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USAAVLABS TECHNICAL REPORT 65-15

STRENGTH PROPERTIES AND RELATIONSHIPS ASSOCIATED WITH VARIOUS TYPES OF FIBERGLASS-REINFORCED FACING SANDWICH STRUCTURE

Final Report

By

Gene M. Nordby

W. C. Crisman

August 1965

U. S. ARMY AVIATION MATERIEL LABORATORIES

FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-893(T)

UNIVERSITY OF OKLAHOMA RESEARCH INSTITUTE

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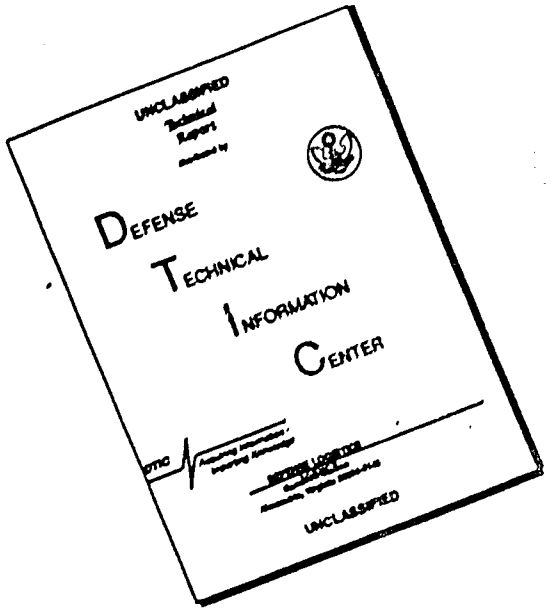

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This work was performed under Contract DA 44-177-AMC-893(T) with the University of Oklahoma Research Institute during the period 25 September 1962 to 31 July 1964.

The information contained in this report is the result of research in the fabrication and evaluation of sandwich panels utilizing fiber-reinforced plastic facings with honeycomb cores.

This work is presented in the hopes of aiding workers in the same field and to contribute to those who need design points in the use of this type of composite for structural application.

Publication of this report by this Command does not necessarily imply endorsement of the test data and results contained herein; the report is published only to make the information available.

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Final Report

By

Gene M. Nordby
W. C. Crisman

Prepared by
University of Oklahoma Research Institute
Norman, Oklahoma

For
U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

ABSTRACT

Contributions to the acceptance of fiberglass-reinforced plastic (FRP) as an aircraft structural material were made through verification of existing theoretical strength relationships by the fabrication and testing of sandwich panels in the laboratory. The four basic failure modes were investigated for sandwich plates and plate columns loaded in edgewise compression. These were general buckling, face wrinkling, shear crimping and face dimpling. To achieve these modes, it was necessary to vary not only the specimen size and boundary conditions but also, in many cases, the dimensions and composition of the constituent materials.

In the development of a suitable structural sandwich, a number of advances were made in the realm of fabrication. These include the development of a multi-ply pre-preg, the establishment of a precure phase in the resin cure cycle as a control of resin flow, and the use of the separately-bonded type of sandwich construction. The effect of adhesive filleting on the core strength and the effect of laminate thickness on facing strength properties were also isolated.

Of the general buckling tests performed, the highest degree of precision was achieved in the tests involving the hinged boundary condition. It was found that the theoretical analysis was conservative for most of the cases investigated. The face wrinkling tests revealed that the symmetrical wrinkle would not always occur in sandwich constructions utilizing honeycomb cores as suggested by the theory. A greater failure stress was generally realized when the load was applied parallel to the core ribbon direction than when applied perpendicular. The limited number of comparisons made showed a greater accuracy in predicting failure stress than for the general buckling mode of failure.

The limited study of shear crimping showed that such failure will not be a problem for honeycomb-core sandwich except for thin panels employing cores of very low shear modulus. The tests on intracellular buckling indicate that this mode will not be important for core cell sizes less than 1/2 inch in combination with 3-ply, or thinner, facings; however, a more thorough theoretical analysis is needed for the intracellular buckling mode.

PREFACE

This report was prepared by the University of Oklahoma Research Institute under U. S. Army Aviation Materiel Laboratories (USAAVLABS)* contract DA 44 177-AMC-893(T). The report contains the test results, conclusions, and recommendations for research conducted on strength properties and relationships associated with various types of fiberglass reinforced facing sandwich structure during the period April 5, 1962, to May 31, 1964.

The research program was directed by Dr. Gene M. Nordby, Dean of the College of Engineering, and Professor L. A. Comp, Professor of Aerospace Engineering, at the University of Oklahoma. Mr. Joseph V. Noyes and Mr. W. C. Crisman were the principal research engineers. The Research staff consisted of Mr. B. J. Harris, research engineer; Mr. Donald Hanson, statistician; and Mr. Terrell B. Warren, test engineer.

The University of Oklahoma Research Institute (OURI) expresses appreciation to the Shell Chemical Company for its advice and assistance pertaining to the use of EPON 828-Z resin, and to Hexcel Products, Inc. for special center cuts of its large-cell paper core material.

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*Formerly, U. S. Army Transportation Research Command (USATRECOM).

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SYMBOLS

a,b	dimensions of the panel with the sides b parallel to the line of action of the compressive load
t_{F1}, t_{F2}	thickness of the two facings
t_c	thickness of the core
t	total thickness of the panel
E_{Fa}, E_{Fb}	moduli of elasticity of the facings in the a and b directions
μ_{Fab}	Poisson's ratio of the facings associated with the contraction in the a direction and extension in the b direction due to a tensile stress in the b direction
λ_F	unity minus the product of the two Poisson's ratios of the facing material associated with the directions a and b
E_{Ca}, E_{Cb}	modulus of elasticity of the core perpendicular to the flutes (parallel to the facings of the sandwich) in the a and b directions, respectively
E_{Cz}	modulus of elasticity of the core in the direction parallel to the flutes (perpendicular to the facings of the sandwich)
G_{Caz}, G_{Cbz}	shear modulus of the core associated with the axis perpendicular to the face of the panel (z) and the axis parallel to the edges of lengths a and b respectively
G_{Fab}	shear modulus of the facings associated with the axes parallel to the edges of lengths a and b
μ_{Cbz}	Poisson's ratio of the core associated with the strains in the b direction and z direction due to a stress in the z direction
f_{Fcr}	critical buckling stress of the facings
P_{cr}	the buckling load per inch of loaded edge
P_{crs}	the buckling load per inch of edge corrected for the effect of shear deformation of the core
A_o	ratio of the facing wave amplitude to half-wave length at no load (initial waviness)

L	half-wave length of facing wrinkles
R_1	nominal cell size (radius)
R_2	measured radius of the core cell inscribed circle
E_r	reduced modulus of elasticity of the facings in the direction of the load
E_t	tangent modulus of elasticity of the facings in the direction of the load
(+R), (=R)	these are used with other symbols to indicate the load is oriented perpendicular or parallel to the core ribbon direction, respectively
(+W), (=W)	these are used with other symbols to indicate the load is oriented perpendicular or parallel to the fiberglass fabric warp direction, respectively

DISCUSSION

OBJECTIVES

A. Introduction

The Army's expanding V/STOL program is placing ever increasing demands on aircraft structures, not only on the design concept but on the structural material itself. For these future generations of aircraft, the structural material must provide a smooth aerodynamic surface for efficient high-speed flight at low altitudes in high density air; it must have high resistance to impact damage that could be produced from sand and gravel set in motion by downwash impingement on unimproved landing areas; it must be corrosion resistant and be easily maintained and repaired; and, most importantly, it must have a high strength-weight ratio. At the present, nonmetalic composite materials stand out as those most able to meet these criteria.

Though present state-of-the-art developements in resin, bonding systems, and fabrication techniques allow construction of composites, as yet, suitable data for design and analysis are not available. Before the composite can be accepted as a primary structural element, it is necessary that clear-cut strength relationships be established. The goal of the research program presented in this report was to contribute to the verification of existing theoretical strength relationships for the very promising structural sandwich employing honeycomb cores and thin facings of epoxy-fiberglass laminates by actual tests performed in the laboratory.

The four basic failure modes were investigated for sandwich plates and plate columns loaded in edgewise compression. These were general buckling, face wrinkling, shear crimping, and face dimpling. Figure 1 illustrates the types of failure. To achieve these modes, it was necessary to vary not only the specimen size but also, in many cases, the dimensions and composition of the constituent materials.

B. Program Analysis and Design

Since existing technical literature is basic to the accurate refinement, modification, or validation of current strength relationships, the first step placed in the design of the research program was the reviewing of pertinent literature. The current strength equations would be used to aid in selecting the initial structural parameters and functional variables.

Because of the exploratory nature of the program, which encompassed fabrication as well as specimen configuration and boundary conditions, the sequential technique of investigation was chosen as the means of achieving the objectives. Thus, each new step in the research would be guided by the previous findings.

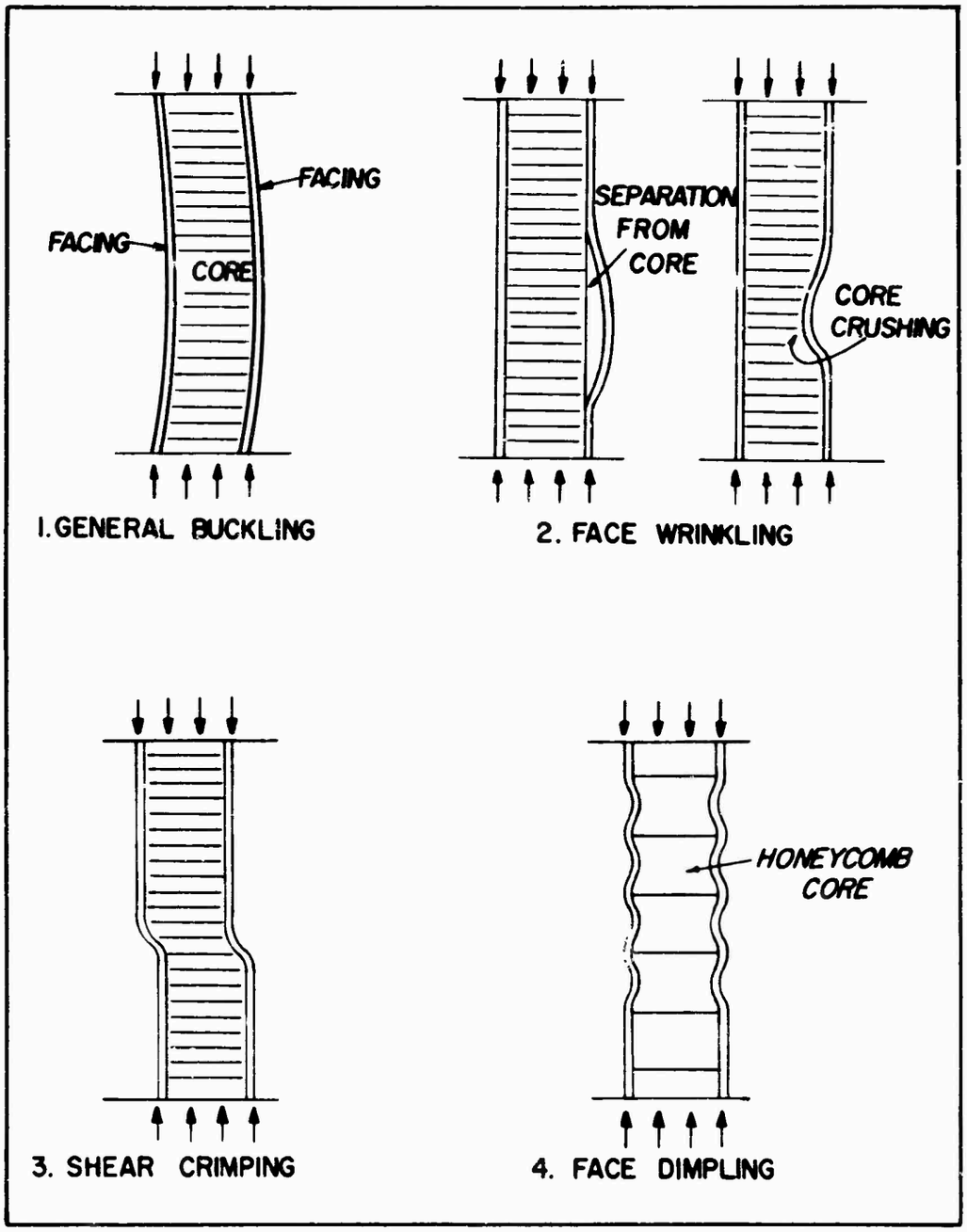

Figure 1. Modes of Failure of Sandwich Construction Under Edgewise Compressive Loads.

It was planned that the primary research, verification of existing strength relationships, would be prefaced with a short fabrication optimization. However, initial experience in fabrication pointed out the existence of just as large a void in this realm as in that of proven panel-strength relationships. Consequently, it was necessary to establish a separate program to develop the needed information on the relation between the fabrication process variables and the final strength properties of the material, while permitting the sandwich-panel strength program to continue. This information was previously reported in USATRECOM Technical Report 64-37, "Research in the Field of Fiberglass-Reinforced Sandwich for Airframe Use," July 1964 (reference 1).

Even more basic than the problem of what levels of the process variables should be used was that of the actual impregnation of the fiberglass fabric and the laminating of the facing. It was discovered that the hand method of impregnation so often used in industry, whereby the resin is worked into the fabric with squeegees, was not adequate to produce void-free reproducible resin distributions consistently. Therefore, a mechanical means of coating the fabric had to be devised before the panel strength study could be accomplished. An extension to the contract was granted, and plans were made for the design and construction of a multi-ply coating machine.

Initially, it was intended that the sandwich be constructed by the single-step method; however, the fabrication was soon shifted entirely to the bonded-type of sandwich in which the facings are prelaminated and bonded to the core in a separate step. This change was made because the separately bonded-type sandwich gave higher and more consistent strength values and because the initial flatness of the facings was found to be essential to preventing premature failure of the specimens.

The test program was planned to consist of two main areas of concern, the tests associated with the panel failure modes and the tests for the facing and core material properties to support the analytical calculations. For conservation of time and funds, the number of supporting tests was held to a minimum; hence, where possible, these tests were used simply to monitor and confirm the published material properties. Consequently, the particular material properties monitored, where necessary, were core flatwise compressive modulus and strength, core plate shear modulus and strength, core-to-facing bond strength, and facing compressive modulus and strength.

All tests were conducted at room temperature, and standard procedures were used where possible. The precision test fixtures necessary to achieve special boundary conditions were designed and built as needed. In addition, a high-pressure hydraulic press was constructed to complement the low-pressure vacuum press initially available.

FABRICATION AND TEST EQUIPMENT

The special equipment used in the fabrication of the FRP materials consisted of a multi-ply coating machine and two laminating presses, one hydraulically operated and the other of the vacuum blanket type. This equipment was developed by the University of Oklahoma Research Institute staff and is described in reference 1. Figures 2, 3, and 4 show the essential details.

Several fixtures to support the laminate and sandwich specimens during edgewise compression loading were designed and constructed for this research program. These are detailed in Figures 5, 6, and 7.

The testing of specimens was conducted on one of three testing machines: a 10,000-pound-capacity Instron, a 100,000-pound-capacity Baldwin, and a 200,000-pound-capacity Tinius Olsen balance-beam testing machine. A number of other commercially available machines were used in the program and are mentioned in the body of the report when pertinent.

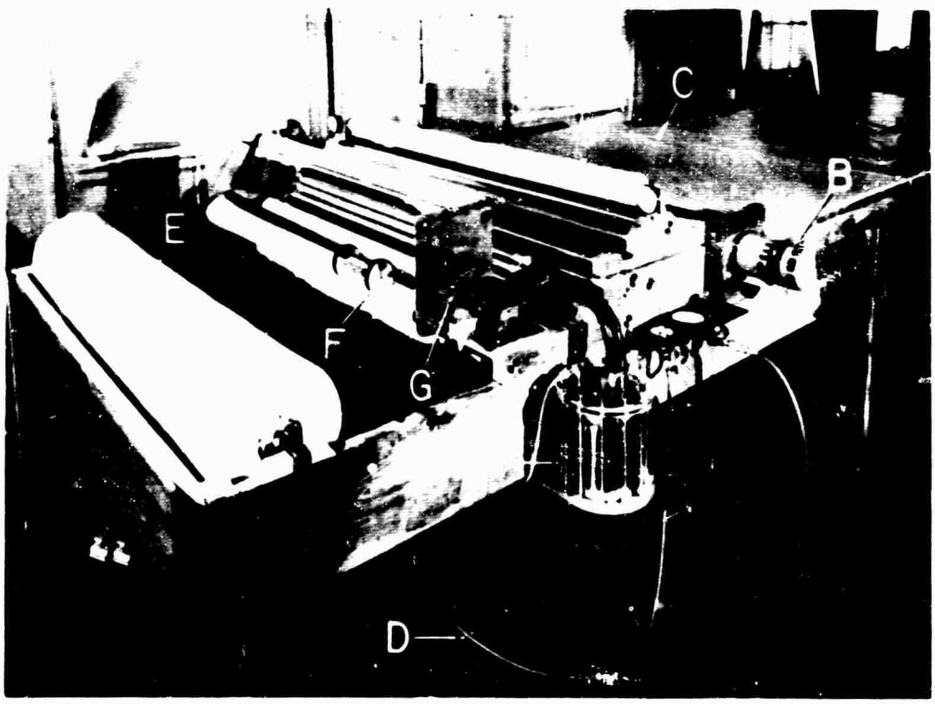

Figure 2. Overall View of Multi-Ply Coating Machine. [A, Pressurized Resin Reservoir (One Other Located On Opposite Side of Table); B, Reduction Drive Motor Which Draws Fabric Through Machine Onto Take-Up Roll; C, Cutting Table; D, Air Lines To Pressurize the Resin Reservoirs; E, Large Supply Roll of Polyethylene Film (Stored in This Position); F, Fabric Length Measuring Counter; G, Mounting Rack Which Carries the Dry Fabric Feed Rolls (3 Ply as Shown)]

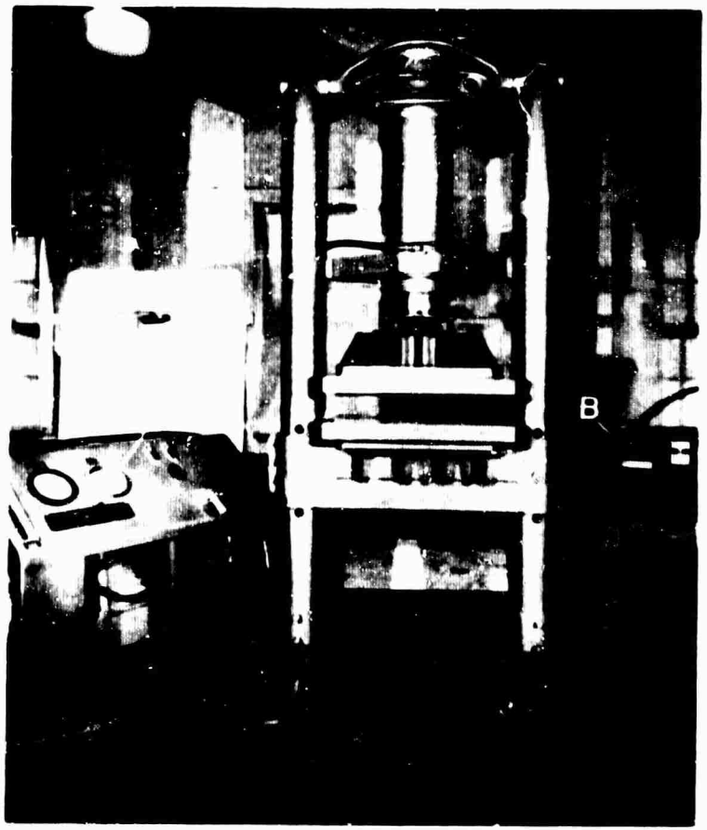

Figure 3. High Pressure Hydraulic Press. (A, Control Gage for Setting the Desired Curing Pressure; B, Control Console for Monitoring and Regulating Temperature of Heated Press Platens.)

