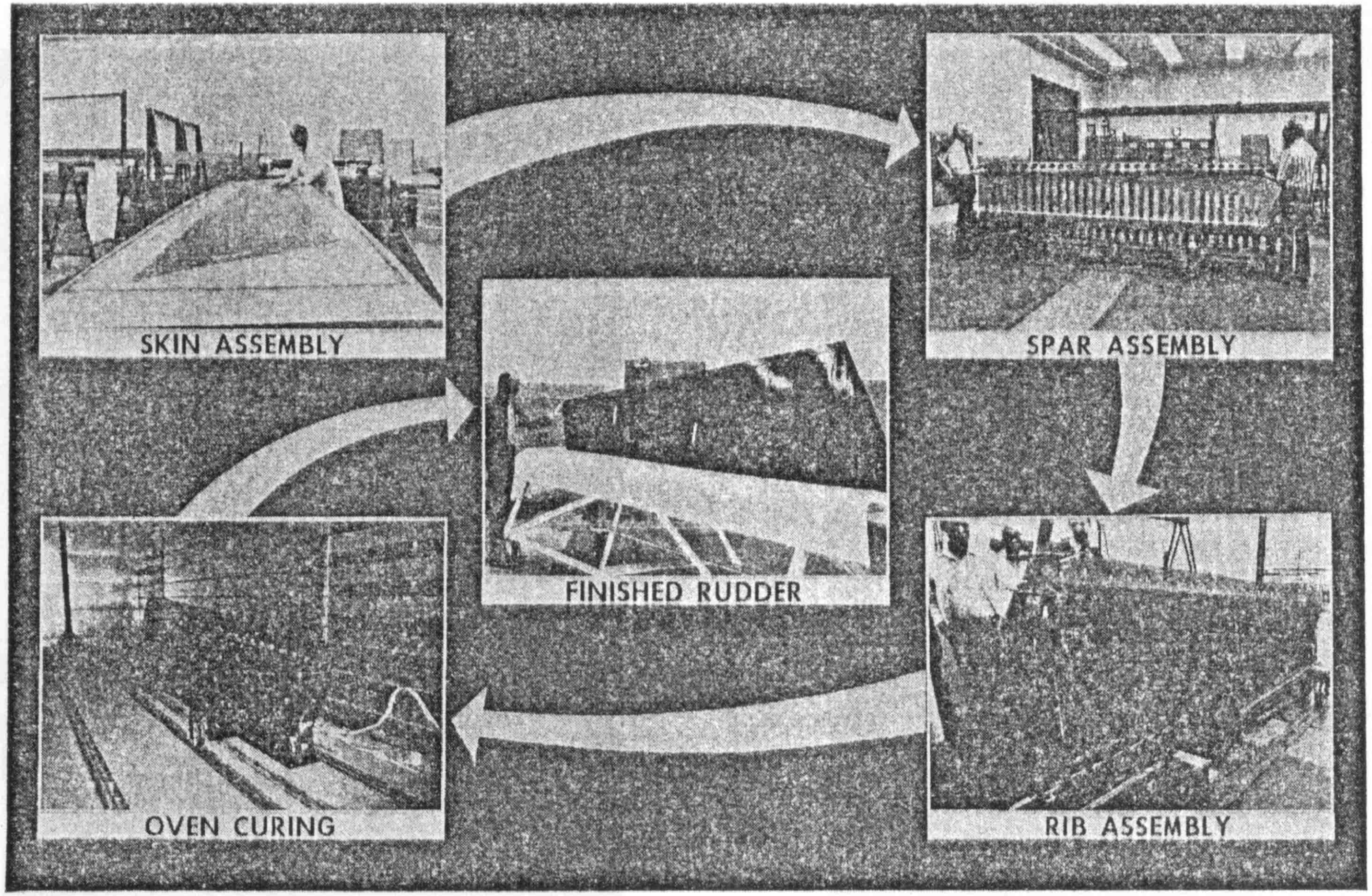

Figure 14.- DC-10 Upper Aft Rudder Manufacturing Sequence

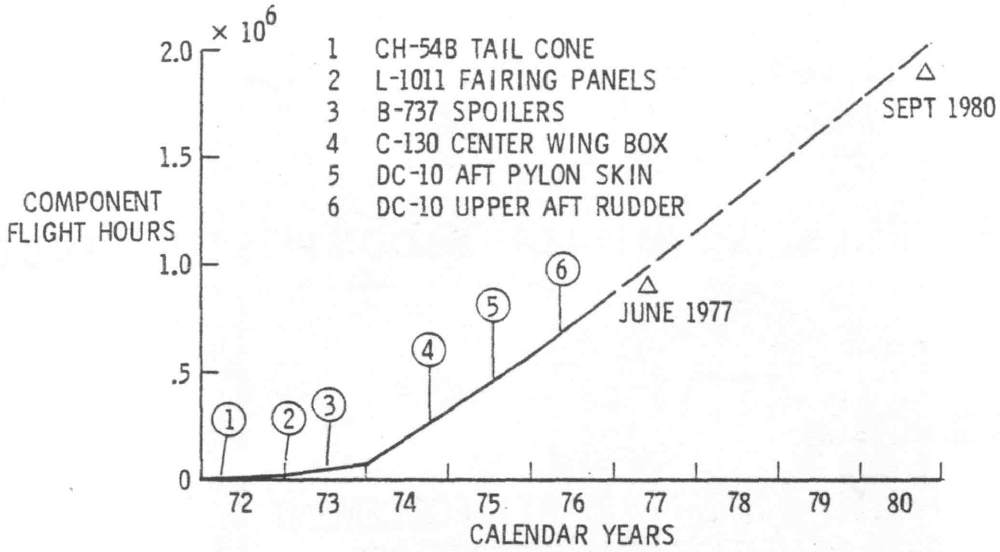

Figure 15.- Cumulative Flight Service Hours,
All NASA Components

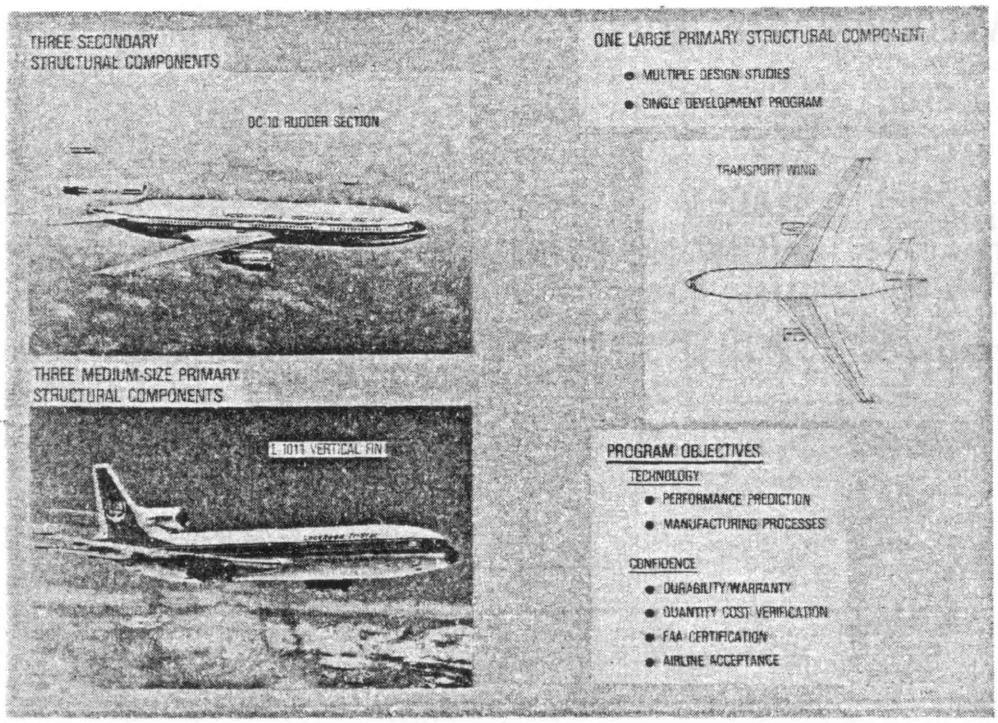

Figure 16.- ACEE Program

REPRESENTATIVE MILITARY COMPOSITE STRUCTURES

PRODUCTION

F-111	WING PIVOT REINFORCEMENT
F-14	HORIZONTAL STABILIZER
F-15	VERTICAL AND HORIZONTAL STABILIZERS SPEED BRAKE
F-16	VERTICAL AND HORIZONTAL STABILIZERS
UTTAS	TAIL ROTOR
AV-8B	WING BOX (PROPOSED)
F-18	WING SKIN (PROPOSED)
B-1	HORIZONTAL STABILIZER (PROPOSED)

