3) A variable share of the market procedure to be input by city-pair. With this modification the airline's current market share could be modified to agree with the flight frequency obtained in the simulation, in proportion to the frequencies of the competition.

4) Inclusion of air delay, especially to fixed-wing aircraft, to more adequately compute true trip times and utilization. Air delay is becoming of increasing importance and realistic data are now becoming available.

Current analyses obtained from using the simulation have produced creditable results but have, as usual, produced additional questions for research in various areas, including a more refined value of time formulation and a better approach to the criterion function after maximum earnings with respect to possible flight removal.

The city-port analysis section has not been fully implemented since the system envisioned for the 1970's is not now in existence. This area of vertiport systems costs is being investigated and will be added to the program when feasible data are premised.[9]

VII. Conclusions

V/STOL operations described by this paper are not in operation today. It is the purpose of the simulation analysis to determine the problem areas that will be encountered and help in finding adequate solutions. If the commercial air transport industry is to continue to grow into the 1970–1980 time period, solutions to the ground and air congestion problems about the major airports must be resolved. The moving of the large-demand short-distance intercity flights away from conventional airports to easily accessible V/STOL ports is a step forward towards the solution. This economic simulation is geared toward determining costs and benefits of introducing V/STOL systems which are the concern of the aircraft manufacturers, airline operators, local and national agencies, and most of all, the air travelers.

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Advanced-Composite-Material Application to Aircraft Structures

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The initiation of flight testing of aircraft structures utilizing advanced composite materials in January 1967 marks a major milestone in the development of this new material concept. Specific developments in the boron fiber-reinforced plastic tape systems leading to this accomplishment will be discussed, together with design, analysis, and fabrication techniques. Composite test results emphasize the need for fiber spacing control and matrix toughness. Data through the 420°F temperature range demonstrate the advantages of this material over other materials. Analytical procedures for predicting laminate stiffness and yield strength have demonstrated satisfactory results with an assumed yield surface theory. A digital program has been written to aid the designer in selecting the minimum-weight laminate for a given point load or stiffness requirement. The flight-test component design features and static test results will be discussed.

I. Introduction

A TEST airplane flew a routine flight-test mission on January 3, 1967 utilizing a component made with an advanced fibrous-reinforced composite. The part used was the left-hand inboard airflow director door, a 1- × 6-ft × $\frac{1}{2}$-in.-thick actuated panel located on the lower surface of the wing. The composite constituents were boron fibers produced by the halide process within an epoxy matrix.

This flight mission, the first in an environmental test program of one year's duration, marks a major milestone in the development of advanced composite materials for aircraft. Under the direction of the Advanced Fibers and Composites Division of the Air Force Materials Laboratory, the Fort Worth Division of General Dynamics Corporation, in conjunction with IIT Research Institute and Texaco Experiment

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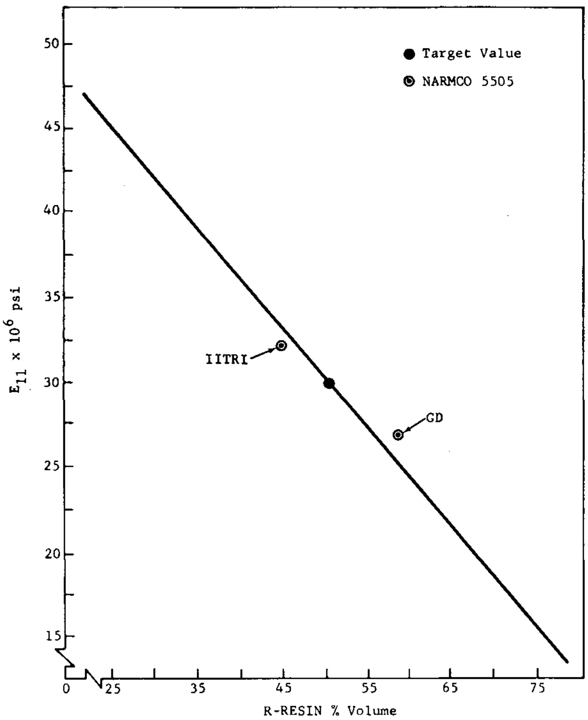


Fig. 1 E_{11} vs percent fiber where $K = 1$.