Figure 4. Vacuum Press. (The fiberglass-reinforced silicone rubber vacuum blanket is shown resting on the lower platen.)

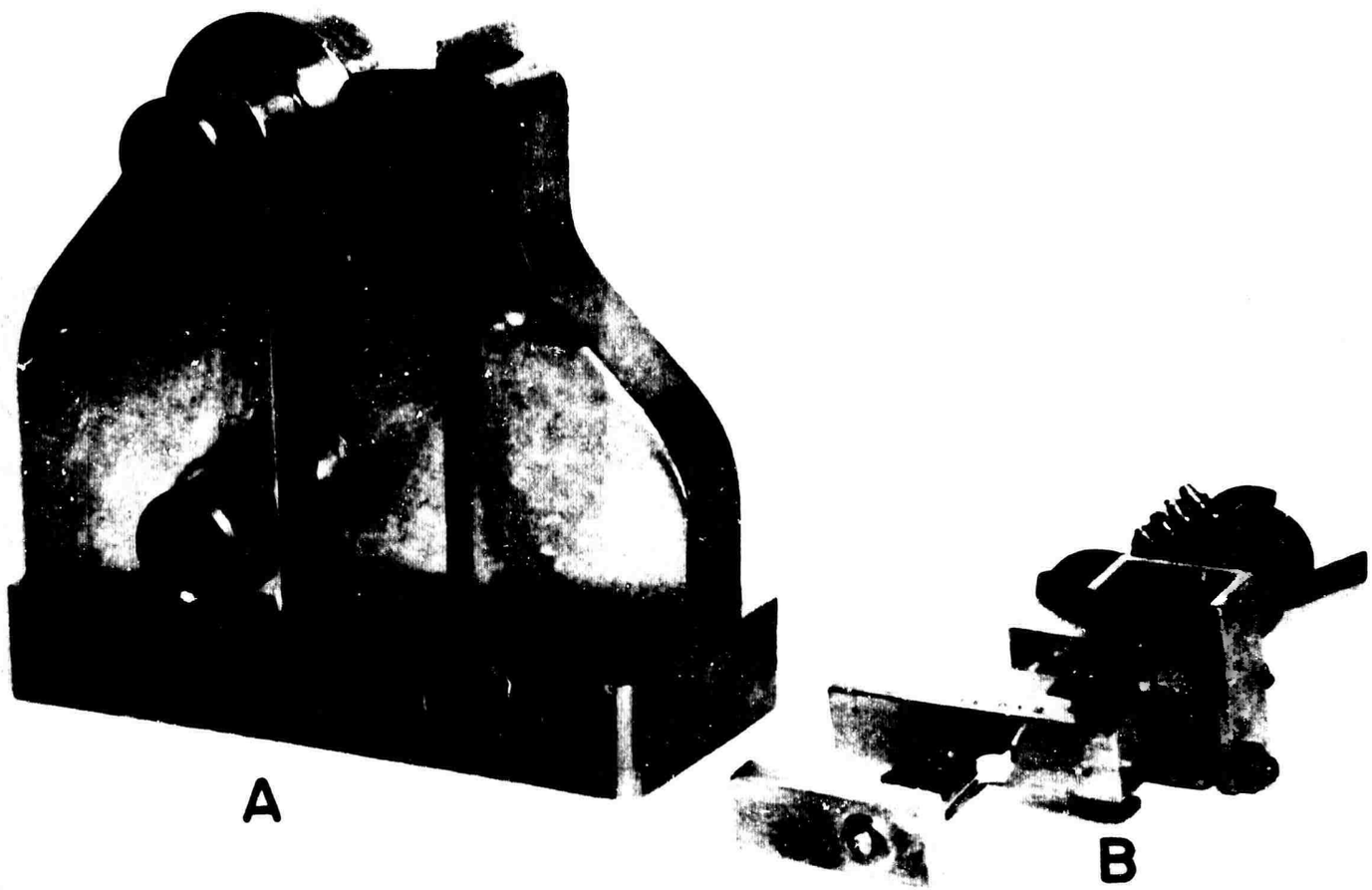

Figure 5. Thin-Laminate Compression Test Apparatus. (A, Compression Fixture with Specimen Inserted; B, Baldwin-Wiedemann Compressometer Used to Monitor Strain During Modulus Testing.)

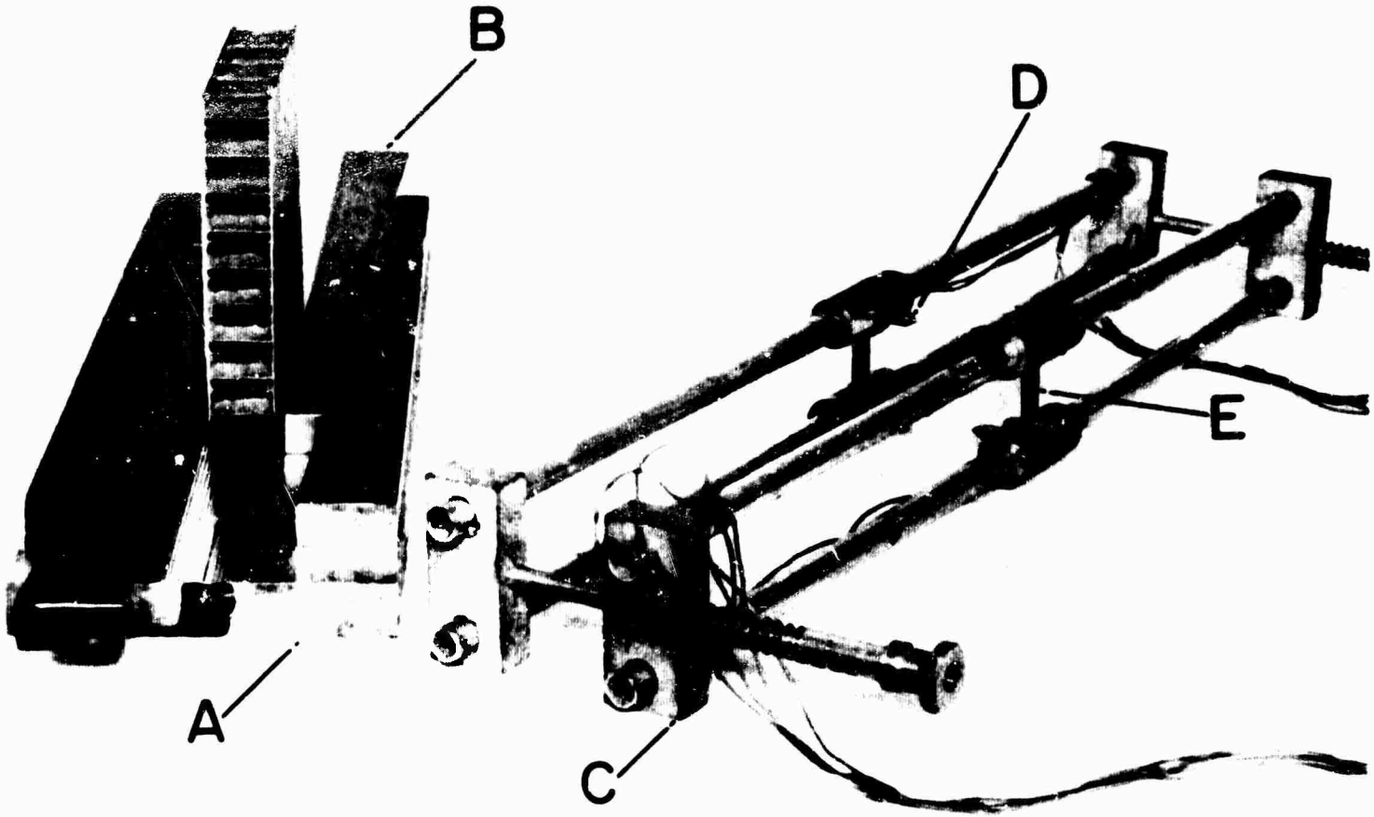

Figure 6. Sandwich Panel Edgewise Compression Test Apparatus for Fixed Loaded Ends. [A, Bottom Loading Plate Showing Guide Lines for Centering (Similar Upper Plate Not Shown); B, Sliding Wedge Grips; C, Sandwich Panel Compressometer; D, Needle Points (Two per Side) for Following Compression Strain; E, Strain Gage Flexure Strip.]

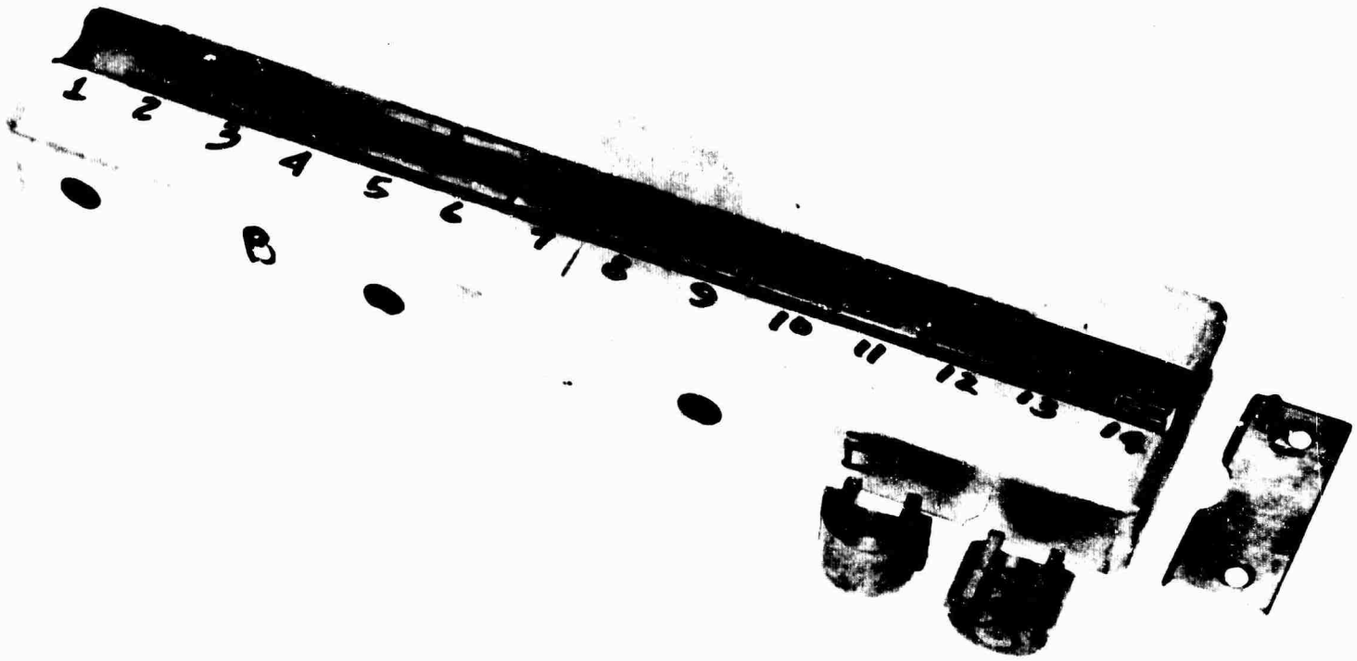

Figure 7. Sandwich Panel Test Fixture Providing a Simply Supported Loaded Edge During Edgewise Compression. (The segmented hinges permit differential rotation along the loaded end of the panel.)

FABRICATION PROCESS AND EVALUATION

As mentioned previously, the initial development work in sandwich fabrication centered around the versatile, low-cost, single-step method of construction whereby the facings were laminated and bonded to the core in one operation, the core-to-facing bond being effected by the basic resin system. The materials used consisted of electrical (E) glass 181-style Volan A-finished fiberglass fabric, EPON 828 epoxy resin activated by curing agent Z, and Douglas Aircomb, a phenolic impregnated paper honeycomb core with hexagonal cells. The press used was a vacuum type employing a 33-inch by 45-inch fiberglass-reinforced rubber blanket.

The resin-curing agent formulation used was 100 parts resin to 20 parts curing agent by weight. To insure complete mixing of the high viscosity room temperature resin with the normally crystallized curing agent, the resin was first heated to 120 degrees Fahrenheit and the curing agent to 150 degrees. The materials were then quickly mixed and used. Impregnation of the fiberglass fabric was accomplished by way of a hand-cranked coating machine which drew a continuous single ply of fabric through a heated resin vat and onto a take-up roll. The fabric was cut from the roll as needed--panels were usually sized 30 by 15 inches for vacuum pressing.

The sandwich facings were formed by stacking the cuts of fabric on a thin (1/32-inch) aluminum caul sheet. The caul sheet was then placed on the heated press platen set at the cure temperature; a polyethylene film was stretched across the wet laminate; and for approximately 2 minutes, the excess resin was hand-squeegeed out of the wet laminate so that the appearance of a uniform distribution of resin was obtained. Much difficulty was encountered in producing facings of predictable resin content (ratio of weight of resin to total weight) and in preventing small air pockets from being trapped between the plies, especially on the larger specimens. Further experience with the hand-working technique clearly established the need for a mechanical means of impregnating and laminating the fabric.

After this operation, the caul sheet was removed from the platen and set aside until another wet laminate could be prepared and squeegeed to provide the opposite facing of the sandwich panel. Upon completion of both wet facing lay-ups, they were either used directly or allowed to B-stage at room temperature (B-staging was used later in the program).

Assembly of the sandwich panel was the next step in the procedure. To insure uniform pressure application to the panels, as well as to protect the vacuum blanket, a wooden frame of sandwich-plus-caul thickness was always assembled tightly around the sandwich on the lower platen of the press. Approximately half of the frame was installed, the sandwich was assembled in place on the press, and then the frame was completed.

Thus, when the raw facings were ready for use, their polyethylene covers were stripped off, and as a first step in the assembly of the sandwich,

one of the caul sheets with a laminate was inserted in the frame on the open press which was preset at the desired cure temperature. Next, a slice of core, previously cut with a sharp knife, was placed on top of this lower facing so that the core ribbon paralleled the fabric warp direction, and finally, the second caul sheet and laminate were inverted and placed on top of the core slice with the laminate against the core to complete the sandwich assembly.

The press was closed, the full vacuum was drawn immediately, and the pressure held until the desired cure time had elapsed. The cure cycle was then completed by a 2-hour postcure (afterbake) of the panel. The postcure was accomplished in an electrically-heated, recirculating, hot-air oven set at 300 degrees Fahrenheit.

Visual inspection of this first series of panels revealed a condition of resin starvation and the presence of air voids in the facings. In addition to an increase in facing resin content (approximately 10 per cent), several other approaches were taken in an effort to obviate, or at least lessen, these phenomena: (1) room temperature B-staging of the wet laminate, (2) two-step assembly where each wet facing was cured to the core independently, and (3) separate precure phase. The precure phase consisted of a dwell period in the closed press prior to the application of pressure to the panel, the time being determined by the time required for the resin to gel at the press temperature (the gel time minus 7 minutes). The B-staging and precure did improve the control of the resin flow during the mold phase of fabrication; and hence, both were used throughout the rest of the program (Figure 8).

Though the variation of the single-step method of construction, whereby the facings were cured to the core one at a time in the lower facing position, was not found to be the solution to the starvation problem and, hence, was not explored further at this point, it was realized that the method of assembly would have merit in certain applications. The method was used with good success and should prove valuable in the molding of sandwich in compound curves when the expense of matched inner and outer dies is not warrented.

In perfecting the fabrication, it was also determined that the full vacuum of 28 inches of mercury available on the press could not be used in the single-step type of sandwich construction. The low pressure on the under side of the facing permitted the resin-curing agent mixture to evaporate; this produced hardened bubbles and poor filleting to the core as cure took place (Figure 8). Twenty inches of mercury was the maximum vacuum found acceptable to avoid this problem.