DEVELOPMENT PROJECTS

F-4	RUDDERS
C-5A	SLATS
AH-1	BLADES
F-14	OVERWING FAIRING, LANDING GEAR DOORS
A-7	WING TIPS
S-3A	SPOILERS

Figure 17.- Military Programs

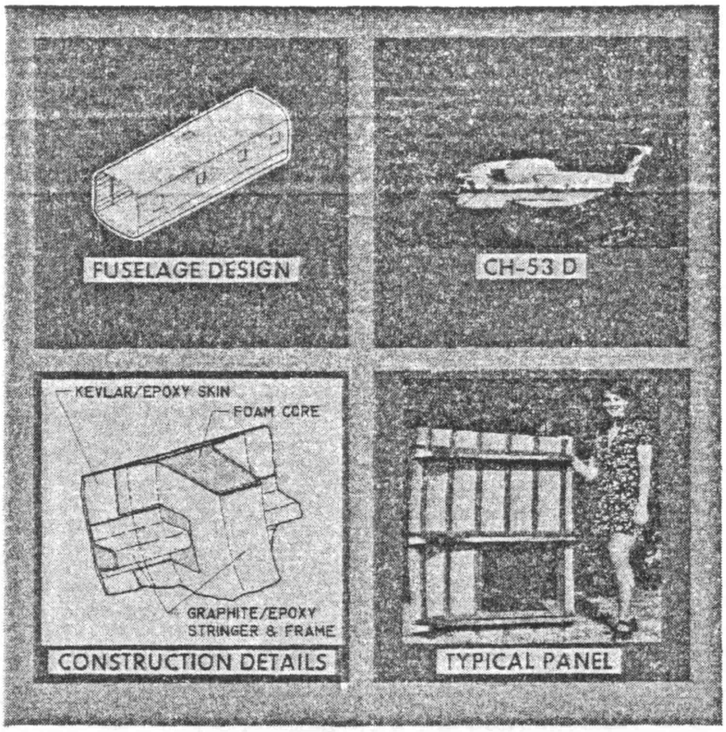

Figure 18.- Composite Structure for Helicopters

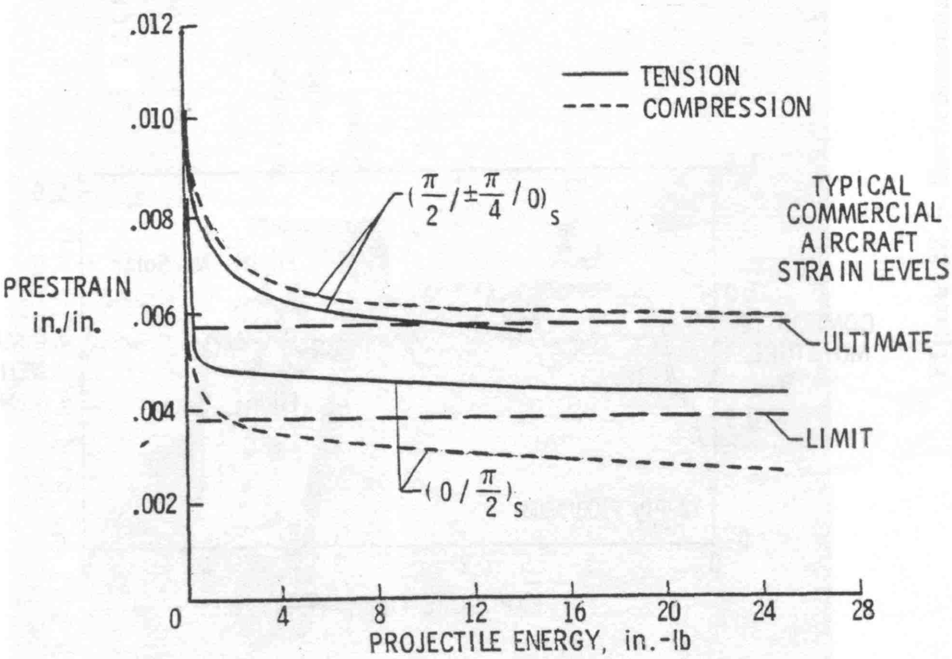

Figure 19.- Failure Threshold Curves for Graphite/Epoxy
Sandwich Test Laminates

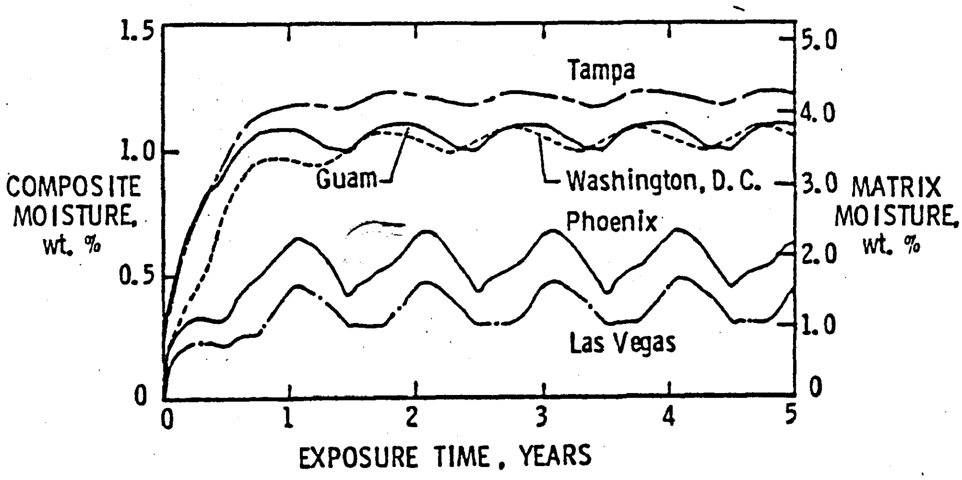

Figure 20.- Calculated Moisture Content for Different Ground Stations (T300/5208 24-Ply Laminate)