Inc., is engaged in exploring the application of advanced composite materials to aircraft structure.

In addition to flight testing, this program is composed of many facets. A considerable effort is devoted to the study of analytical methods, material properties, and structural-element testing; however, the principal goal of the project is the design, fabrication, and test of a full-scale, flight-weight, horizontal tail structure. Use of boron fiber-reinforced plastic on this component results in a 300-lb (26%) weight reduction. This weight saving, coupled with projected material and manufacturing cost reduction, indicates that this is a significant cost-effective application.

The rapid advancement of the use of fiber and composite materials, including the successful component demonstration described herein, lends credence to the predicted early application of composites to production aerospace systems. This timetable is acutely dependent on advancements in composite design. It is the purpose of this paper to discuss some of the aspects and implications of the composite technology on design.

II. Composite Properties

The material form with which the designer is provided and on which he is to base his design concepts is preimpregnated unidirectional tape of boron fibers; these are collimated one layer thick with a spacing of approximately 212 fibers/in. The fibers are nominally 4 mils in diameter, which results in an average gap between fibers of 0.7 mil and a ply thickness of 5.3 mils at a resin content of 50 volume percent.

The designer must build up his structure with laminations of this highly orthotropic material. The first requirement he faces is that of designing a laminate that possesses a maximum strength or stiffness (or both) in the principal load direction and adequate strength or stiffness in the secondary load direction. This is accomplished by orienting the various laminations in different directions.

To accomplish this task, the designer must be provided the mechanical properties of the single-ply unidirectional lamina and he must have the analytical tools to obtain the best combination of ply orientations.

Modulus theory, as proposed by Tsai,[1] has been used with some degree of success to predict the ply properties, based on the constituent properties. Two factors in Tsai's approach are of particular interest to the designer and the tape supplier. A "K" factor is found in the expression for properties parallel to the fibers. This factor is intended to compensate for the fact that, in practice, fiber collimation may be less than perfect. Figure 1 shows the relationship between certain selected data and this expression, assuming $K = 1$, or perfect collimation. (It can be seen that this theoretical factor is not a significant consideration in actual practice.)

A contiguity factor "C," which accounts for the irregularity found in fiber spacing, is used in predicting transverse properties. A contiguity factor of zero corresponds to the mathematical lower bound. Figure 2 shows the relationship between transverse modulus and resin content for various C values. It is indicated that this is a very important term which will, in all probability, have to be established for each preimpregnating process or producer. The contiguity factor may become a specification value through a tolerance on transverse modulus.

Two data points are shown in both Fig. 1 and Fig. 2. These data points represent the results from two organizations using the same preimpregnated tape material but different processing and test methods. Processing is being refined to achieve the target resin content and testing procedures for E_{22} are being reviewed.

Composite properties predicted by the foregoing procedures and confirmed by solid-laminate tests of unidirectional laminates may be used for design purposes, providing a means may be found of predicting properties under combined load. Several failure theories have been written for predicting yield and ultimate. For example, St. Venant's maximum strain theory (Ref. 2, p. 449) is applicable to brittle material, and Von Mises' maximum energy of distortion theory (Ref. 2, p. 451) is best suited for ductile materials. However, before these are reviewed for applicability, a complication in their use (due to orthotropy) needs to be discussed. Figure 3 presents graphically the general form of these two theories for an isotropic material. The values on the axis represent the axial tension (or compression) properties. Halfway be-

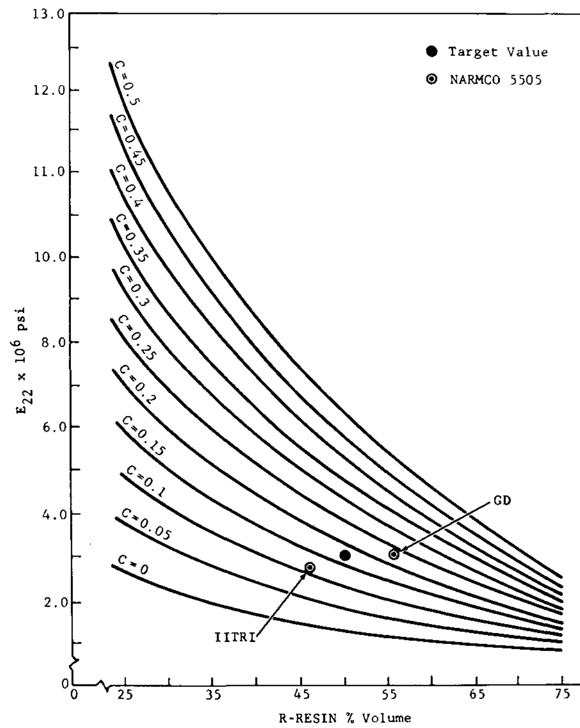


Fig. 2 E_{22} vs percent fiber; C = variable.