The needed gel points (the relations between resin temperature and gel time at various states of B-staging) were determined by inspection. Twelve-inch-square, 3-ply patches saturated with resin were placed on a heated press platen, covered with a felt insulation blanket, and probed periodically with a small wooden stick until the resin string pulled out

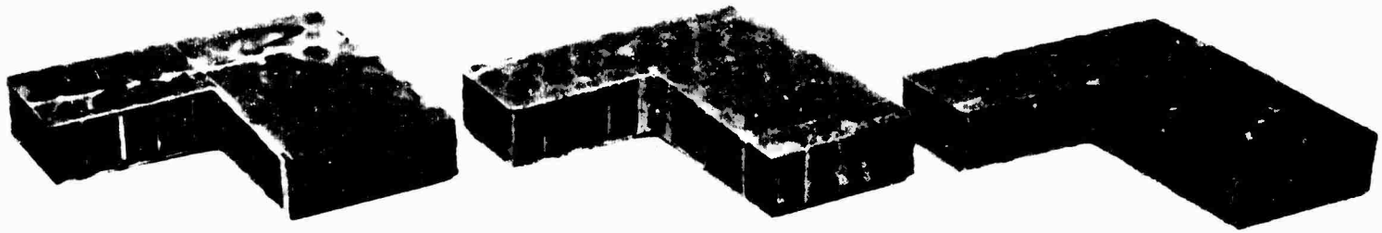

Figure 8. Typical Examples of the Three Types of Resin Flow Conditions Observed in Single-Step Fabrication Study. (From left to right: Controlled Resin Flow Produced by B-Staged Facings, Resin Starvation Produced by Direct Use of Wet Facings, and Resin Bubble Produced in Honeycomb Cells by High Vacuum.)

would break at 2-inch or 3-inch lengths. For each temperature and state of B-staging, the time between heat application and resin string break was the desired time value. Figure 9 is a plot of the data thus obtained.

As the quality of the facings was improved, the precise effect of the process variables (pressure and temperature) on the sandwich strength became more pronounced. Hence, the panel strength program was advanced cautiously until adequate fabrication information was generated, to insure that fabrication effects did not obscure the strength-theory verifications sought. To accelerate the acquisition of needed fabrication knowledge, a separate program was established in another USATRECOM contract (see reference 1), and further, the existing program was extended to include the design and construction of a multi-ply coating machine. The details of the coating machine that was developed were reported with the separate fabrication program (reference 1) since machine coated laminates were used, and that program was completed at an earlier date.

It wasn't until later in the program that machine coated laminates became available; therefore, the opportunity was taken to evaluate two of the promising commercial "pre-pregs", Coast Manufacturing and Supply Company's F150-11 and 3-M Company's 1002 style Scotchply. The F150-11 is a single-ply E-glass 181-style fiberglass fabric, B-staged epoxy impregnation; and the Scotchply is a nonwoven E-glass cross-ply fabric, B-staged epoxy impregnation.

It should be mentioned that, after the beginning tests with paper core, the other material variables (such as the facing thickness and the core materials, thickness and cell size) were set by the particular failure mode being studied.

Initially, the pre-pregs were used in the single-step construction of sandwich. The sandwich was assembled both with and without an intermediate adhesive using a precure of 3 minutes. In the former case, EC-1595 paste adhesive was used and was applied directly to the B-staged material. No postcure was used for these panels, since the cure was at high temperature (350 degrees Fahrenheit for 60 minutes). It was during this particular single-step construction work that aluminum core (of 5052 aluminum, 0.0013-inch-thick perforated foil) was introduced into the program. The use of this core required a different molding technique.[1] The molding was accomplished in two steps: a short time period (usually 8 minutes) at the desired laminating pressure (20 psi was found to be the maximum possible) followed by the remaining cure time at half the laminating pressure to prevent core crushing from thermal stresses. The crushing was attributed to the more severe curing conditions required by the commercial pre-pregs. Following the manufacturer's recommendations (350 degrees Fahrenheit) not only caused the core to expand as its temperature increased but also brought about a decrease in core strength resulting in a failure situation when the desired laminating pressure was applied throughout the cure period.

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[1] Two-step cure used for specimen groups 6, 9, 11, 13, 14, 16, 17, 18, and 19.

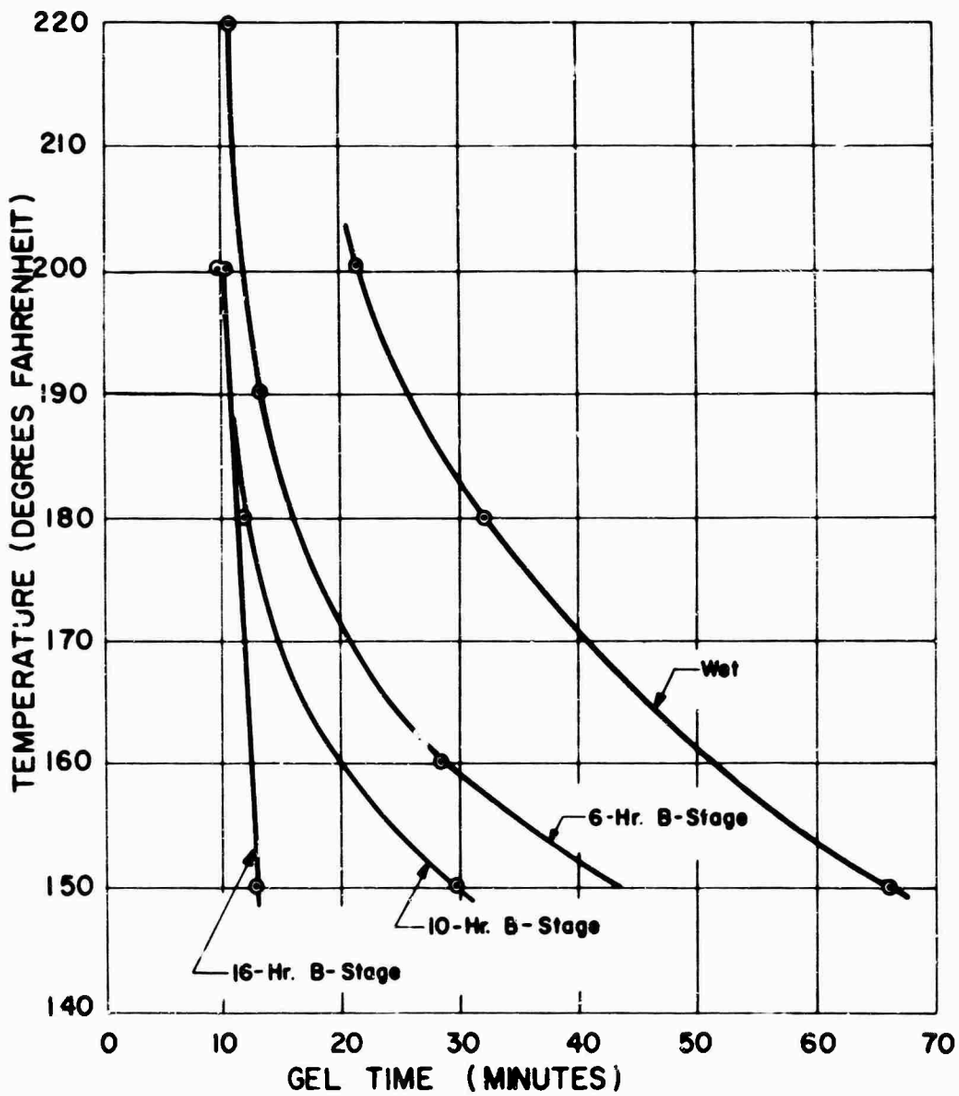

Figure 9. Effect of Temperature on Gel Time of Shell EPON 828-Z Resin in 3-Ply Lay-Up of 181 Volan A Fabric.

Prelamination of the facings and sandwich assembly by a separate bonding cycle was also investigated with the commercial impregnations. The results were so much better than those from the single-step construction that, at this point, the single-step method was abandoned in favor of this two-step method. Also, the molding was transferred to the newly constructed hydraulic press.[2] The size of the panels molded on the hydraulic press was usually 22 by 28 inches, except the HRP-core panels which were 19 by 28 inches.

More specifically, the transition to the separately bonded type of sandwich was made because of the adverse effect initial eccentricities had on the initiation of the panel failures (see face wrinkling under Experimental Results and Evaluation). Since the lamination of the single-step molded facings was brought about by the pressure applied by the ends of the core cell walls, invariably the laminate thickness and resin content were less in this joint. This was easily verified by visual inspection. In addition, the final rupture of the faces always followed the core cell walls. Also, this condition no doubt contributed to the warping, which always occurred in the thinner panels. In general, the separately bonded type of construction where the smooth flat prelaminated facings are bonded to the core in a second step was found much more suitable for the precise laboratory strength tests. As an indication of the capabilities of the smooth facing laminates, their strength weight ratio (ultimate compressive strength divided by the specific weight) is approximately ten times that reported for 2024-T36 sheet aluminum.[3]

Since the facing laminate must be kept clean for bonding, a parting agent was not used between the laminate and the caul sheet as for the single-step sandwich (thinned Dow-Corning DC-7 was used previously). Instead, clean 3/32-inch-thick Teflon sheets were used for caul sheets. This approach proved effective, though chipping away resin and scrubbing the caul sheets with soap and water after each use became a vital part of the process.

To continue with the actual fabrication process, the pre-pregs were kept under refrigeration and had to be thawed (usually 25 to 30 minutes) before being stripped of their film covers and stacked for lamination.

The following procedure was used in the construction of the facings for the separately bonded sandwich. After the cuts had been stacked to obtain the desired thickness and rolled flat with a heavy steel bar, the cauls and laminate were placed in the heated press, precured for 3 minutes, and then pressed at the temperature and pressure recommended by the

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[2] A photograph of the press can be found on page 5.

[3] B-staged strength data for 180° F, 90 minutes, and 70 psi cure was obtained from Table 12 in reference 1.

manufacturer (350 degrees Fahrenheit for 60 minutes at 50 psi). These and all other laminates were postcured prior to being bonded into sandwich even though the bonding was accomplished at high temperature in some cases. This was done to insure that full strength would be developed for the laminate properties tests. The usual postcure of 2 hours at 300 degrees Fahrenheit was employed.

Three types of adhesive were employed to effect the core-to-facing bonds: 3-M Company's EC-1595, a single-component thixotropic paste; Armstrong Resin Company's A-12, a two-component thixotropic paste (mixed 1 to 1 by weight); and 3-M Company's AF-110B, a B-staged supported film. Prior to bonding, the facing laminates were lightly sanded (00 grit paper) and degreased with acetone. The paste adhesive systems were then applied to the facings by a 3-inch-long notched edge scraper (eight notches per inch at 3/64-inch depth) and the film supported adhesive, by cutting the desired size and placing it on the facing or core.

As before for the single-step method, the core ribbon was oriented parallel to the facing warp except for the HRP cores which were oriented perpendicular because these cores could be purchased only with a maximum dimension of 19 inches in the ribbon direction. The assembled sandwich was then inserted in the press and the pressure set at 10 psi with the temperature and time regulated according to the manufacturer's recommendations (350 degrees Fahrenheit and 60 minutes for EC-1595 and AF-110B, and 250 degrees Fahrenheit and 30 minutes for the A-12 adhesive).

After the multi-ply coating machine became operational, both 2- and 3-ply simultaneously impregnated lay-ups were used for sandwich facings with excellent results. This was true of the 143-style fabric (used in intracellular buckling tests in the latter part of the project) as well as the 181-style. Four-ply laminates were also fabricated by stacking the 2-ply pre-pregs. The handling of the machine output was similar to that of the commercial pre-pregs, particularly with regard to cold storage. The impregnation was unrolled from the machine take-up reel after 10 hours of room temperature B-staging, cut to the desired sizes with scissors, and, to retard the resin cure, stored in a freezer set at 5 degrees Fahrenheit.

This completes the description of the basic work on fabrication that was accomplished during the present program. The next step in the evalution of fiberglass sandwich for aircraft structures should be the fabrication of curved structural panels. No doubt many special techniques and adaptations of those for flat panels will be required. Certainly, the FRP fabrication explorations made in this program point out the great need for knowledge in this area--not only regarding the sandwich constructions optimum for each special application permitted by the materials great versitility, but also pertaining to the design allowables for the facing laminates, once the particular optimum curing conditions have been determined.

EXPERIMENTAL PROCEDURE

Because of the vast number of process variables that affect the final strength of FRP materials, the procedures used in testing and in the preparation for testing are reported in great detail to facilitate thorough data analysis. The procedures are described in two sections according to the nature of the experiments.

A. Tests for Determining Material Properties

In order to compare calculated values of sandwich failure stress with test values, it was necessary to confirm or, in some cases, obtain the strength properties of the constituent materials. The procedures used in these supporting tests are presented in the following paragraphs. The descriptions include specimen preparation as well as specimen measurment procedures, and, in some cases, mention is made of the data reduction techniques.

1. Sandwich Plate Shear Test

To obtain the shear properties of the honeycomb cores used in the sandwich constructions, it was found convenient to utilize the pieces of the sandwich panels remaining after the removal of the specimens designated for the buckling tests. This procedure had the additional advantage of testing the core after it had undergone sandwich fabrication, permitting the detection of any adverse effect by comparison with the manufacturer's published data. It can be seen from the values in Table 11 that the cores fared fabrication rather well--in fact, in most cases the OURI test values are slightly higher than those listed by the manufacturers.

Two-inch-by-6-inch specimens were cut both perpendicular and parallel to the core ribbon direction and tested in plate shear (shear parallel to the facings) according to MIL-STD-401A. The cutting was accomplished on a table saw equipped with a 10-inch-diameter, extra course grit, tungsten carbide abrasive wheel (PERMA-GRIT Number 19758).

To accomplish the tests, it was necessary to bond 1/2-inch-thick steel loading plates to the facings of the specimens. The facings were prepared by lightly sanding their surfaces with number 00 grit sandpaper and then degreasing with acetone; and the plates, by stripping off all of the adhesive remaining from previous tests, washing in water and drying, and then sandblasting the contact surface.

EPON 6 adhesive was used throughout most of the program for the specimen-to-fixture bonding. The plates were warmed to 120 degrees Fahrenheit; then the paste adhesive was applied with a notched edge scraper. The specimen was placed on the prepared surface of one of the plates and pressed onto the adhesive film with a slight twisting motion to insure uniform distribution of the adhesive. The other steel plate was then placed on top of the sandwich and seated in the same manner. During this operation, the specimen was carefully positioned with respect to guide lines on the plates so that the line of action of the applied force would be directed through the diagonal corners of its core (Figure 10). After alignment, the entire assembly was placed in a recirculating hot-air oven and cured for 1 hour at 200 degrees Fahrenheit.

The specimens were installed in the testing machine by means of self-aligning hinged fixtures as shown in Figure 10. Shear deformation was measured with a dial gage graduated in ten thousandths of an inch which was mounted on one of the steel loading plates with its stem in perpendicular contact with an anvil fixed on the other plate (Figure 11).

Either a 100,000-pound-capacity Baldwin testing machine or a 10,000-pound-capacity Instron testing machine with an x-y recorder was used for these tests. Each specimen was preloaded twice to about 20 per cent of the anticipated ultimate load. The test run was begun after taking the slack out of the system with a 200-pound load and then setting the dial gage at zero. The crosshead speed used was 0.050 inch per minute.

The test data were accepted for modulus calculation regardless of the type of failure (steel plate-to-facing bond failure, core-to-facing bond failure, core rupture or yield); however, only core rupture or yield was logged as an ultimate failure. The shear area for each specimen was obtained before the test from length and width measurements made with an engineering scale read to the nearest 0.01 inch.

The calculation of the ultimate shear strength followed the usual definition of load divided by area and the shear modulus, the usual definition of the slope of the stress-strain curve multiplied by the core thickness (reference 10, bottom of page 7). These data are tabulated in Table 10 on page 63, and the results noted are presented on pages 32, 33, and 34.

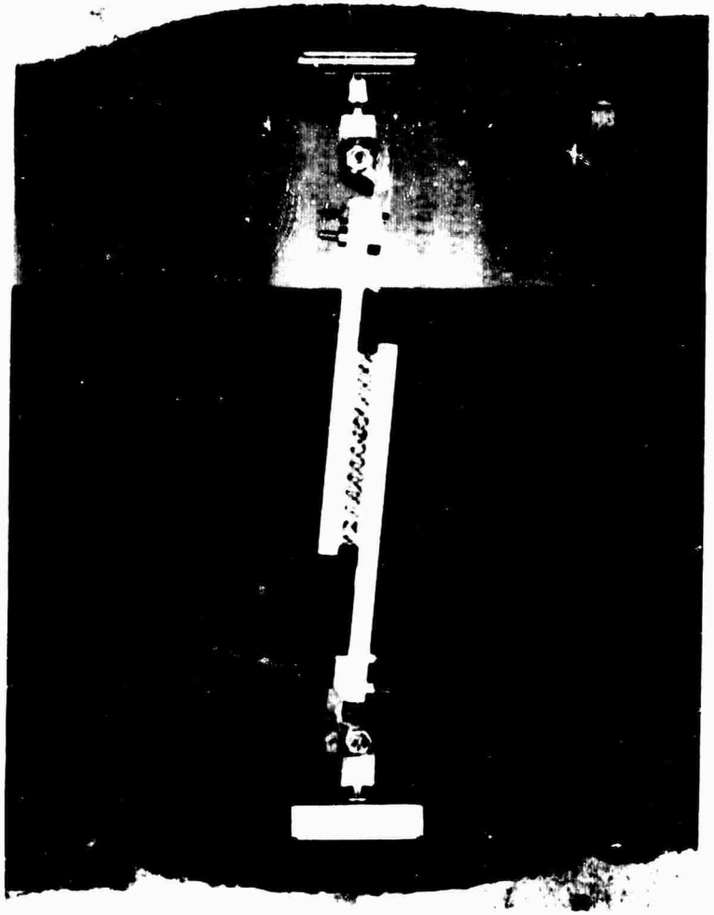

Figure 10. Test Setup for Sandwich Plate Shear Test Showing Specimen Installation.