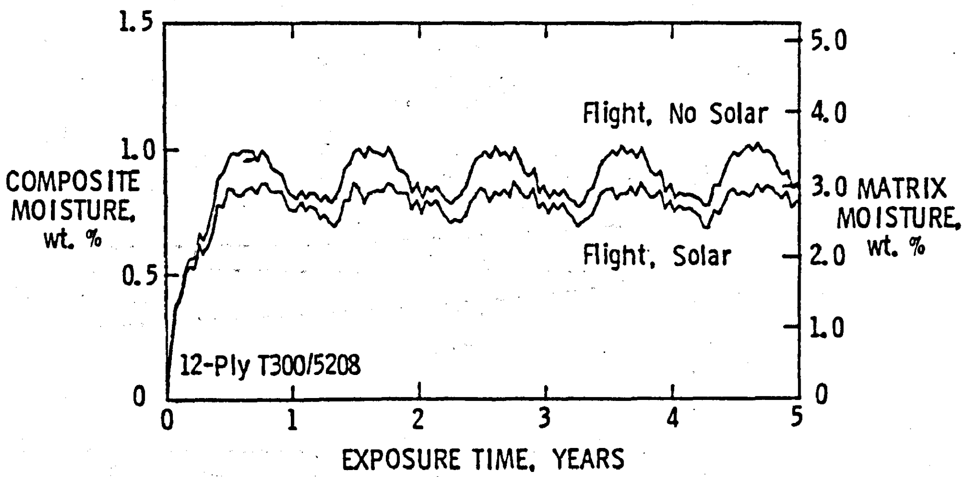

Figure 21.- Effect of Flight Service and Solar Radiation on Moisture Absorption

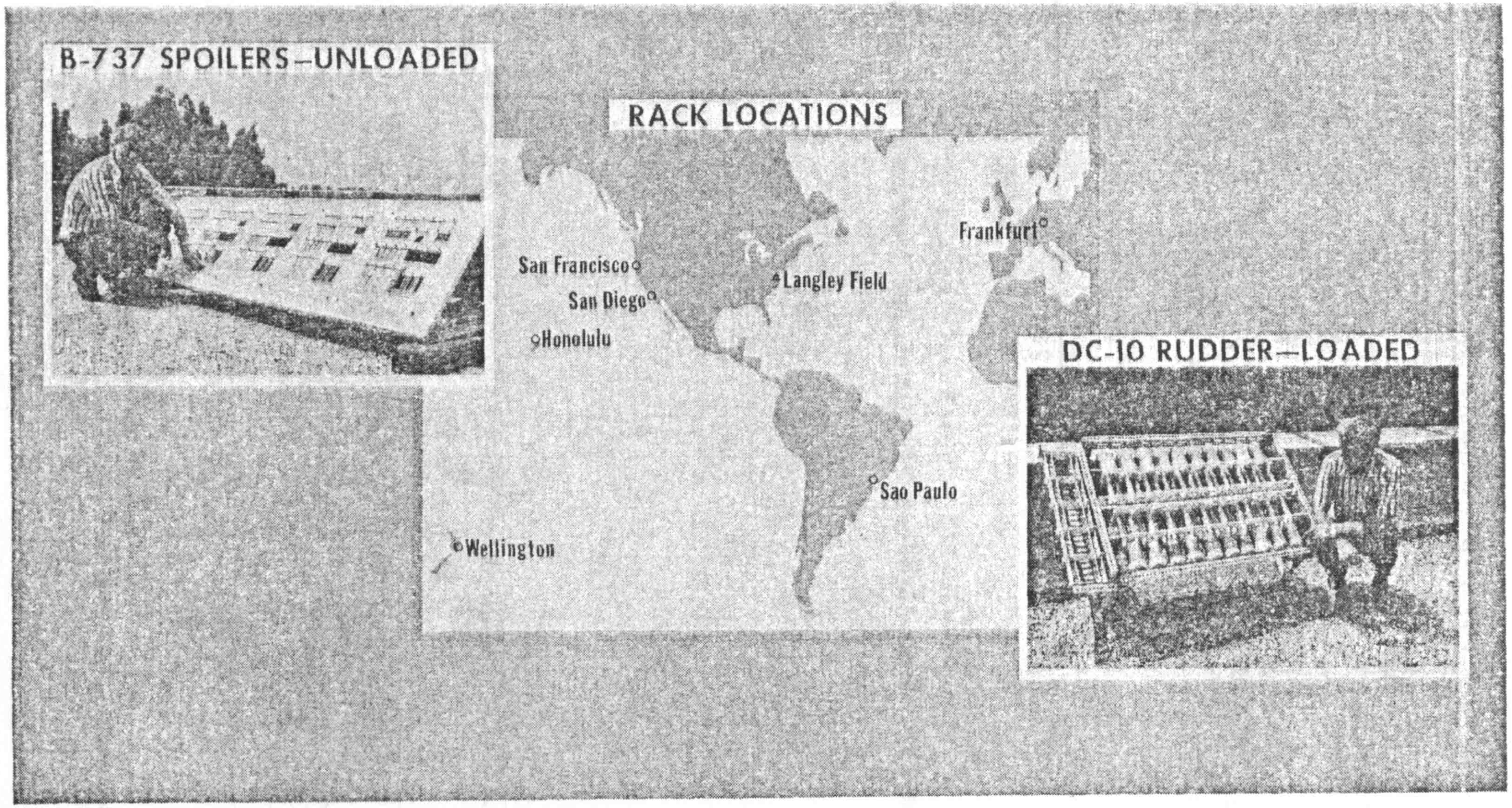

Figure 22.- Environmental Exposure of Composite Materials
Used in Flight Service Programs