tween the tension and compression axes is pure shear, and halfway between the tension-tension (or compression-compression) axes is the biaxial tension (or compression) condition. This graphic format is easily used and understood, since any plane state (σ_x, σ_y, and τ_{xy}) can be resolved into the principal stress (σ_{max} and σ_{min}), where $\tau = 0$.

In the case of a highly orthotropic material such as a single ply of parallel fibers, the axial properties may be conveniently referenced to axes parallel and transverse to the fibers. A stress state σ_x, σ_y, τ_{xy} must then be resolved to the fiber axes 1 and 2, thereby maintaining a shear component. This procedure is represented graphically in Fig. 4; in this surface the 1 and 2 axes represent the properties of the single ply parallel and transverse to the filament and the 3 axis is the shear component.

For a single lamina, the surface is bounded by a plane parallel to the (1, 2) plane but offset a distance of plus or minus τ_{12} on the 3 axis. The failure envelope is a surface, rather than a plane, because the specified allowable stresses are not invariant to a coordinate rotation, as in an isotropic material.

The maximum-strain theory illustrated in Fig. 4 is currently used at General Dynamics; however, it should not be construed that this is the theory best suited to the boron-epoxy composite. In actuality, there is very little information available with which to compare the various failure theories. Tsai[1] suggests a Hill-type[3] surface. Waddoups has selected St. Venant's maximum-strain theory, since it seems to best fit the Scotch ply data reported by 3M.[4] However, a single datum point obtained recently with the Shockey "cross beam" test[5] indicates that the Hill theory is the more applicable. Because of the unusual dependence of composite design on predicted laminate properties, extra emphasis should be placed in immediate studies on obtaining controlled data to determine which failure theory is most applicable to the boron-epoxy composite.

Stress-strain curves for unidirectional lamina along its principal axes and a crossply laminate are shown in Fig. 5. The purpose in presenting these data is to show typical properties for the E-787 resin system in comparison with other resin systems. The DEN 439 system (Fig. 5) dramatically illustrates the dependence of composite properties on the resin system. The DEN 439 system, which uses the 16-hr postcure at 450°F, is believed to be brittle; that is, the resin lacks toughness and possesses insufficient plastic strain capability. DEN 439 was rejected as a candidate high-temperature resin system.

The Narmco 5505 system, also a candidate for high-temperature use, possesses properties comparable to the E-787 room-temperature values. Experience with the Narmco 5505 system to date indicates that it possesses the strength and toughness, as well as batch reproducibility, that warrant widespread application. Advanced-composite matrix requirements are still ill defined; therefore, it is important that an approach of cautious confidence be taken in composite application.