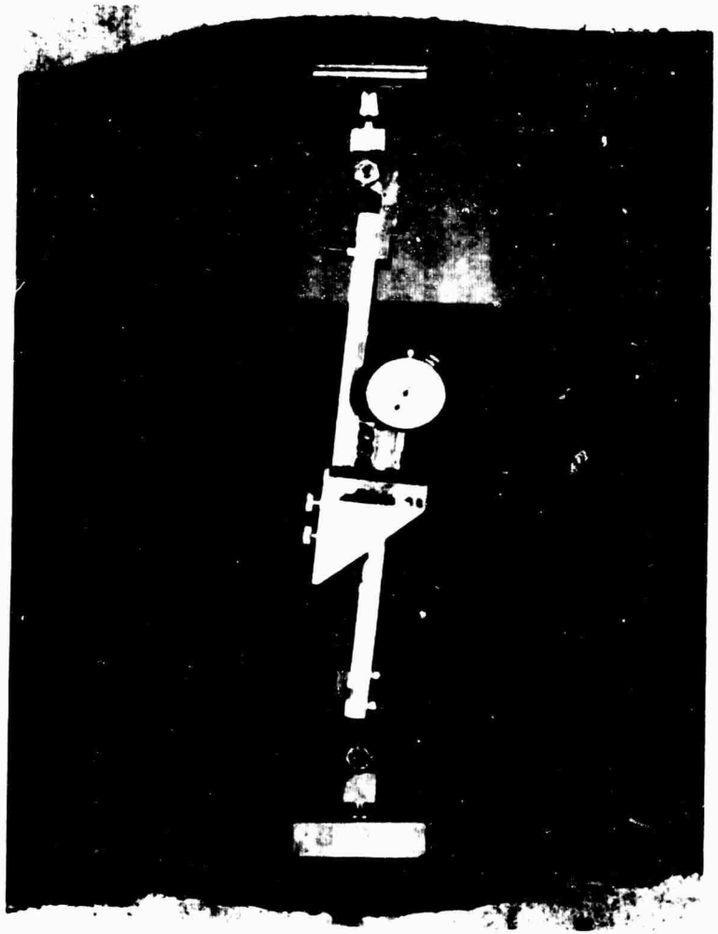

Figure 11. Test Setup for Plate Shear Test of Sandwich Shear Modulus Showing Test Specimen and Dial Gage Arrangement for Measuring Deformation.

2. Flatwise Tension and Compression Test

A limited number of tests were conducted to monitor the flatwise tension and compression properties of the honeycomb cores in sandwich construction. These tests served to not only sample the core properties, but also to permit observation of any fabrication effects and, in the case of the tension tests, to monitor the core-to-facing bond strength.

The same cutting procedure as previously described was used to obtain specimens for these tests. For flatwise tension, it was necessary to bond loading blocks to the specimen faces. This was accomplished in the same manner as for the shear specimens.

Figure 12 shows the flatwise tension test setup in the 10,000-pound-capacity Instron testing machine. The load was applied at the rate of 0.05 inch per minute of crosshead travel. Since the 1-inch-square specimens (MIL-STD-401A) were cut to within 0.01-inch accuracy, the ultimate load was read directly as the ultimate stress.

As discribed in MIL-STD-401A, 2-inch-by-2-inch sandwich test specimens were used for the flatwise compression tests which were conducted in the 100,000-pound-capacity Baldwin testing machine operating at a crosshead speed of 0.033 inch per minute. Crosshead movement measured with a dial gage was taken as the core deformation in these tests.

The data from this series of tests are tabulated in Tables 11 and 12 on pages 64 and 65, respectively. Figure 20 on page 33 illustrates the major result of the tests.

3. Core Modulus of Elasticity Test

To sample the compressive properties of the core perpendicular to the flute direction, 5-inch-wide by 10-inch-long specimens were cut from the large sheets with a sharp knife, and their ends were cast in polyester resin reinforced with molding plaster. Specimens were cut with the length demension running both perpendicular and parallel to the core ribbon.

The test fixture consisted of two vertical slotted guides between which the core was placed. Thus, during the vertical compressive load application, the specimen was free to expand perpendicular to the flute direction while being restrained from buckling in the flute direction. The load was applied in increments by evenly weighting the upper end of the specimen (the weights were accurate to 0.01 gram), and the deflection of the end was measured with an engineering scale to the nearest 0.01 inch. Prior to each test, the cross-sectional

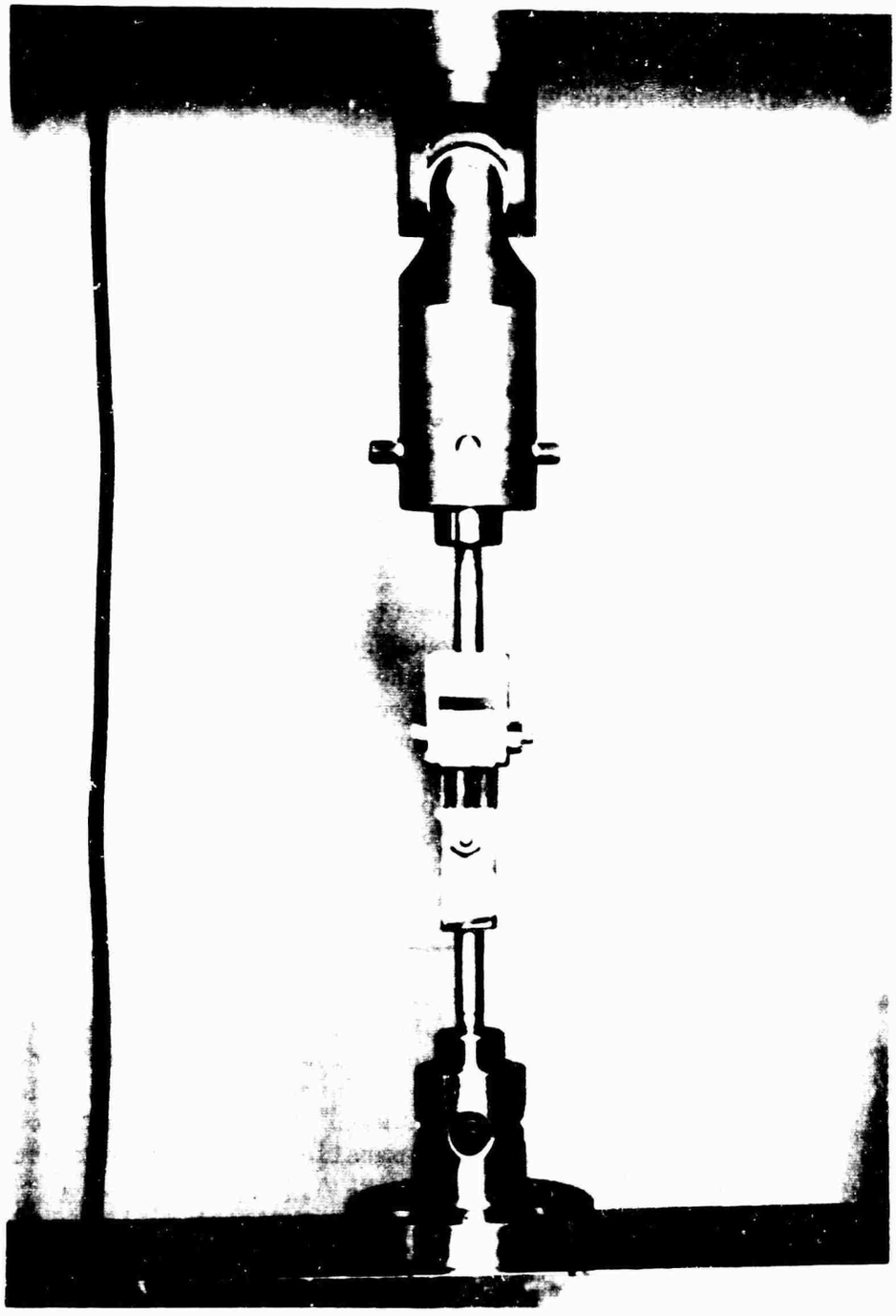

Figure 12. Test Setup for Flatwise Tensile Properties of Sandwich.

dimensions were measured to within 0.05 inch for the stress area calculation. The strength values are tabulated in Table 13 on page 65.

4. Compression and Tension Tests of Facing Laminates

During the latter part of the research program when the separately bonded type of sandwich was being used exclusively, strips were cut from the prelaminated facings to test for materials data in support of the theoretical calculations. One-inch wide strips were cut from the facings with a large sheet-metal-type shear, further trimmed to length, and then ground to final dimensions and squareness (compression specimens: 0.875 inch by 3.675 inches; tension specimens: 0.750 inch by 9.0 inches).

To prepare the compression specimens for testing, they were coated with a powered molybdenum disulphide lubricant (Molykote Z) and lightly clamped (screwed finger tight) in the test fixture. The particular test fixture used was developed in the separate fabrication program (reference 1). It functioned to prevent buckling of the thin laminates, as can be seen from the photograph of Figure 5 on page 6.

The fixture with the test specimen was placed on the lower platen of the testing machine (either the Baldwin or the Instron) with the top of the specimen fitted into a tapered slot in the upper loading block. The specimen was then vertically aligned and a wedge inserted into the slot along the end of the specimen to provide a fixed-end condition during loading (Figure 13).

Figure 13 also shows the installation of the Baldwin-Weidemann B3M extensometer which was converted to measure compression strain. The instrument was used in conjunction with an x-y recorder which plotted directly the load versus deformation curve. In the test, only a portion of the curve was obtained in that the instrument was removed at 75-per cent load to prevent its damage.

At a crosshead speed of 0.050 inch per minute, each specimen was loaded to failure. Except in a very few cases, the compression failures occurred within the supported length of the specimen as typified by B, C, D, and E in Figure 14 on page 23.

The final step in the preparation of the tensile specimens was the cutting of the 0.007-inch influences in the edges. As discussed in reference 1, these tiny influences in the straight-sided specimens served to preclude failure in the grips.

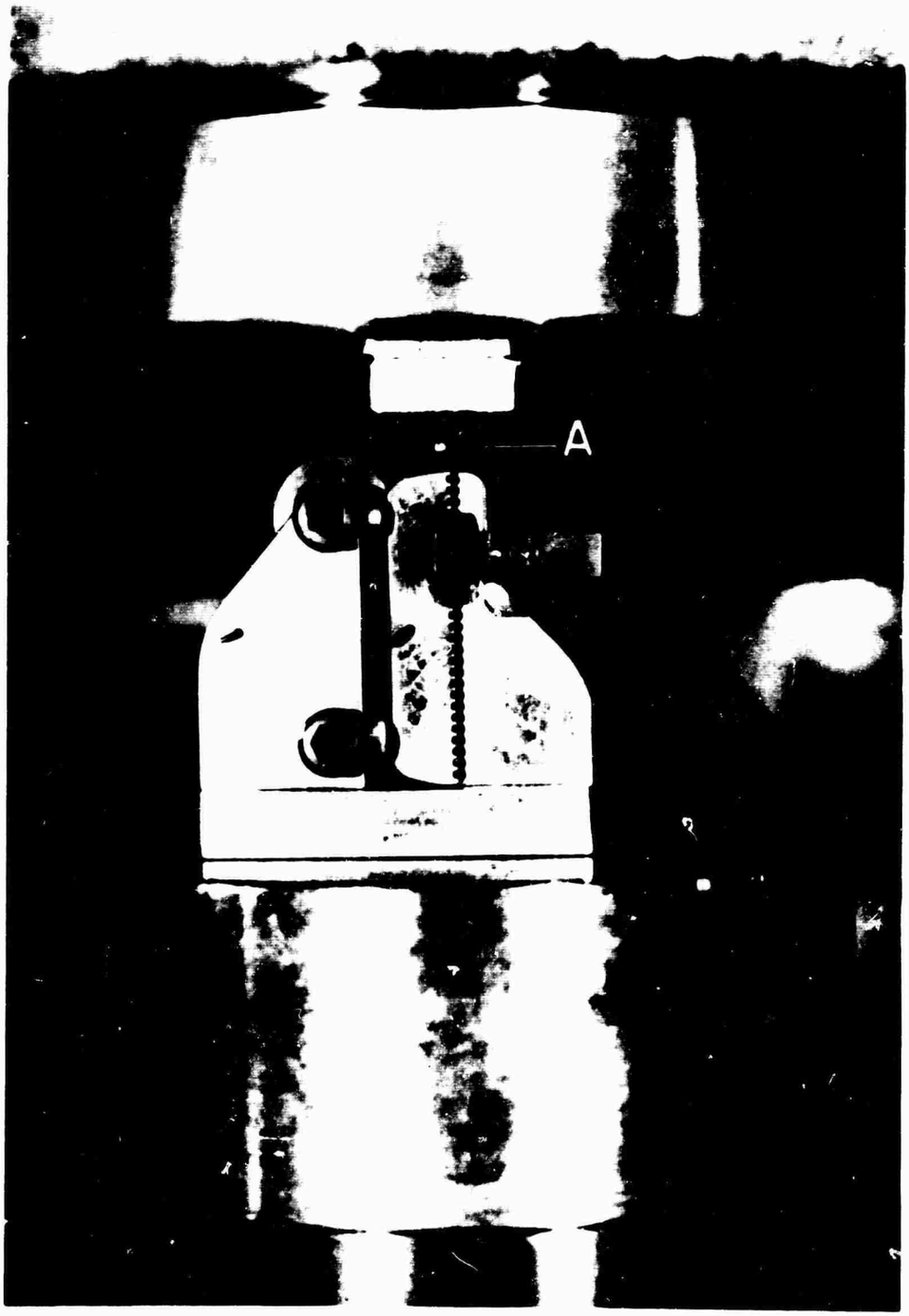

Figure 13. Test Setup for Compression Test of Thin Laminates Showing Specimen and Compressometer Installation. (Pointer A identifies the wedge grip securing the specimen in a fixed-end condition during loading.)

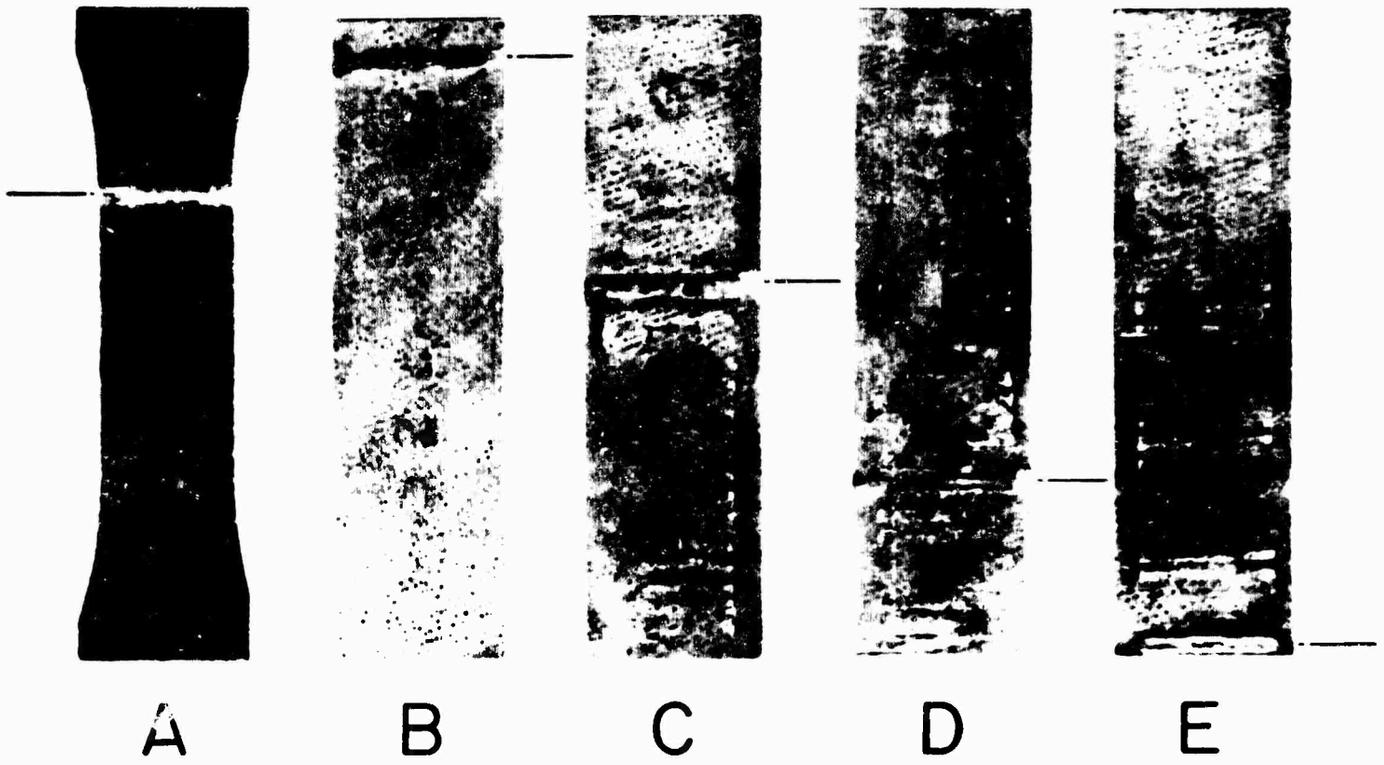

Figure 14. Laminate Compression Specimens. (A, Specimen Shape Initially Evaluated; B, C, D, E, Straight Sided Specimens Which Were Used Throughout Test Program Showing Various Points of Failure That Occurred Within Fixture Grips.)