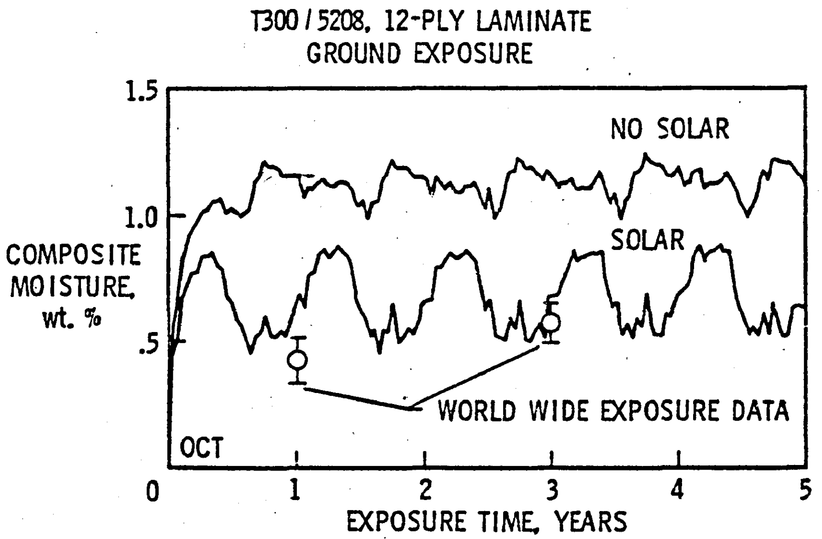

Figure 23. Comparison of Predicted and Measured Moisture Contents

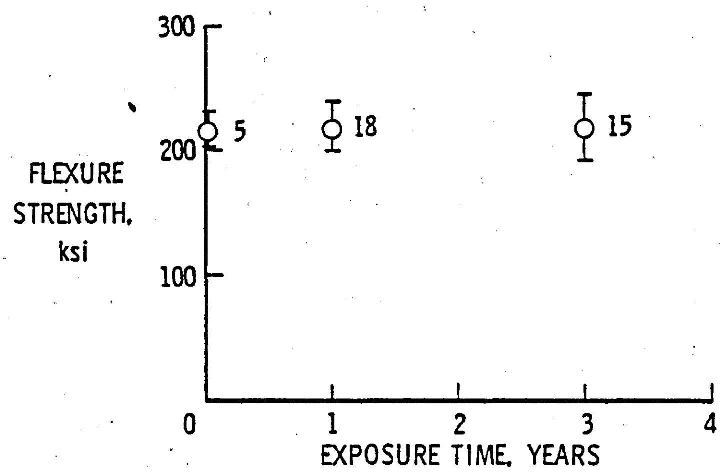

Figure 24.- Test Results After Ground Based Exposure
(T300/5209 Graphite/Epoxy Composite)