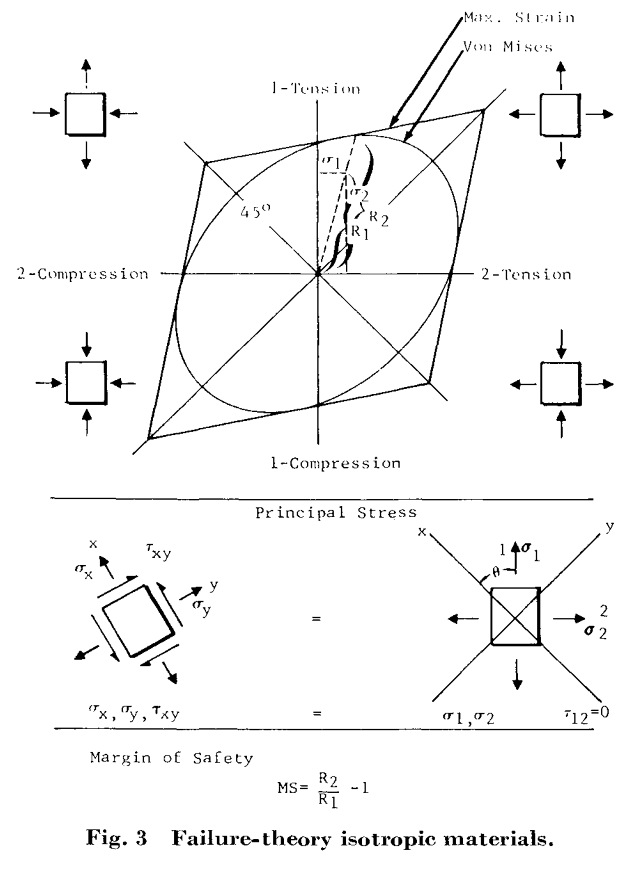


Fig. 3 Failure-theory isotropic materials.

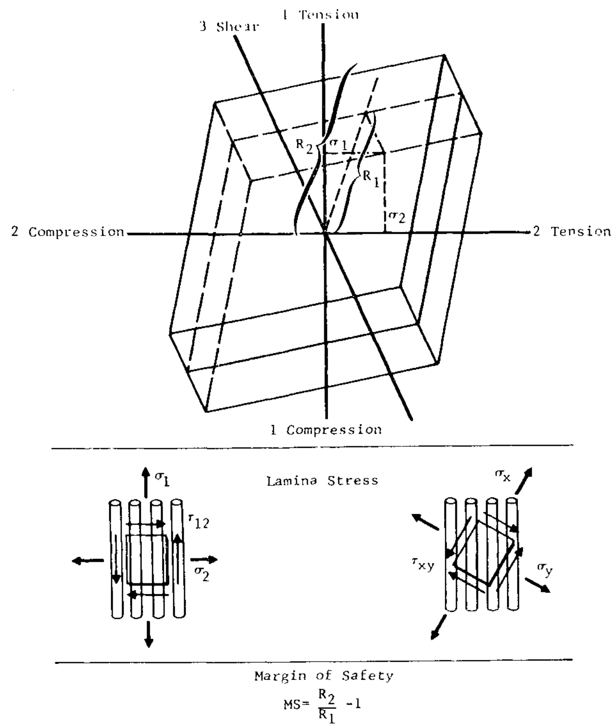
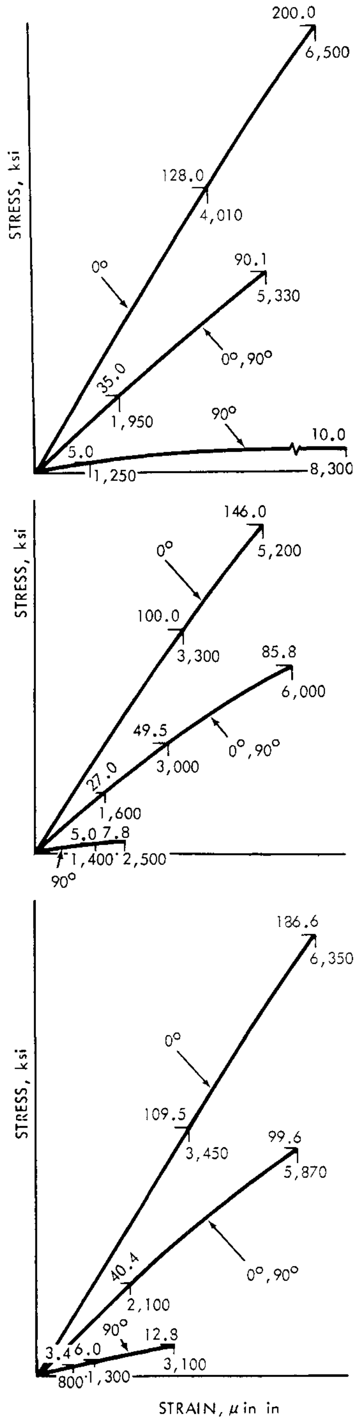


Fig. 4 Failure-theory anisotropic materials.

III. Design Procedures

Characterization of the composite and analytical expressions for its properties are only input data for the generation of design tools. The designer, through study of a part's function, geometry, and external loads, determines the part's internal load distribution. The designer then knows the approximate shear and normal loads for any given point on the structure. In order to select the proper laminate for the determined point load, lamina properties (previously discussed) must be presented in the form of combined stress-interaction curves for the total laminate.