The test coupons were placed in Templin grips with self-adjusting jaws which were attached to the loading heads by bolts resting on cylindrical seats to assure true alignment. Figure 15 shows the setup. The same extensometer used in the compression tests was clamped to the edges of the specimen with the knife edges being vertically equidistant from the specimen notches. The remaining details of the tests are identical with the compression tests.

For both types of tests, stress area was based on width measurements taken with a caliper graduated in thousandths of an inch and on thickness measurements taken with a vernier micrometer. The width measurement was obtained by a random sampling of each group of specimens; however, after testing, thickness measurements were made for each individual specimen at a point 1/2 inch on either side of the rupture; the two readings were then averaged.

Because the size of panel that could be fabricated on the laboratory press was limited (22-by-28-inch platens), it was not always possible to obtain laminate specimens having the preferred orientation relative to the direction of the weave of the fiberglass fabric, especially when the large buckling panels had to be extracted from the finished sandwich panel. Table 14 is a tabulation of the strength values obtained in these tests. The results may be found on page 35 (Figure 23).

B. Buckling Tests

A large number of tests are involved in the sandwich buckling studies; therefore, for the convenience of the reader, the test data are arranged in tables according to mode of failure and boundary conditions (Tables 4 through 9). The specimens employed in each specific test are denoted by a group number which permits complete identification of the structural material. In Table 3 the specimens are identified according to the constituent materials, the fabrication method, and adhesive. The details of fabrication may be found in the section devoted to fabrication. For the sandwiches fabricated by the separately bonded technique, Table 14 is provided to specify the conditions of fabrication of the facings.

The sandwich specimens from both the single-step and the separately bonded constructions were prepared in a similar manner for all the tests. They were cut from the press-size panels with the table saw described in the section on the plate shear test, and then the load-bearing edges were reinforced with a potting compound. The reinforcement prevented localized failures and provided a more uniform loaded-edge condition. Polyester resin filled with a high-strength

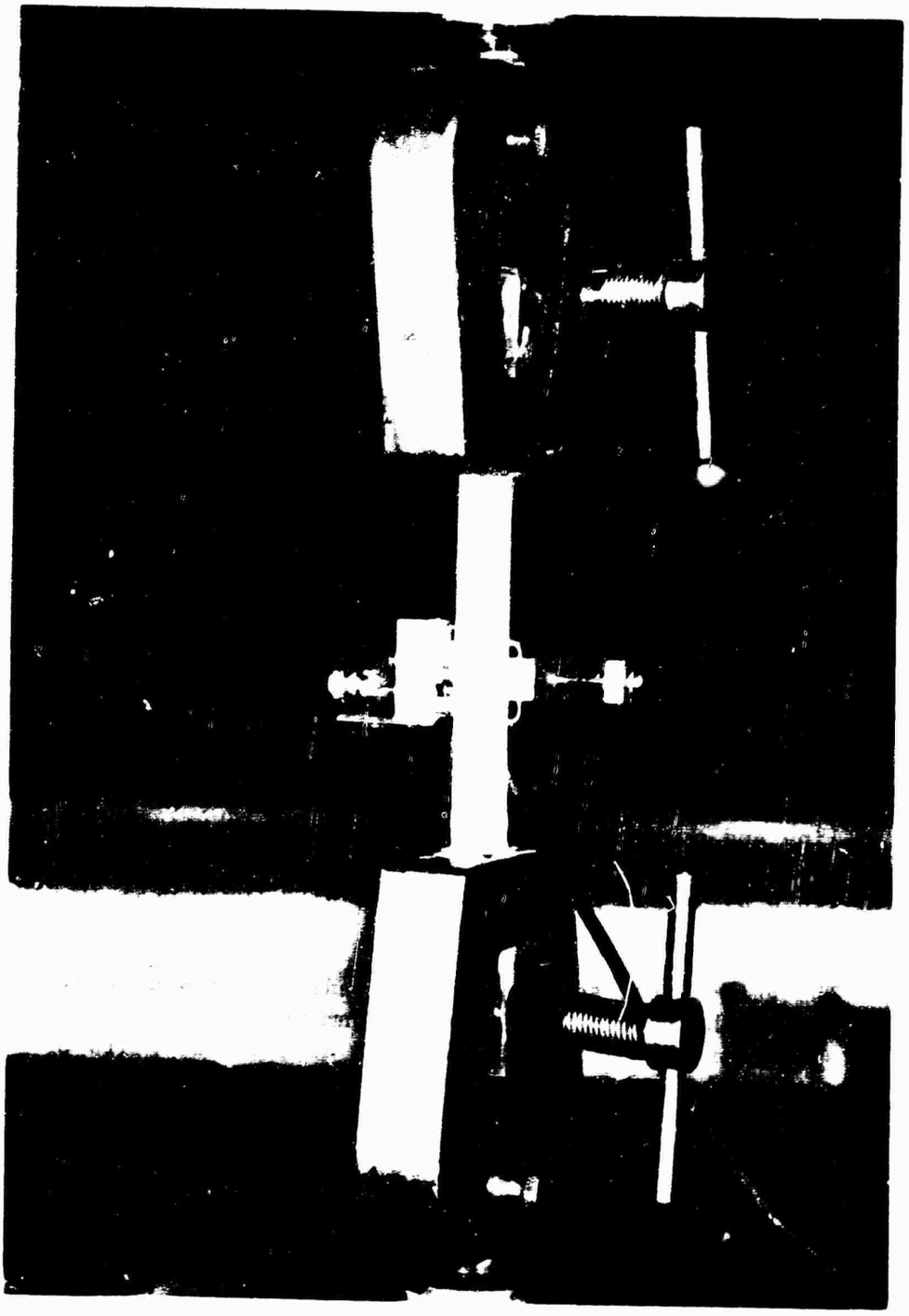

Figure 15. Test Setup for Tension Test of Thin Laminates Showing Specimen and Baldwin-Wiedemann Extensometer Installation.

molding plaster (30 per cent by weight) functioned well for this purpose. Initially, an aluminum filled epoxy resin was tried, but better results were obtained with the polyester.

When the reinforcing resin had cured (1 hour at room temperature), the loaded edges were ground flat and parallel to each other and orthogonal to the facings. The accuracy of 0.001 inch run-out along each of these edges was found acceptable to insure equal strain in the facings under load in the test jigs.

1. General Buckling Tests

The initial test work in the general buckling of sandwich panels concerned the development of testing techniques and specimen restraint systems. At first, a limited amount of testing was done on hinged-end plate columns to observe the threshold of panel buckling. These data are recorded at the top of Table 8 for reference. Next, the condition of clamped loaded ends and hinged sides was explored.

A 200,000-pound-capacity Tinius Olsen balance-beam testing machine, operating at a crosshead speed of 0.033 inch per minute, was used to apply the edgewise compression load to the specimens. An important step in setting up the tests was the securing of the mill-faced loading blocks to the upper and lower loading platforms of the testing machine such that the load would be uniformly distributed across the edge of the specimen. The blocks were shimmed as necessary until their surfaces were perpendicular to the load line and parallel to each other throughout their surface area. This condition was verified before each series of tests by feeler gage measurements with the blocks in close proximity.

The guide lines scribed on the loading blocks were used to center the specimens. After alignment and centering, the specimens were locked in place at each end by pairs of accurately machined steel wedges. The blocks and wedges are shown in Figure 6 on page 7.

The side clamps were then screwed snugly against the specimen and the large panel compressometer[4] was installed, as facing strain was monitored during these tests. The facings had been drilled previously to receive the needle points of the compressometer (number 53 drill).

[4]Figure 6, see reference 1 for description

To insure that equal strain was occurring in each facing, the specimens were preloaded twice to 50 per cent of the anticipated maximum load prior to the test. This proof loading further served to eliminate the initial modulus of the facings (reference 11, page 60).

The load at which the beam of the testing machine dropped was recorded as the failure load; however, a more accurate failure criterion was subsequently developed. Though the loads were not considered precise, the stresses were calculated and tabulated (Table 7) for reference. The stress area was based on the average length measurement of the two loaded edges (measured to within 0.010 inch), and a thickness value was obtained by multiplying the nominal thickness per ply (0.01 inch) by the number of plies of fiberglass fabric in the facing laminate.

The improvements derived from the series of tests just described were used to investigate more thoroughly the general buckling of flat, rectangular sandwich panels when all edges were restrained as hinges, all panels being sized to buckle in a single half wave.

The hinged or simply supported edge condition was achieved for the loaded edges of the panels by a unique set of loading blocks or plates. These fixtures are in essence segmented hinges mounted in the steel loading plates. There are 14 1-inch segments per plate. Each segment consists of a roller block, grooved on top to receive the edge of the sandwich specimen and machined in a simicirular shape on the bottom to form the inner race of a roller-type bearing. The outer race for each block was machined into the loading plate. This recessing of the bearing into the loading plate placed the center of rotation of each segment precisely at the edge of the specimen.

The side clamps were a modified version of those used in the first series of tests. The grips were originally too wide, giving more of a fixed edge condition than the desired hinge. The final configuration consisted of a pair of steel angles fitted with 1/8-inch-by-1-inch steel plates which were tapered and ground to a 1/16-inch-thick knife edge at the point of contact with the specimen (Figure 16). When installed, these clamps extended past the loaded end of the specimen to avoid having any part of the specimen unsupported.

The setting up for the tests was similar to the procedure followed previously. With the roller blocks in line, the loading plates were fastened to the upper and lower tables of the testing machine and aligned as described before. After the loading

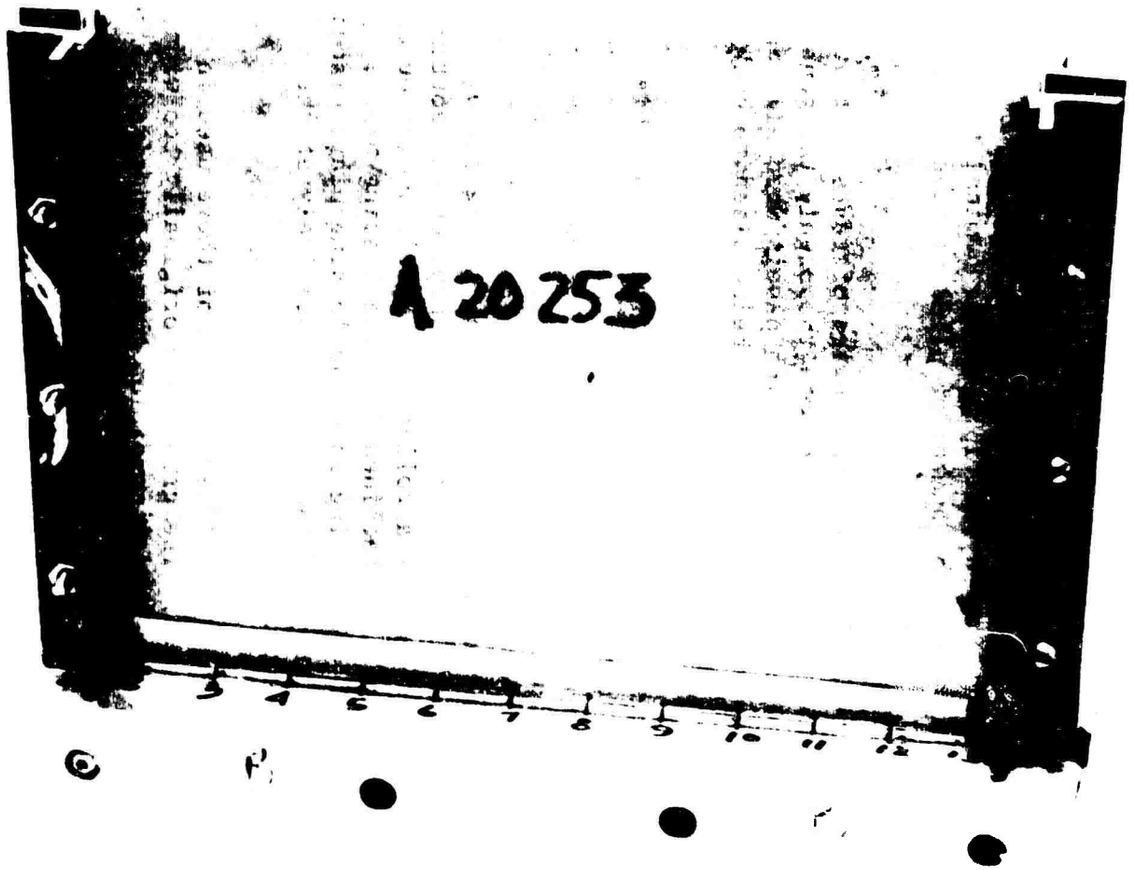

Figure 16. Installation of Sandwich Panel Test Fixtures Which Provide Simply Supported Edges During Edgewise Compression Loading. (The upper segmented-hinge loading fixture has been removed to show the knife-edge side grips. The side clamps were placed in close proximity of the hinges during the tests.)

head had been lowered on the specimen and the specimen had been tightened with paper shims, the side grips were bolted snuggly against the edges (but not curshing them). The segments of the hinges not in use were always removed prior to side-clamp installation. This permitted the specimen to overhang the active set of segments approximately 1/4 inch on each side so that the side clamps could be placed against the outside hinges, leaving none of the specimen unsupported.

As mentioned previously, a more accurate means was used to locate the failure load than just the drop of the balance-beam. Side deflection was the method chosen, as measured by a ten-thousandths-dial gage centered against and orthogonal to one facing.

Each panel was loaded continuously at a crosshead speed of 0.033 inch per minute until the panel failed. Side deflection was recorded at each increment of load until the deflection rate increased rapidly, at which time the gage was removed to prevent its damage. All specimens that buckled exhibited the same pattern of failure. The center deflection began as soon as load was applied and continued at a uniform rate until the critical stress was approached, at which time there would be a rapid deflection of as much as 1/2 inch.

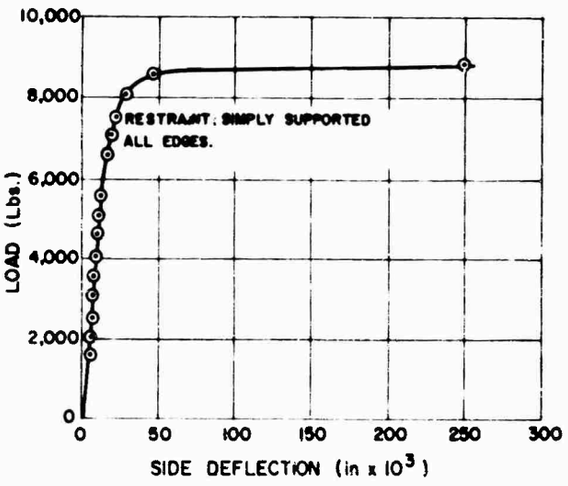

Figure 17. Typical Curve of Load Versus Side Deflection for General Buckling of Sandwich Panels. (Data taken from specimen group 12, a = 9 inches.)

The failure loads recorded were obtained from the plots of load versus side deflection. The load corresponding to the inflection point on the curve, as best as could be determined, was the accepted load. One of the better cases is shown in Figure 17.

Usually, the panels showed no apparent structural damage at the point of buckling. The heat resistant phenolic (HRP) core panels would return near to their original shape when unloaded, while the aluminum core panels would retain a permanent warp. Continued loading would cause the panels to break free from the test fixture, leaving a large wrinkle near one of the loading plates.

All of the specimens used in these tests were of the separately bonded type; hence, the actual facing thicknesses were used in the stress calculation. For the theoretical calculations, the effective dimensions of the panels (distances between clamps) were measured also (0.01-inch accuracy) as well as the thickness (t) of the finished sandwich (0.0001-inch accuracy). The data are tabulated in Table 8, and the results are presented and discussed on pages 35 through 39 (Figure 24 and Table 1).

2. Face Wrinkling and Shear Crimping Tests

The test procedures for these two buckling phenomena were similar and will be discussed together. Actually, the procedures were identical to those of the early general buckling tests except at the point where the side clamps were installed. No restraints were placed on the sides of the specimens used in the face wrinkling or shear crimping tests--only the loaded edges. Clamped or fixed loaded ends (produced by the previously described wedge grips) were used when the shear crimping was sought, and both clamped and hinged loaded ends when face wrinkling was deliberately sought.

The hinge fixture used here was the predecessor to the one described under general buckling. It differed in that differential rotation along the loaded edges of the specimens was not provided for (the hinge was not segmented) and the center of rotation was not precisely at the edge of the specimens.

At a crosshead speed of 0.033 inch per minute on the Tinius Olsen testing machine, the specimens were loaded to failure--until the load decreased abruptly. The nominal fabric thickness per ply was again used to calculate stress area. Many of these tests preceded the general buckling tests and, hence, served to assess the initial efforts at fabrication as well as assist in the development of the test fixtures and test techniques. The data are recorded in Tables 4, 5, and 6. The results from these tests are discussed on pages 40 to 47. Table 2 and Figures 25, 26, and 27 are part of the presentation.

3. Intracellular Buckling Tests

For the intracellular buckling (face dimpling) investigations, the fixed-end plate column was again used. Detection of the phenomenon was accomplished by a battery of dial gages measuring certain side deflections. The gages were mounted in pairs so that on one side of the panel, the stem of a gage was resting against the facing over the center of a core cell, while the gage on the opposite side was placed with its stem over the wall of the same cell. By comparing readings of such a

pair of gages, it was determined whether the movement indicated the expected face dimpling or lateral translation of the panel as a whole. Figure 18 shows the dial gage arrangement.