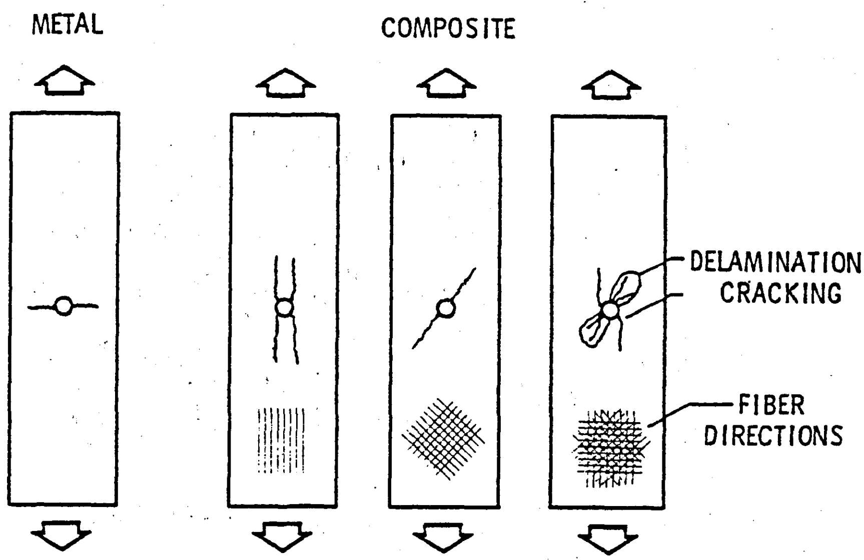

Figure 25.- Fatigue Failure Modes

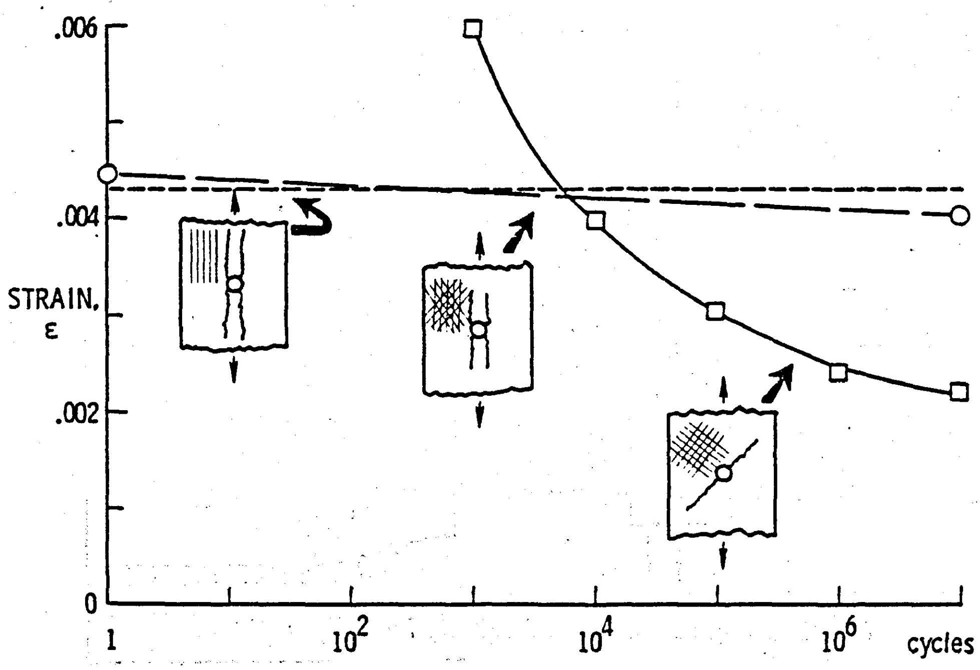

Figure 26.- Effect of Fiber Orientation on Fatigue Behavior

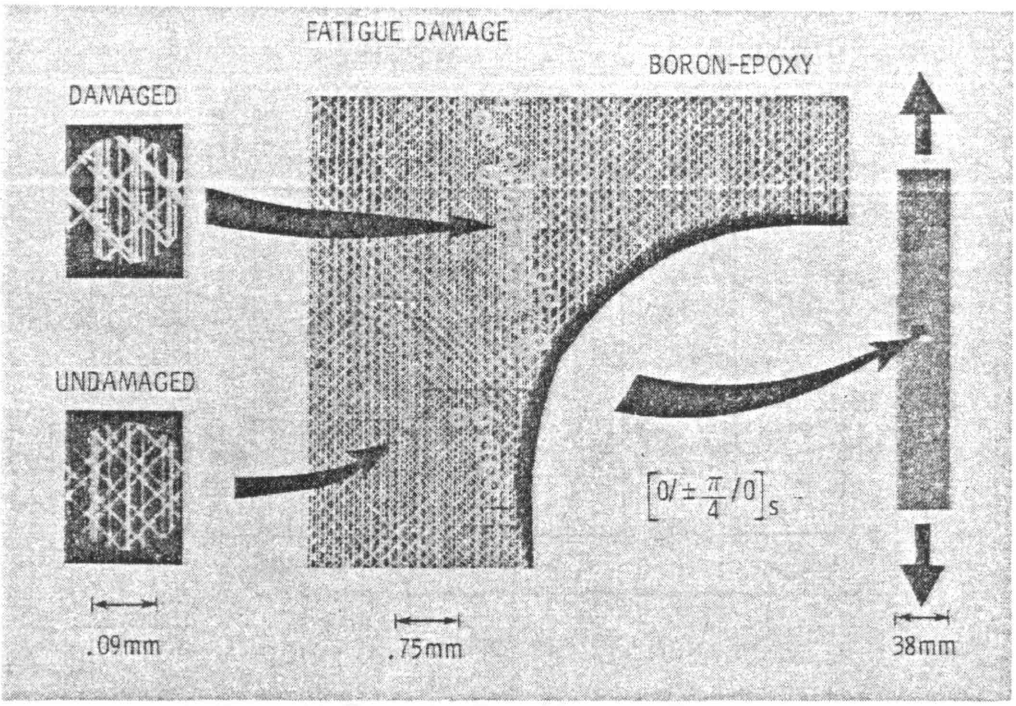

Figure 27.- X-Ray Detection of Fiber Failures by Fatigue