Two approaches of design application are currently being considered. In the first method (Fig. 6) the lamina properties have been related through transformation equations[6] to yield total laminate properties. The designer is provided with interaction curves describing the yield surface for the given laminate orientation. In the example, illustrated for a laminate of any thickness, the application is symmetrical about the centerline, with 40% of the plies at ±45° and 60% at 0°. (It may be noted that the lamina relationship is a surface similar to that shown in Fig. 4, except that the total laminate results in a more complex surface, since σ_x and σ_y do interact with τ_{xy}.) The designer may use this method to select a laminate thickness for his point requirement. To



a) **Narmco Materials-E787**

b) **Narmco Materials-Den 439**

c) **Narmco Materials-5505**

Fig. 5 Stress-strain relationships.

select the best orientation, he must repeat this process for many different orientations to determine which one will yield the minimum laminate thickness.

A second procedure is being developed whereby the computer is employed to "tailor" the laminate to the point loads (Ref. 7, p. 3). This procedure uses the same laminate input data and transformation equations as the first method. The lamina is constructed one ply at a time; each ply is added at the orientation that gives the maximum strength, as determined by reduction in distance to the yield or ultimate surface. Each time a ply is added, the previously selected ply orientations are perturbed in 5° increments for 30° from the original orientations. This procedure is completed for four plies; the next four are added individually without the perturbation cycle. When the laminate reaches a thickness of eight plies, a check is made to determine if more desirable results can be achieved by repeating the 4-ply configuration. This process continues until a positive margin of safety is achieved for all load conditions. This technique provides a great deal of flexibility for the designer because input data may be more readily changed and, when desired, laminate optimization may be directed toward stiffness, as well as strength.

A sample problem and its computed results are summarized in Fig. 7; the problem illustrated is a case of pure shear at a relatively low load level. The optimum laminate thus determined is a 3-ply laminate that appears to be efficient, but, on the contrary, is directional. The shear stress provided in the positive direction is higher than expected; yet, in the negative direction, it is low. A review of the problem after the fact indicates that this effect is obvious. However, the fact that it was not anticipated serves as a caution to designers using such programs to select the point load conditions.

An additional program by Petit (Ref. 7, pp. 7–10) is available for use in selecting the optimum laminate for a panel buckling application. This program determines the maximum compressive and shear-buckling coefficients for various orientation angles. Additional programs have been written by Wheeler,[8] Hackman,[9,10] and others.

The aforementioned procedures will provide adequate tools for the designer to select the structural concept, laminate thickness, ply orientation, and, in some cases, constituent materials. However, it is the designer's responsibility not only to understand his material and the design tools provided him, but also to solve the practical problem of detail design, function, and manufacturing.

IV. Component Development

The designer learns primarily from experience; however, the field of advanced composites is young, and, consequently, there are few case histories to review. In an attempt to gain valuable information in this area, General Dynamics Fort Worth Division has recently designed and built two aircraft components, in addition to the flight article previously men-

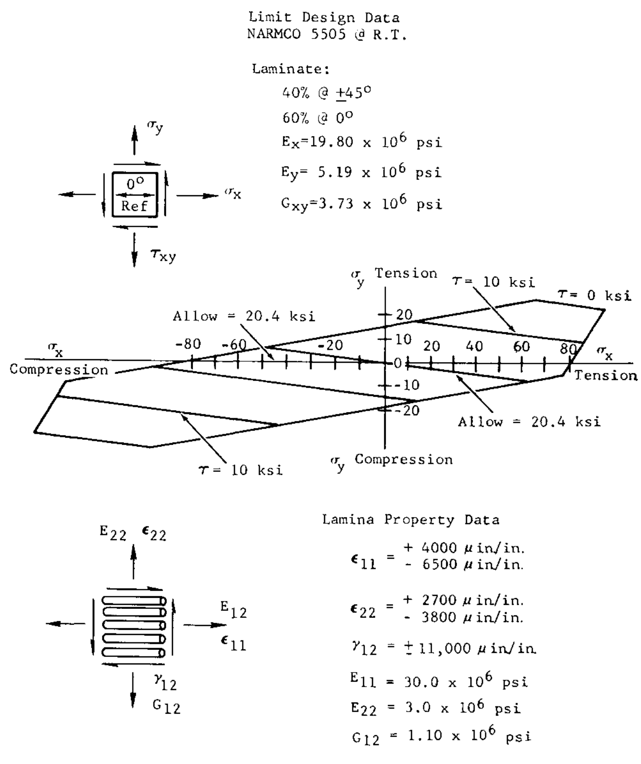


Fig. 6 Laminate yield surface.

tioned. These will be discussed briefly to document certain distinct design features.