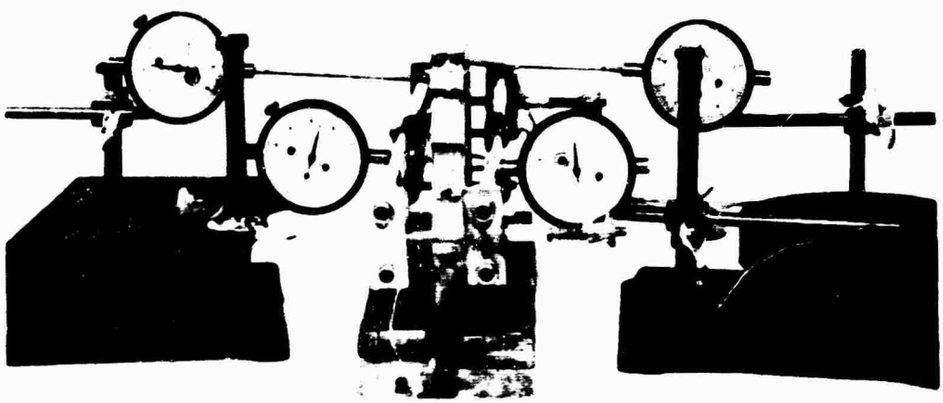

Figure 18. Test Setup for Intracellular Buckling Failure Mode. (Opposing dial gages were placed over a cell center and cell wall respectively. The upper set of gages were placed near the center of the panel during the tests.)

The setup and alignment of the specimens followed that described previously. The large panel compressometer was used to monitor facing strains and was especially beneficial in confirming the alignment of the specimens. With the specimen unloaded, all dial gages were then placed in position and their initial readings recorded. The load was applied at the rate of 0.033 inch per minute in the Tinius Olsen testing machine. The loading was stopped momentarily in 500-pound increments to facilitate reading of the dial gages.

As in the case of the general panel buckling tests, it was necessary to remove the dial gages prior to specimen failure to prevent their damage. Since the failure criterion was also the same (the point at which the center deflection of the

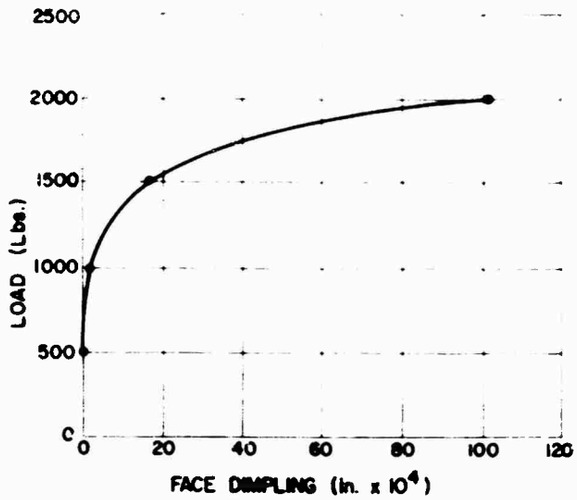

Figure 19. Typical Curve of Load Versus Amplitude of Dimpling. [Data taken from specimen group 31 (+R)]

facing spanning the cell opening rapidly increased) and since the specimens failed catestrophically shortly beyond the dimpling, needless to say it was difficult to obtain the desired data to permit accurate pinpointing of the failure load. Figure 19 shows one of the better plots obtained from the data.

The specimens for these tests were of the separately bonded type; hence, facing thickness obtained by actual measurement was used in the stress area calculation. The data are listed in Table 9, and the results are presented on pages 47 to 49 (see Figure 28).

EXPERIMENTAL RESULTS AND EVALUATION

The experimental results of the research program are discussed according to the type of experiment and mode of failure as follows:

A. Results of the Material Properties Tests

To accomplish the main objective of the research, the verification of existing strength relationships for FRP sandwich, it was necessary to obtain properties of the materials used to build the sandwich and to monitor these for effects of sandwich fabrication. The findings are duscussed according to the constituent and its property.

1. Sandwich Plate Shear Tests

As mentioned in the introduction to the procedure followed in these tests, core strength data were extracted from the manufacturers' publications and included with those obtained in this program (Table 10). It can be seen that the cores are up to par in strength and that no adverse effect was produced by sandwich fabrication. In fact, quite to the contrary, sandwich fabrication was noticed to increase the stiffness of the core in certain cases.

The fabrication process appears to influence the core properties through the filleting of the adhesive and/or resin during cure. Figures 21 and 22 are plots of shear strength and modulus, respectively, versus core thickness for the various adhesives employed in sandwich construction. The plotted data are averages obtained from Table 10. These plots indicate that the adhesive effect becomes more pronounced as the thickness of the core decreases. Greater effect is seen to occur in the case of the EC-1595 adhesive when it was applied to the B-staged fabric before cure.

2. Flatwise Tension and Compression Tests

These tests were very limited in scope and were intended to confirm published data where available and where not, to generate a sampling. The flatwise tension tests further served as a means of observing the sandwich core-to-facing bonds. The data are presented in Tables 11 and 12 and reveal two specimen groups lower in strength than the core materials. These are groups 13 and 16 where the EC-1595 was brushed on the pre-preg prior to single-step construction.

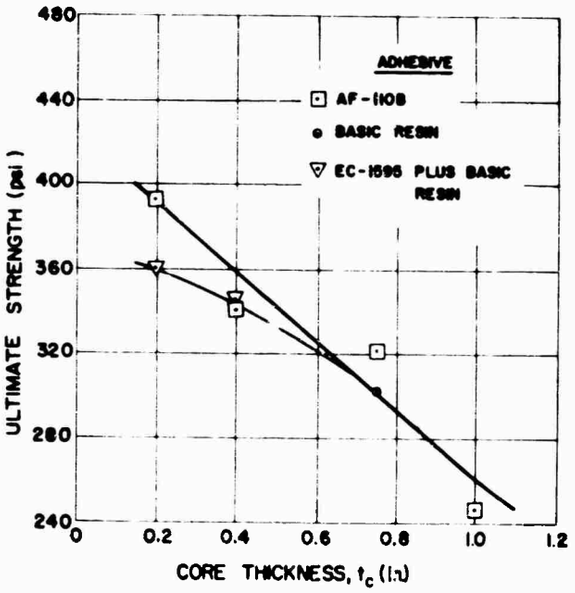

Figure 20. Relation Between Flatwise Tensile Strength of 3/8-Inch-Cell, 5052-0.001P Aluminum Core in Sandwich Construction and Nominal Core Thickness.

As could be suspicioned, the core tensile strength also displayed the adhesive effect noted in the shear properties. Figure 20 is a plot of flatwise tensile strength versus core thickness.

3. Core Modulus of Elasticity Tests

These exploratory tests were made to confirm the very low values of modulus of the core materials in the perpendicular to flute direction predicted by other investigators. The values obtained are tabulated in Table 13 on page 65.

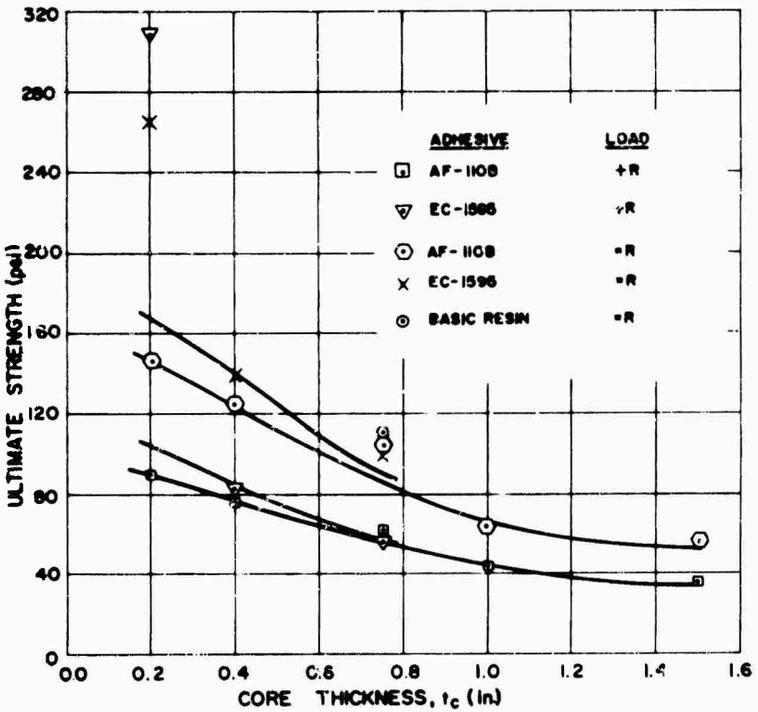

Figure 21. Relation Between Core Shear Strength of 3/8-Inch-Cell, 5052-0.001P Aluminum Core in Sandwich Construction and Nominal Core Thickness.

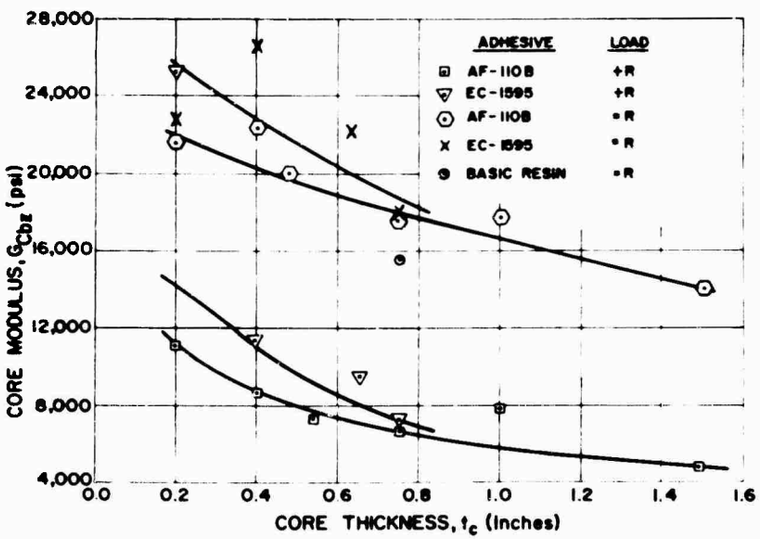

Figure 22. Relation Between Core Shear Modulus of 3/8-Inch-Cell, 5052-0.001P Aluminum Core in Sandwich Construction and Nominal Core Thickness.

4. Compression and Tension Tests of Facing Laminates

The properties for all the laminates are given in Table 14. Several thicknesses of facing laminates of the one material, F150-11 pre-preg, were used in the buckling studies, especially in general buckling. The properties of these facings were obtained by test specifically for use in the theoretical calculations of panel failure stress; however, an overall view of the properties themselves revealed the interesting thickness dependency noted by other investigators (reference 11).

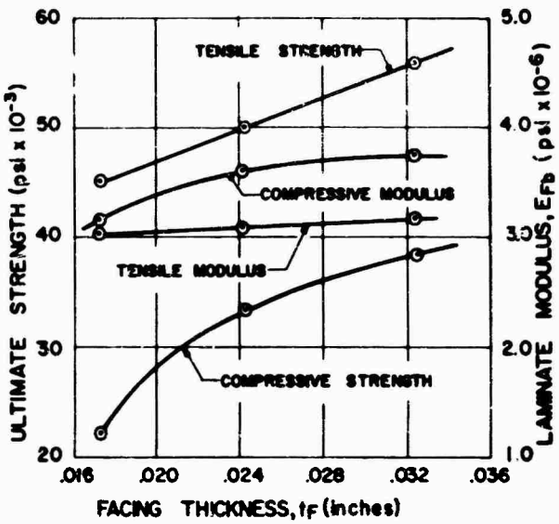

Figure 23. Variation of Facing Modulus and Ultimate Strength with Thickness.

Figure 23 illustrates the trend in which the strength properties are seen to decrease with thickness. The occurence is probably a surface phenomenon. The tiny surface flaws produced during fabrication no doubt have a greater influence on the strength of the thinner laminates. The data for these curves were obtained by averaging the values given in Table 14 for the 2-, 3-, and 4-ply F150-11 pre-preg laminates.

B. Results From Buckling Tests

Each of the buckling modes is discussed separately in the following four subsections.

1. General Buckling Tests

As explained in the test procedure, the beginning tests in general buckling functioned to develop testing techniques and fixtures, with the most precise work being done in the last series of tests with the hinged boundary condition. Location of failure load was seen to be a problem; therefore, a definite failure criterion was established for the more precise tests. Failure would be based on the character of the mid-panel side deflection measured during each test.

On the basis of the findings in references 5 and 6, it was anticipated that the panels would continue to take increased loads after the occurrence of the first large side deflection; hence, it was desired that the inflection point of the plot of load versus side deflection be used as the failure load. However, in practice, the side deflection took place very rapidly as the critical load was approached. In only a few cases was it possible to detect the secondary loading. The deflection-load plot was usually very near horizontal at the time when the dial gages were removed to prevent their being damaged; hence, this point was taken as the failure load (Figure 17, page 29). There were cases where the slope was not horizontal, and in these cases the inflection point was estimated.

In view of the rapid occurrence of the buckles for the small sandwich panels and the need for precision in locating the failure load, a more sophisticated system should be used for future tests. A more rugged system capable of automatically tracking the entire failure process should be considered.

The equations presented in references 7 and 8 treating flat rectangular panels with orthotropic facings and cores were used to predict the buckling stresses. These equations in terms of the symbols adopted in this report are as follows:

Equation 31 of reference 7:

$$P_{crs} = \frac{P_{cr}}{1 + \eta} \tag{1}$$

where

$$P_{cr} = (t^3 - t_c^3)\,\frac{T}{6} \tag{2}$$

$$T = \frac{\pi^2}{2\lambda_F a^2}\,(E_{Fa}\, b^2/a^2 + E_{Fb}\, a^2/b^2 + 2C) \tag{3}$$

$$C = E_{Fa}\,\mu_{Fab} + 2\lambda_F\, G_{Fab} \tag{4}$$

$$\eta = \frac{t_c\, t_F\, T}{K'} \tag{5}$$

$$K' = G_{Cbz} + G_{Caz}(b^2/a^2) \tag{6}$$

Equation 1 of reference 8 (symbols retained):

$$f_{Fcr} = \frac{\pi^2}{4} \frac{t_{F1}\, t_{F2}}{a^2} \left[\frac{t + t_c}{t - t_c}\right]^2 \frac{(E_{Fa}\, E_{Fb})^{1/2}}{\lambda_F} K \tag{7}$$

where

$$K = K_F + K_M \tag{8}$$

$$K_F = \frac{1}{3}\left[\frac{t_{F1}}{t_{F2}} + \frac{t_{F2}}{t_{F1}} - 1\right]\left[\frac{t - t_c}{t + t_c}\right]^2 \left[\frac{\alpha b^2}{a^2} + 2\beta + \frac{a^2}{\alpha b^2}\right] \tag{9}$$

$$K_M = \frac{\frac{\alpha b^2}{a^2} + 2\beta + \frac{a^2}{\alpha b^2} + VA\left[\frac{ra^2}{b^2} + 1\right]}{1 + V\frac{ra^2}{b^2}\left[\frac{\alpha b^2}{a^2} + \gamma\right] + V\left[\frac{a^2}{\alpha b^2} + \gamma\right] + V^2 rA\frac{a^2}{b^2}} \tag{10}$$

$$A = 1 - \beta^2 + \gamma\left[\frac{\alpha b^2}{a^2} + 2\beta + \frac{a^2}{\alpha b^2}\right] \tag{11}$$

$$V = \frac{t_c\, t_{F1}\, t_{F2}}{t - t_c} \frac{\pi^2}{a^2} \frac{(E_{Fa}\, E_{Fb})^{1/2}}{\lambda_F\, G_{Cbz}} \tag{12}$$

$$r = \frac{G_{Cbz}}{G_{Caz}} \tag{13}$$

$$\alpha = \left[\frac{E_{Fa}}{E_{Fb}}\right]^{1/2} \tag{14}$$

$$\beta = \alpha \mu_{Fab} + 2\gamma \tag{15}$$

$$\gamma = \frac{G_{Fab}\, \lambda_F}{(E_{Fa}\, E_{Fb})^{1/2}} \tag{16}$$

$$\frac{1}{G_{Fab}} = \frac{4}{E_{45}} - \frac{1 - \mu_{ab}}{E_{Fa}} - \frac{1 - \mu_{ba}}{E_{Fb}} \tag{17}$$

Calculated critical buckling stresses for comparison with test values were obtained from equation 7 since its derivation was more rigorous than that of equation 1. An approximate method was used to account for core shear in the case of equation 1.

A comparison of calculated values for equation 1 and equation 7 is shown in Table 1 on the following page (sample calculations are given in Appendix II). It must be noted that it was necessary to use approximations from the literature for E_{45} (facing modulus at 45 degrees to fabric warp), μ_F, and the ratio of facing strength in the warp and weave directions in order to make these calculations. Both methods yielded values which were close to each other, thus indicating that core shear plays only a small roll in the buckling of panels with the hinged boundary condition. The values calculated by equation 7 are plotted with the test values in Figure 24. Complete validation of the buckling theory is not deemed appropriate until further tests can be made with a more sophisticated side deflection instrumentation; however, it is noted that in most cases the calculated values were conservative (below the test values).

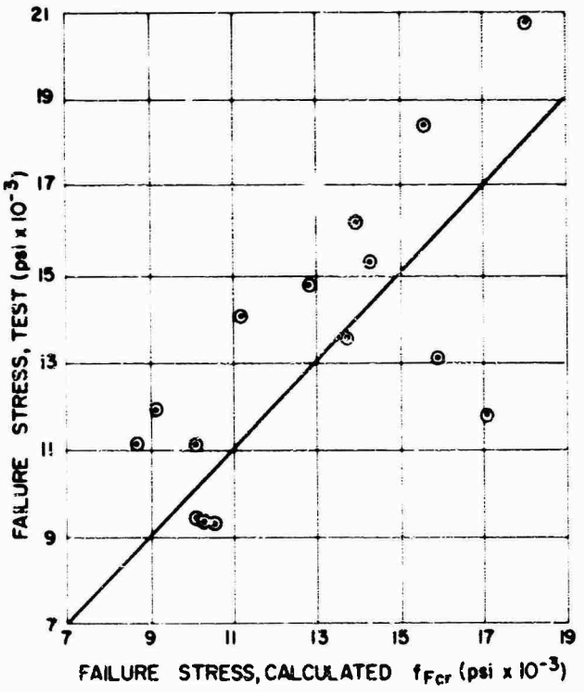

Figure 24. Comparison of Calculated and Test Values of Failure Stress for General Buckling of Sandwich Panels with All Edges Simply Supported.