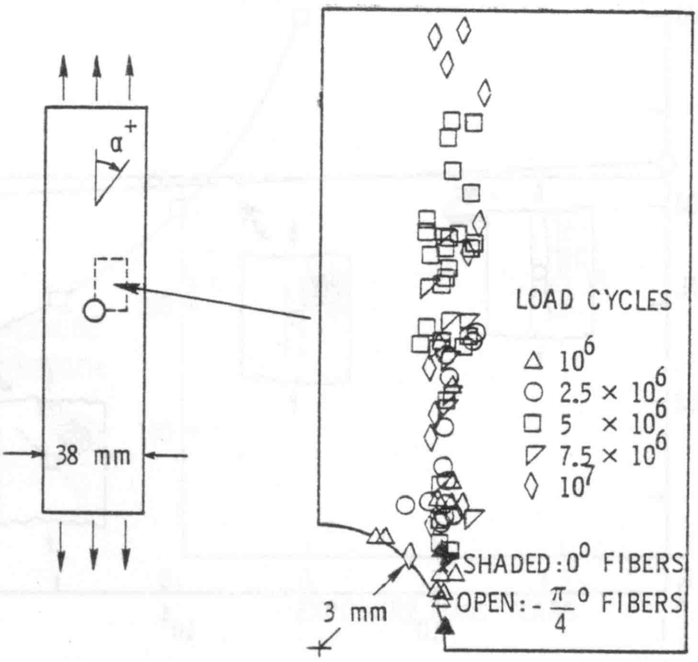

Figure 28.- Fiber Failures in Boron-Epoxy Laminates Under Cyclic Loading $(\frac{\pi}{4}/0/-\frac{\pi}{4}/0)_s$

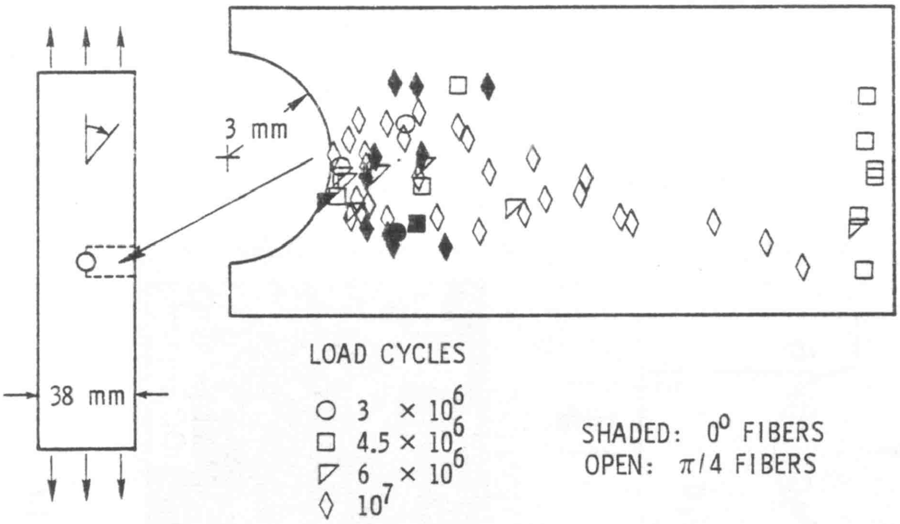

Figure 29.- Fiber Failures in a $(\frac{\pi}{4}/\frac{\pi}{2}/-\frac{\pi}{4}/0)_s$ Laminate Under Cyclic Loads