The airflow director door, Fig. 8, is a flat rectangular panel supported by a continuous hinge along the forward side and by an actuator attachment at the geometric center. The present construction is aluminum sandwich with a one-piece machined aluminum edge member. The actuator fitting attachment point is solid aluminum and is an integral part of

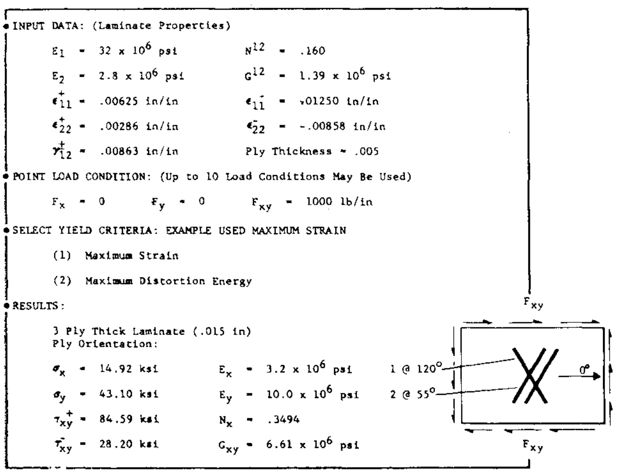


Fig. 7 Laminate optimization program.

the edgemember. Two loading conditions were considered: 1) 3.38-psi limit pressure when supported by the hinge and actuator, and 2) 3.65 psi when supported by the hinge and aft edge. A maximum allowable twist of 14° was derived from interference criteria.

The redesign of this door in boron-epoxy composite material retained the sandwich design with the one-piece aluminum edge member. The aluminum skins were replaced and a 4-ply composite skin oriented at ±25° and ±65° with the reference axis parallel to the hinge axis. The hard point for the fitting was constructed of solid fiberglass to provide thermal stress compatibility.

The most significant problem of this design was the residual stress between the aluminum slug and the composite skin. The relative coefficients of thermal expansion are 13.0 in./in./°F for the aluminum and 6.5 in./in./°F for the composite. The manufactured part was void-free and of good quality with the exception of slight warpage. This warpage was reduced to acceptable limit when installed. Static test of the part demonstrated the design stiffness and 10% positive margin on strength. Failure occurred in core shear slightly removed from the fiberglass insert, and this propagated a delamination in the skin. It is concluded that aluminum parts may be used in conjunction with boron-epoxy composites providing the residual stress is anticipated.

The second design panel (Fig. 9) is a fixed component on the upper surface of the wing, aft of the structural box, forward of the flap, and directly over the airflow director door. The present panel, approximately 6 ft long and 3 ft wide, is an aluminum sandwich wedge supported by two fitttings at the forward edge and attached at the forward, outboard, and inboard sides. There is no support on the aft side.

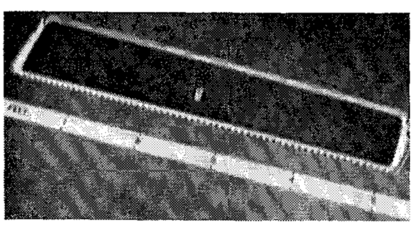
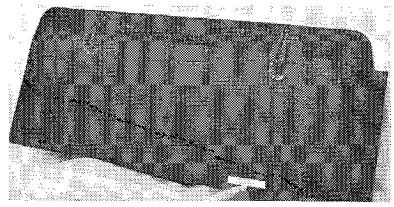


Fig. 8 Airflow director door.

Fig. 9 Wing trailing edge panel.