It is very noticeable from the test data (Table 8) that for the dimensions and types of materials expected to be used, several values of general buckling stresses are much lower than for those of the other modes of failure. It is concluded that the other basic boundary conditions (hinged ends and clamped edges, clamped ends and hinged edges, and clamped ends and edges) should be investigated in greater detail. From the designer's as well as the analysist's point of view, the theoretical prediction of this mode of failure should be thoroughly backed with test data for structural sandwich of honeycomb cores and epoxy-fiberglass facings.

TABLE 1
CALCULATED VALUES,
GENERAL BUCKLING OF SANDWICH PANELS

Specimen Group Identification[3]	b/a	α	G_{Fab} (psi x 10^{6})	γ	β	A	V	r	K_M	K_F	K	f^1_{Fcr} (psi)	c^2 (psi x 10^{-6})	f^2_{Fcr} (psi)
12	0.8070	0.9735	0.7701	0.2260	0.5786	1.4265	0.03061	1.9459	3.1135	0.0149	3.1284	10,160	1.9380	10,200
12	0.8333	0.9735	0.7701	0.2260	0.5786	1.4138	0.04997	1.9459	3.2329	0.0146	3.2475	15,620	1.9313	15,720
20	0.7986	0.9735	0.7701	0.2260	0.5786	1.4310	0.01369	1.7556	3.2733	0.0145	3.2878	10,600	1.9380	10,610
20	0.8233	0.9733	0.7568	0.2123	0.5512	1.3921	0.04117	0.5696	3.0665	0.0141	3.0806	17,020	1.9380	17,150
22	0.8278	1.0272	0.7803	0.2365	0.6065	1.4215	0.08071	2.1220	2.7246	0.0074	2.7320	12,840	1.9535	13,060
22	0.8000	1.0272	0.7803	0.2365	0.6065	1.4343	0.05009	2.1220	2.9651	0.0075	2.9726	8,670	1.9535	8,770
22	0.8274	1.0272	0.7198	0.1743	0.4821	1.3061	0.14155	2.1220	2.2732	0.0124	2.2856	14,380	1.9691	14,670
22	0.7965	1.0272	0.7198	0.1743	0.4821	1.3167	0.08747	2.1220	2.5621	0.0127	2.5748	10,010	1.9691	10,150
22	0.8231	1.0272	0.7340	0.1889	0.5113	1.3347	0.16970	2.1220	2.1927	0.0199	2.2126	13,770	1.9671	14,060
22	0.8124	1.0272	0.7340	0.1889	0.5113	1.3385	0.11010	2.1220	2.4654	0.0200	2.4854	10,030	1.9671	10,190
23	0.8308	1.0272	0.7833	0.2396	0.6127	1.4260	0.03636	1.9355	3.0481	0.0067	3.0548	13,980	1.9574	14,080
23	0.7910	1.0272	0.7833	0.2396	0.6127	1.4450	0.02246	1.9355	3.2219	0.0069	3.2288	9,130	1.9574	9,170
23	0.8148	1.0272	0.7306	0.1855	0.5045	1.3191	0.09595	1.9355	2.5103	0.0200	2.5303	18,210	1.9539	18,430
23	0.7908	1.0272	0.7306	0.1855	0.5045	1.3373	0.05310	1.9355	2.8089	0.0207	2.8296	11,270	1.9539	11,350
23	0.7863	1.0272	0.7436	0.1987	0.5309	1.3681	0.16970	1.9355	2.9637	0.0146	2.9783	10,280	1.9528	10,330
23	0.8333	1.0272	0.7436	0.1987	0.5309	1.3494	0.11010	1.9355	2.7225	0.0142	2.7362	15,960	1.9528	16,110

[1] Calculated by Equation 7.

[2] Calculated by Equation 1.

[3] Entries In Same Order As Items Footnoted 7 In Table 8.

2. Face Wrinkling Tests

References 2 and 3 served as the theoretical basis for the face wrinkling investigations. The equations, which were developed from the theory of elasticity, are lengthy and will be repeated only as necessary to show the calculations. As stated in these papers, the practical face wrinkling problem is not one of instability; but rather one of progressive deformation due to initial irregularities and eccentricities in the facings. During edgewise compression of the sandwich, the irregularities of the faces increase gradually, thereby increasing the load on the core and glue line until failure occurs, at which time rapid facing deflection takes place to form the wrinkles.

It was estimated that during fabrication, the core cells influenced the formation of these irregularities. This reasoning appears sound, particularly for the single-step constructed FRP sandwich. Since in fabrication the facings were laminated by the pressure applied through the ends of the core cells, the facing thickness was less at these locations and greater over the center of the cells, although the outside surface was relatively flat. Thus, in effect, initial waves were built into the facings which were of a half-wave length equal to the core cell size. The same condition exists for the separately bonded sandwich but to a lesser degree.

On page 5 of reference 3, it was concluded that sandwich panels with honeycomb cores would wrinkle symmetrically; consequently, equations 9 and 10 of that report were used in the present study. The equations are repeated here in terms of the symbols adopted for this report.

$$f_{Fcr} = \frac{\pi^2 t_F^2 E_{Fb}}{12 L^2 \lambda_F} \quad \frac{t_c + \frac{24 E_{Cz} \lambda_F}{\pi^4 E_{Fb} t_F^3} L^4}{t_c + \frac{2 E_{Cz} A_o}{\pi F} L} \tag{18}$$

Introducing the parameters a, B, and b, the equation may be abbreviated as follows:

$$f_{Fcr} = \frac{B}{L^2} \quad \frac{t_c + aL^4}{t_c + bL} \tag{19}$$

These equations are in essence reductions of a more general equation developed in referenc 2. The application to sandwich of honeycomb core was made on the basis of an examination of test data for typical honeycomb cores--those with low values of elastic modulus perpendicular to the flute direction (E_{Cb}) as compared to that parallel to the flute direction (E_{Cz}) and to the shear modulus (G_{Cbz}). Stated mathematically, this condition prevails when the parameter κ is much less than 0.5 where:

$$\kappa = \frac{\sqrt{E_{Cb} E_{Cz}}}{2 G_{Cbz}} - \mu_{Cbz} \sqrt{\frac{E_{Cb}}{E_{Cz}}} \qquad (20)$$

It was stated that if κ is very small (say in the hundredths), L may be taken equal to the cell size of the core and the value of b calculated using test values of strength from specimens of a certain core thickness. Then it is possible to compute, by use of equation 19, the wrinkling stresses of specimens having other core thicknesses. Of couse, it is understood that the b obtained will apply only to sandwich having similar facings as regards the parameter F/A_o (the ratio of the glue-line strength to the ratio of the amplitude to half-wave length of the facing initial irregularities).

A spot check of κ for one of the cores used in this program (see sample calculation), readily confirmed, as expected, that the parameter is small for honeycomb cores suitable for aircraft construction. Even so, the theory was applied with reservation in that the predicted symmetrical wrinkle was not consistently obtained in the tests. The 1.5-inch-core specimens produced the best results in this regard, as is shown in Figure 25.

It is interesting to note that the final collapse of the specimens took place in various ways. As shown in the figure, those most predominate were face rupture, core-to-facing bond failure, and core crushing. In the beginning of the work, an effort was made to catalogue the final collapse of the specimens, and this information is recorded in the data tables for reference (see Figure 29 for the classifications). Observation during the testing of the specimens that collapse because of face rupture did not reveal positively that wrinkling was the cause of failure--the collapse was too abrupt. However, the stresses seem to agree with the remaining data indicating that face wrinkling did precipitate the failure.

Another observation that can be made from the figure is that the half-wave length of the wrinkles is not equal to the cell size of the core. Nevertheless, the theory was applied. One

Figure 25. Typical Face Wrinkling Failure of FRP Facing--Aluminum Honeycomb Core Sandwich. (Only the thicker specimens exhibited the fully developed symmetrical wrinkle.)

set of specimens from each of three classes of sandwich (class 1: specimen groups 17a and 19; class 2: specimen groups 1, 2, and 3; class 3: specimen groups 4, 5, 7, 8, and 10) was used to calculate the parameter b which contains the elusive parameter F/A_o, the ratio of bond strength to initial waviness. The choice of the particular group used for the b calculation was arbitrary. These values were then used according to equation 19 to predict the wrinkling stresses for the other cases of core thicknesses. The calculated values are displayed in Table 2 on the following page, and a sample calculation is presented in Appendix II. The particular test data used in the anaysis are noted in Table 2. Not all the data in Table 6 were suitable for use, in that column instability obviously preceded the critical wrinkling stress.

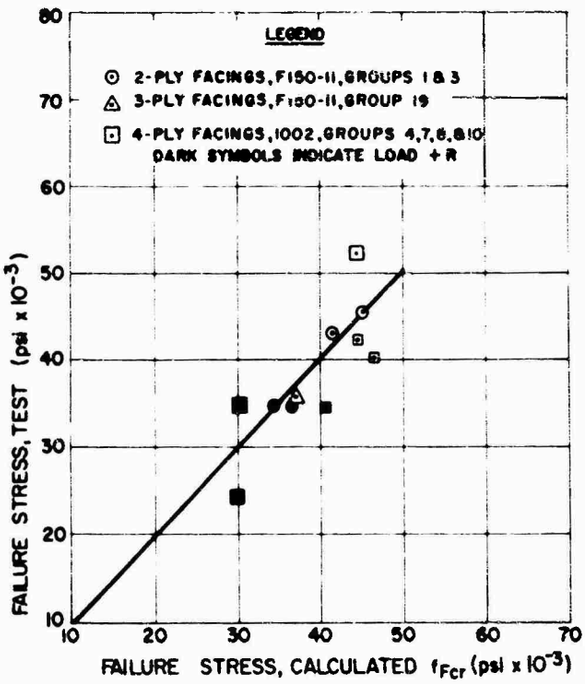

Figure 26. Comparison of Calculated and Test Values of Failure Stress for Face Wrinkling of Sandwich Panels.

The predictions plotted in Figure 26 indicate that reasonable agreement was obtained. No doubt the accuracy of the calculations would be improved by more accurate input data--more accurate values for material properties. At several points in the analysis, even with the large number of supporting tests that were conducted, it was necessary to estimate or use nominal values for properties. These are pointed out in the sample calculation and in the calculated data table.

It is particularily noticeable that the specimens consistently failed at lower stresses when the load was orientated 90 degrees to their core ribbon directions. This is true for the cross-ply as well as the 181-style fabric. It was possible to predict failure in this orientation of the core by an appropriate b calculation; however, the reason for the change is not clear. One possible answer is that the core crushes differently in this orientation as the wrinkles develop.

TABLE 2
CALCULATED VALUES,
FACE WRINKLING OF SANDWICH PANELS

Specimen Group Identification	Core Data			Fabrication Method	Facing Data			Load Orientation[4]	B	B_{Cs}/E_{Fb} (x 10^{-2})	e	b, Calculated	f_{Fcr}, Calculated (psi x 10^{-3})	f_{Fcr}, Tests (psi x 10^{-3})
	Cell Size (in.)	t_c (in.)	E_{Cs}[1] (psi x 10^{-4})		Material[2]	Nominal t_F (in.)	E_{Fb}[3] (psi x 10^{-6})							
17a	3/16	0.40	18.51	Separately Bonded	F150-11	0.03	3.81	=R	2868.73	4.85827	435.26	7.5801	-	41.8
19	3/16	0.20	18.51	Single-Step	F150-11	0.03	3.81	=R	2868.73	4.85827	435.71	-	37.14	35.2
2	1/4	0.75	26.70	Separately Bonded	F150-11	0.02	3.16	+R	1057.47	8.44937	2557.42	17.5908	-	35.3
							3.33	=R	1114.36	8.01802	2426.85	13.9671	-	43.0
1	1/4	1.00	26.70	Separately Bonded	F150-11	0.02	3.16	+R	1057.47	8.44937	2557.42	-	34.45	34.6
							3.33	=R	1114.36	8.01802	2426.85	-	41.60	43.0
3	1/4	0.40	26.70	Separately Bonded	F150-11	0.02	3.16	+R	1057.47	8.44937	2557.42	-	36.64	34.6
							3.33	=R	1114.36	8.01802	2426.85	-	45.26	45.4
5	3/8	1.00	9.26	Separately Bonded	1002	0.04	3.70	+R	4952.73	2.50270	94.54	6.1118	-	30.7
							3.70	=R	4952.73	2.50270	94.54	3.5145	-	43.6
4	3/8	1.50	9.26	Separately Bonded	1002	0.04	3.70	+R	4952.73	2.50270	94.54	-	40.58	34.1
7	3/8	0.75	9.26	Separately Bonded	1002	0.04	3.70	+R	4952.73	2.50270	94.54	-	30.33	34.5
							3.70	=R	4952.73	2.50270	94.54	-	44.61	52.0
8	3/8	0.75	9.26	Separately Bonded	1002	0.04	3.70	=R	4952.73	2.50270	94.54	-	44.61	42.1
10	3/8	0.40	9.26	Separately Bonded	1002	0.04	3.70	+R	4952.73	2.50270	94.54	-	29.70	24.0
							3.70	=R	4952.73	2.50270	94.54	-	46.53	40.0
17a	3/16	0.40	18.51	Separately Bonded	F150-11	0.02	3.33	=R	1114.36	5.55856	1682.45	6.9370	-	44.4
17b	3/16	0.40	18.51	Single-Step	1002	0.02	3.70	=R	1230.18	5.00270	1514.28	6.2830	-	41.5
13	1/4	0.75	13.88	Single-Step	1002	0.04	3.70	+R	4952.73	3.75135	141.70	17.3030	-	27.0
							3.70	=R	4952.73	3.75135	141.70	10.8651	-	29.8
14	1/4	0.40	13.88	Single-Step	1002	0.02	3.70	=R	1230.18	3.75135	1135.45	8.2883	-	38.5
5	3/8	1.00	9.26	Separately Bonded	F150-11	0.04	3.73	+R	4992.89	2.48257	93.78	6.4639	-	29.6
							3.94	=R	5273.98	2.35025	88.78	4.6824	-	37.5
6	3/8	0.75	9.26	Single-Step	1002	0.04	3.70	+R	4952.73	2.50270	94.54	9.3900	-	21.6
							3.70	=R	4952.73	2.50270	94.54	6.2836	-	29.7
							3.70							
6	3/8	0.75	9.26	Single-Step	F150-11	0.03	3.61	+R	2718.14	2.56510	230.07	10.0337	-	22.78
							3.81	=R	2868.73	2.43045	217.93	7.8293	-	30.2
11	3/8	0.20	9.26	Single-Step	F150-11	0.04	3.94	=R	5273.98	2.35025	88.78	5.5786	-	32.0
24	7/16	0.75	4.90*	Single-Step	OUR1	0.02	3.24	=R	1084.25	1.51235	457.76	6.3963	-	28.0*
26	7/16	0.50	4.30*	Single-Step	OUR1	0.02	3.24	=R	1084.25	1.32716	401.71	4.6413	-	34.1*
26	7/16	0.50	4.30*	Single-Step	OUR1	0.03	3.75	=R	2823.56	1.11581	100.07	2.8632	-	35.1*

[1] Values were calculated by the equation given on page 6 in reference 4, except those marked with an asterisk which were taken from the Douglas Aircraft Company brochure titled "Aircomb, Test and Technical Data," dated October 1963 (Figure 3).

[2] See "Fabrication Process and Evaluation" for Pre-Preg Identification.

[3] Values of modulus for facings of 181 fabric are average values obtained from Table 15 and used directly or transformed by a ratio of properties parallel to warp to those perpendicular to warp obtained from reference 8 (see sample calculations). The facing modulus for the 1002 Scotchply was obtained from the Minnesota Mining and Manufacturing Company Technical Data Sheet, dated December 1957.

[4] The symbols +R and =R indicate perpendicular and parallel to the core ribbon direction, respectively.

[5] Test values taken from Table 5 except in the following cases: The values underlined are from Table 6, the value suffixed with the letter A is the average of the data in Tables 5 and 6, and the values marked with an asterisk are from Table 4.

Perhaps the core shear modulus should appear, explicitly, in the stress equation. This is the case for the equations given in references 14 and 15 for the thick-core sandwich. Thus, it appears that the application of such theory would be beneficial in the analysis of FRP-honeycomb core sandwich and should be considered for future work.

Viewing the presently considered theory, it is apparent that (within the range of b values encountered in the present study) the sandwich can be expected to carry higher edge loads as the thickness of the core is decreased, at least to the point where shear instability or column instability becomes a problem. The calculated data in the first half of Table 2 shows this trend.

The latter part of Table 2 is devoted to the calculation of values of the parameter b for the purpose of observing the character of the parameter for the FRP constructions. The calculated values of b were plotted against core cell size in Figure 27 for this study. The replication of panels is not sufficient to establish any functional relationships; however, there are several valuable observations that can be made. First, note that the parameter b for the cross-ply facings tends to be lower than for the woven fabric, suggesting that there is less initial waviness present.

Next, it is seen that, for a given cell size and facing material, parameter b is greater when the specimens were loaded perpendicular to the core ribbon (+R) and greater also for panels fabricated by the single-step method. There is one exception to the former observation. This is found in specimen group 13. Here the reverse situation is true--the parameter b is greatest for loading parallel to the core ribbon direction (=R). It was previously noted that the bond strength was low for this group (see page 33); hence, the anomaly is attributed to inconsistent core-to-facing bond.

Another observation is that b varies inversely as facing thickness. Actually, in view of physical considerations, this and the two previously mentioned observations are as anticipated.