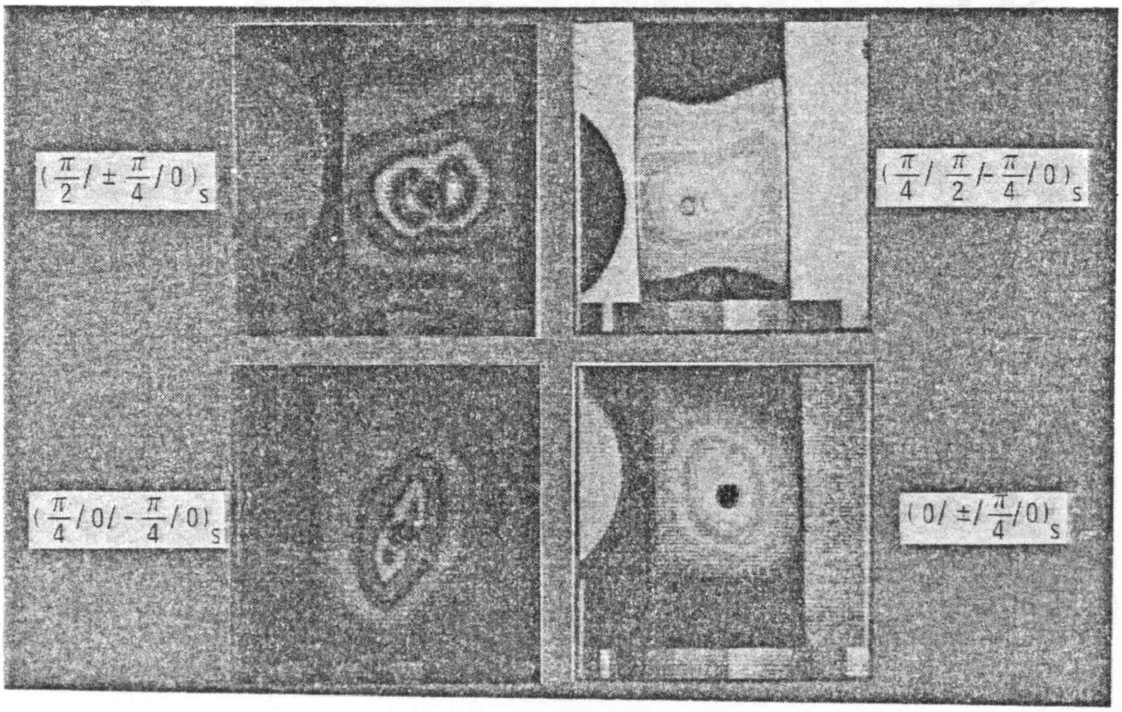

Figure 30.- Heat Emission of Boron-Epoxy Fatigue Specimens (10^7 Tension Load Cycles-Frequency of 10HZ)

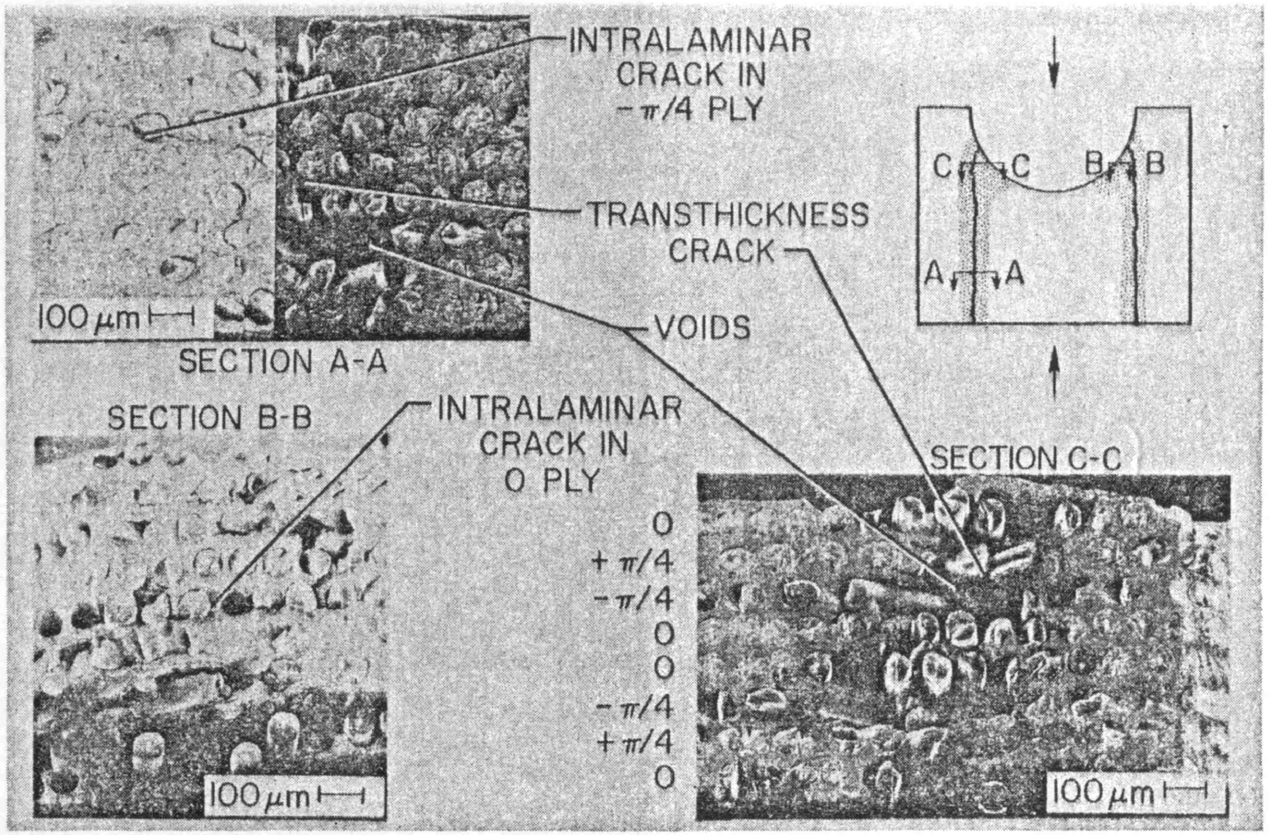

Figure 31.- Matrix Damage in a $(0/\pm\frac{\pi}{4}/0)_s$ Laminate 10^7 Compression Loads