The boron-composite replacement design consisted of the same structural arrangement, using either boron composite or glass composite except for the aluminum honeycomb core and titanium inserts in the fastener area. The skins consist of a 3-ply laminate with ±30°, 90° orientations with the reference axis parallel to the forward edge. The skin is built up to a total of 14 plies near the actuator. The build-up is a tapered 90° lay-up that transfers the load from the fittings into the part. There were two basic problems to be solved in this test. One was whether a minimum-gage skin, i.e., 3-ply, would prove manageable in manufacturing, and whether it would perform structurally as predicted in spite of complete disregard for symmetry lay-up. The second problem was the transition of the complex load from the thin skin to the build-up. To compensate for these problems, the part was conservatively designed in the build-up area.

Fabrication of the part proved satisfactory, although the thin skins curled considerably when removed from the curing tool. One skin was rejected because excessive bleeding of resin occurred during cure, resulting in a resin-starved laminate. The bleeder material was reduced from 0.020 to 0.005 in., with no further difficulties encountered.

The loading requirement was a uniform negative pressure of 3.0 psi at limit. Static testing demonstrated the design integrity. Loading was stopped at 145% of ultimate load because of test jig strength limitations. Small skin failures occurred at the corners, where the tiedown fasteners caused excessive bending of the thin edge. It is apparent that the conservative approach used in the build-up area was unnecessary. However, the flight-test part will not be redesigned to eliminate this conservatism, since the purpose of the test is to obtain service experience rather than to achieve maximum weight reduction. This composite design established the structural integrity of a 3-ply (0.015 in.) skin for a sandwich panel.

The third component (Fig. 10) is the main-landing-gear aft door. The present part, an aluminum sandwich construction, is approximately 4 ft wide, curved to the lower fuselage contour, and is 3 ft long. Support is provided by two fittings on the forward edge of the door. The critical load occurs when the forward door is operating as a speed brake, causing severe turbulence and buffeting in the vicinity of the main-landing-gear aft door.

a) Inner skin

b) Outer skin

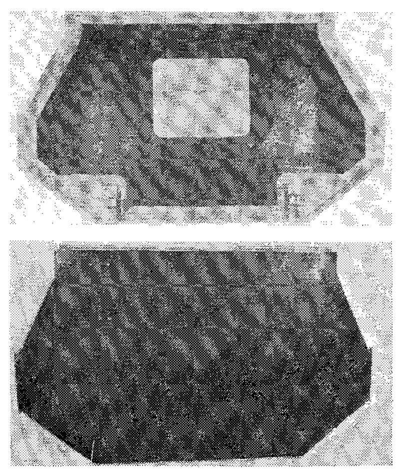


Fig. 10 Aft main-landing-gear door.

The redesign in boron fiber is similar in construction, except the aluminum skins are replaced with boron-epoxy composite skins. The loads in the skins are redundant and variable. This composite door provides the challenge of obtaining structural integrity and demonstrates an application to contoured surfaces. A solid boron-epoxy skin covers the entire surface of the component.

This component has been successfully manufactured to tolerance, and a proof test has been completed, qualifying the door for flight. The appearance of the boron-epoxy laminate is excellent. The surface is smooth and uniform in color, and the fiber alignment appears to be nearly perfect. This may be due, in part, to the addition of a preimpregnated type 108 glass cloth on the outside surface of the laminate. The glass ply, cured with the laminate, adds approximately 3-mils thickness to the skins, and its purpose is to protect the boron fiber.

V. Summary

The milestones by which the development of a new structural material is measured are being achieved at a rapid rate and with exceedingly healthy regularity. Development from the first fiber to the first flight component has been achieved in less than a decade.

In addition to the rate of development, composite technology also shares the broad scientific base that has been common to other major aerospace advances. A broad technology base is being established for each step in the development of the fiber, the matrix, and the component. From a purely material standpoint, the scientific disciplines of mathematics and mechanics, coupled with today's computer technology, have established the direction for composite development, as opposed to the empirical techniques of the past. This analytical base at the material level offers a natural extension to component design and analysis, further strengthening the base from which system applications will be planned.

The composite-design engineer of today has at his disposal better and more sophisticated tools for certain of his tasks than his metal-technology counterpart. It is true that the inventory of such procedures is far from complete and is almost totally unproven; however, the limited experience to date supports the optimism through which this work is being pursued.

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