The most interesting aspect of the plotted values of b is the maximum that appears to occur at the 1/4-inch cell size. The initial waviness of the FRP facings in sandwich construction probably accounts for this phenomenon; yet, the data should be much more extensive before a definite conculsion can be drawn and certainly before a b-function can be isolated.

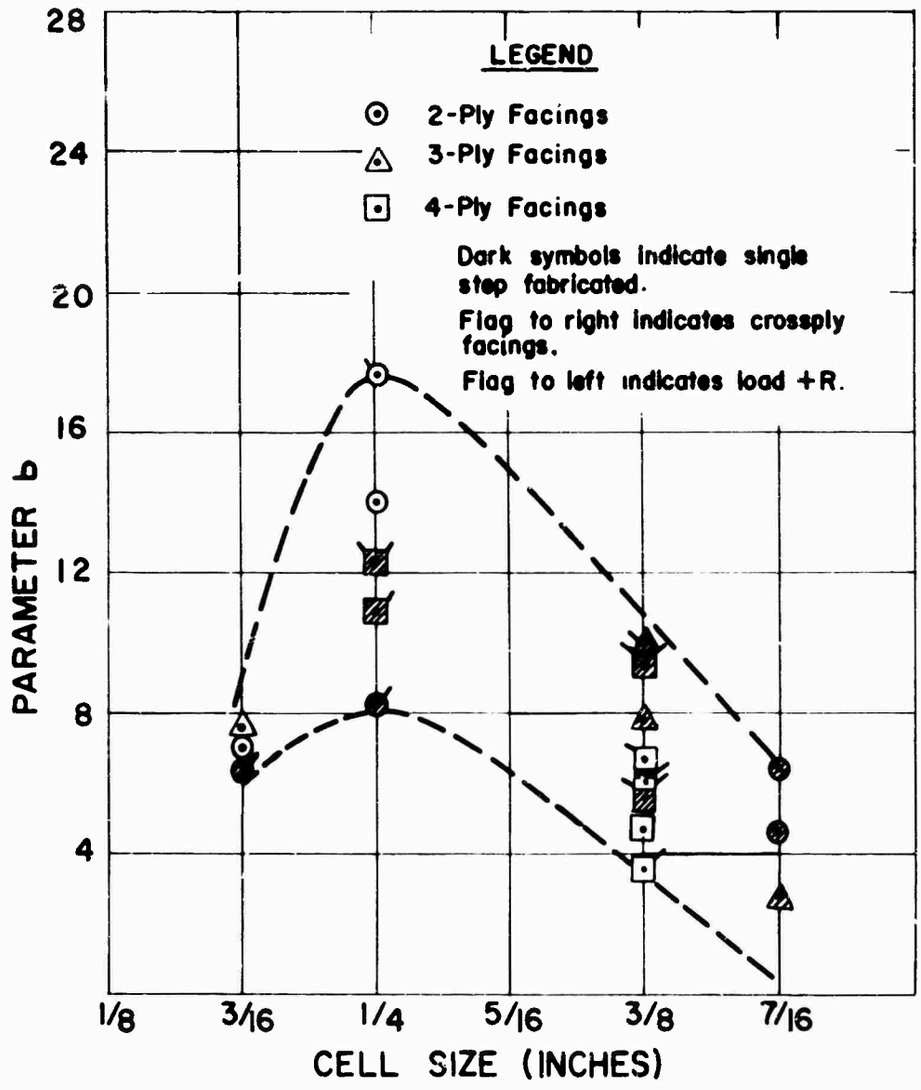

Figure 27. Relation Between Calculated Values of Face Wrinkling Parameter b and Core Cell Diameter.

In consideration of the discrepancies between theory and test noted in this study, the theory of reference 3 should be used with care until further substantiation can be accomplished. Indeed, there is a need for more extensive research to be done regarding the facing wrinkling phenomenon. Other theories should be examined and additional tests conducted until the failure can be predicted with confidence.

3. Shear Crimping Tests

According to reference 3, the face wrinkling analysis also provides a criterion for shear instability. Shear failure is expected when the test value is greater then the value of stress calculated by the following equation:

$$f_{Fcr} = \frac{t_c}{2t_F} G_{Cbz} \qquad (21)$$

The specimens for these tests, most of which were from the vacuum press early in the fabrication effort, were constructed of commercially available core materials. As predicted by the equation, even these paper cores were a bit too still to permit shear instability to develop. There was no evidence of the classical shear crimp. The sample calculation given in Appendix II shows that shear was a possibility for specimen group 25, had the facings been of sufficient strength. It appears, however, that the facings ruptured first on groups 25 and 27. Of course, this type of failure was of a catastrophic nature ultimately involving the core and the core-to-facing bond.

Evidence of dimpling and face wrinkling was noticed in specimen groups 24 and 26, and hence, these were considered in the face wrinkling analysis.

Shear instability doesn't appear to be of concern in sandwich of thickness suitable for aircraft structures; however, to cover the special or unforeseen applications, further tests should be accomplished to insure that this mode of failure can be accurately predicted.

4. Intracellular Buckling Tests

A search of the literature revealed no theoretical analysis of the phenomenon of intracellular buckling (face dimpling) of honeycomb core sandwich. An empirical approach to the problem was taken by the investigators of reference 9. The test data obtained in this project were compared with the equation given in that report.

Equation of reference 9:

$$f_{Fcr} = (E_r/3)\ (t_F/R)^{3/2} \tag{22}$$

where

$$E_r = \frac{4\ E_{Fb}\ E_t}{(\sqrt{E_{Fb}} + \sqrt{E_t})^2} \tag{23}$$

A battery of dial gages was used to detect the dimpling phenomenon and to locate the load at which failure occurred. Opposing dial gages (Figure 18) were used to detect the movement of the facing over the core cell opening relative to that over the core cell wall and/or to detect transverse movement of the specimen. As in the case of general buckling, the dimpling of the facings was discovered to occur rather rapidly; and to prevent their damage, the dial gages could not be left in position consistently to trace completely the history of the side deflection, especially near the critical load. In addition, there was a multitude of cells where dimpling could occur in each panel. It was not possible to monitor more than two cells because of the physical size of the dial gage setup. An automatic monitoring system could be designed to track the deflections on a number of cells simultaneously, but such an elaborate system was not possible in this pilot study. Thus, it is possible that the cell being tracked may not have buckled while an unmonitored cell did.

The collapse took place suddenly as a core failure or a core-to-facing bond failure or a combination of the two. In only a few cases was it possible to record the characteristic load versus side-deflection curve. There were two such cases with the following combinations: 3/4-inch cell, 3-ply facing; and 3/4-inch cell, 2-ply facing. The best curve is illustrated in Figure 19 with the combination of the 3/4-inch cell, 3-ply facing.

The investigation was further complicated by the difficulty of obtaining the required large-cell core material. The commercially available material--Kraft paper core--was very nonuniform in cell size and shape. This condition, no doubt, has a great influence on the test results.

The three cases for which dimpling could reasonably be identified are plotted on a graph of equation 22 (Figure 28). Because of the lack of accurate values of tangent modulus (E_t) for thin laminates, an average value was used

to plot the curves rather than a reduced value (E_r). Where test values of modulus for the desired orientation of warp direction were not available, values were calculated from ratios obtained from reference 8 for 181- and 143-style fabric (see sample calculation). To lessen the effect of the deformed core cells, the cell radius (R) for the parameter t_F/R was obtained by averaging four measurements made on each specimen.

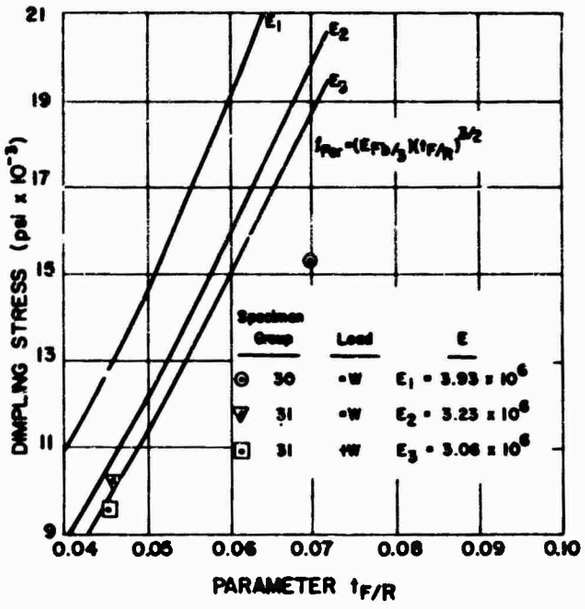

Figure 28. Intracellular Buckling of Sandwich Panels. (Comparison of Test Data with an Empirical Equation.)

It is concluded from this pilot investigation that intracellular buckling could be a problem for cell sizes of 1/2 inch and larger in combination with 3-ply and thinner facings. A more extensive investigation, including better, more elaborate instrumentation would establish the trends in detail. A better theory as an improvement upon the present empirical relationship should be attemped in connection with the experimental investigation.

CONCLUSIONS AND RECOMMENDATIONS

The major conclusions and recommendations drawn from the research are as follows:

1. For the single-step method of sandwich construction, room temperature B-staging of the resin impregnated facings and the use of a separate precure phase (a gellation period) in the resin cure cycle were found beneficial in controlling the resin flow from the facings during fabrication. It is recommended that these techniques be employed in the fabrication of fiberglass-reinforced plastics.

2. The response of the resin-curing-agent mixture to low pressures is important in the single-step method of sandwich construction by the vacuum blanket technique. Twenty inches of mercury is a safe upper limit of vacuum to prevent bubbling of EPON 828-Z resin in the 200-degree-Fahrenheit Temperature range. It is recommended that cure pressures be limited to the 20-inch mercury vacuum.

3. Sandwich construction with FRP materials is quite versatile, and it is believed that combinations of the single-step and the separately-bonded methods may be employed to achieve any desired balance between strength properties, mold shape, and economy. Hence, it is recommended that the optimum method of sandwich construction for vital parts of an aircraft structure be determined through suitable research, giving due regard to such items as surface smoothness, compound curvature, severity of loading, and molding time and cost.

4. The construction of sandwich by the single-step method is limited by the influence of the conditions of facing cure on the core material, especially the metallic cores.

5. Postcure of the facings may be achieved in the mold or press during the bonding of the facings to the core when high curing temperatures are required for the adhesive.

6. The filleting of the core-to-facing adhesive can be expected to increase both the flatwise tensile strength and the shear properties of the core as the thickness of the core becomes small.

7. The irregularities and eccentricities built into the facings of the single-step constructed sandwich tend to cause premature buckling failures. The separately-bonded type of sandwich construction eliminates this tendency to a great extent by providing smoother facings of more uniform thickness.

8. A characteristic of thin laminates of 181-style fiberglass fabric and epoxy resin is a decrease in the strength properties with decrease in thickness when the thickness is largely established by the number of plies of fabric.

9. More extensive and sophisticated instrumentation than dial gages is needed to record side deflection in order to establish the failure load for general buckling and intracellular buckling of small, flat sandwich panels. In this pilot investigation of the general buckling of small sandwich panels, the above condition lessened the benefit of excellent test fixtures for panels supported on hinges on all edges, and therefore, additional testing is required for the all around hinged restraint before the adequacy of the theory can be definitely stated.

10. Further, extensive and detailed testing is needed to confirm the theory of general buckling of flat, FRP facing-honeycomb core

sandwich panels restrained according to the additional basic edge conditions: simply supported loaded ends and clamped sides, clamped loaded ends and simply supported sides, and clamped loaded ends and clamped sides. The observations made in this project indicate that general buckling may be the critical mode of failure in normal applications. Therefore, it is recommended that these tests be conducted in the near future.

11. Contrary to some predictions, the facings of sandwich of honeycomb cores will not consistently wrinkle in the symmetrical fashion nor are the half-wave lengths of the wrinkles necessarily equal to the core cell size. Nevertheless, the theory of reference 3 yielded reasonable values of calculated failure stress in the limited study made in this report and should be investigated in greater detail--this is particularly true regarding the character of the parameter b for FRP facings. It is recommended that more extensive tests be conducted to substantiate the theory of reference 3.

12. Higher face wrinkling stresses result when the edgewise loads are applied parallel to the core ribbon direction of honeycomb core sandwich, indicating that the core shear properties play a part in the face wrinkling phenomenon. Therefore, it is recommended that tests according to the theory advanced in references 14 and 15 also be conducted in conjunction with the work outlined in paragraph 11 of this section.

13. Shear crimping should not be a problem for honeycomb core sandwich except for thin, flat panels employing cores of very low shear modulus (say less than 5,000 psi). However, for these special applications the existing theory should be confirmed by tests in which the shear failure actually occurred. It is recommended that additional testing be done to achieve the shear failure. Cores of materials other than Kraft paper would probably give more predictable and identifiable results.

14. The tests reported herein indicate that intracellular buckling probably will be important for cell sizes equal to or exceeding 1/2 inch in combination with 3-ply, or thinner, facings. Additional work in this area is, therefore, recommended. The intracellular buckling phenomenon should be established on a firm theoretical basis and confirmed with precision tests.

15. Fabrication of structures with curvatures and investigation of the effect of curvature on the basic failure modes should be accomplished as the next step in the evaluation of fiberglass-reinforced plastics for aircraft use. Consequently, in combination with the work suggested in paragraph 3, it is recommended that work on curved panels and simple geometic shapes be initiated as soon as possible.

REFERENCES

1. Research in the Field of Fiberglass-Reinforced Sandwich Structure for Airframe use, TCREC Technical Report 64-37, U. S. Army Transportation Research Command, Fort Eustis, Virginia, July 1964.

2. Norris, Charles B., Wrinkling of the Facings of Sandwich Constructions Subjected to Edgewise Compression, Report Number 1810, Forest Products Laboratory, Madison, Wisconsin, April 1961.

3. Norris, Charles B., Wrinkling of the Facings of Sandwich Construction Subjected to Edgewise Compression, Sandwich Constructions Having Honeycomb Cores, Report Number 1810 A, Forest Products Laboratory, Madison, Wisconsin, June 1953.

4. Kuenzi, Edward W., Mechanical Properties of Aluminum Honeycomb Cores, Report Number 1849, Forest Products Laboratory, Madison, Wisconsin, September 1955.

5. Boller, K. H., Buckling of Flat Sandwich Panels in Compression, the Buckling of Flat Sandwich Panels with Edges Simply Supported, Report Number 1525 A, Forest Products Laboratory, Madison, Wisconsin, January 1954.

6. Boller K. H., Buckling Loads of Flat Sandwich Panels in Compression, Buckling of Flat Sandwich Panels with Either All Edges Simply Supported or All Edges Clamped, Report Number 1525 E, Forest Products Laboratory, Madison, Wisconsin, March 1948.

7. March, H. W., Effects of Shear Deformation in the Core of A Flat Rectangular Sandwich Panel, Report Number 1583, Forest Products Laboratory, Madison, Wisconsin, August 1955.

8. Norris, Charles B., Compressive Buckling Curves for Simply Supported Sandwich Panels with Glass-Fabric-Laminate Facings And Honeycomb Cores, Report Number 1867, Forest Products Laboratory, Madison, Wisconsin, December 1958.

9. Norris, C. B., and W. J. Kommers, Short-Column Compressive Strength of Sandwich Constructions as Affected by the Size of the Cells of Honeycomb-Core Materials, Report Number 1817, Forest Products Laboratory, Madison, Wisconsin, March 1956.

10. "Sandwich Construction and Core Materials; General Test Methods", MIL-STD-401A, Military Standard, June 1956.

11. "Plastics for Flight Vehicles, Part 1, Reinforced Plastics", MIL-HDBK-17, Military Handbook, November 1959.

12. Ericksen, Wilhelm S., and H. W. March, Compressive Buckling of Sandwich Panels Having Dissimilar Facings of Unequal Thickness, Report Number 1583 B, Forest Products Laboratory, Madison, Wisconsin, November 1958.

13. Youngs, Robert L., Poisson's Ratios for Glass-Fabric-Base Plastic Laminates, Report Number 1860, Forest Products Laboratory, Madison, Wisconsin, January 1957.

14. Yusuff, S., "Theory of Wrinkling in Sandwich Construction", Journal of the Royal Aeronautical Society, Volume 59, January 1955, pp. 30-36.

15. Yusuff, S., "Face Wrinkling and Core Strength in Sandwich Construction", Journal of the Royal Aeronautical Society, Volume 64, March 1960, pp. 164-167.

APPENDIX I

TABULATIONS OF TEST RESULTS

TABLE 3
SANDWICH IDENTIFICATION CODE

Specimen Group Identification	Cell Size (in.)	Core Thick. (in.)	Cell Wall Thickness (in.)	Core Type	Intermediate Adhesive[5]	Fabrication Method
1	1/4	1.00	0.0025	5052[1]	AF-110B	Separately Bonded
2	1/4	0.75	0.0025	5052	AF-110B	Separately Bonded
3	1/4	0.40	0.0025	5052	AF-110B	Separately Bonded
4	3/8	1.50	0.0013	5052	AF-110B	Separately Bonded
5	3/8	1.00	0.0013	5052	AF-110B	Separately Bonded
6	3/8	0.75	0.0013	5052	None	Single-Step
7	3/8	0.75	0.0013	5052	AF-110B	Separately Bonded
8	3/8	0.75	0.0013	5052	EC-1595	Separately Bonded
9	3/8	0.40	0.0013	5052	EC-1595	Single-Step
10	3/8	0.40	0.0013	5052	AF-110B	Separately Bonded
11	3/8	0.20	0.0013	5052	EC-1595	Single-Step
12	3/8	0.20	0.0013	5052	AF-110B	Separately Bonded
13	1/4	0.75	0.0013	5052	EC-1595	Single-Step
14	1/4	0.40	0.0013	5052	EC-1595	Single-Step
15	3/16	0.75	0.0013	5052	None	Single-Step
16	3/16	0.75	0.0013	5052	EC-1595	Single-Step
17a	3/16	0.40	0.0013	5052	EC-1595	Separately Bonded
17b	3/16	0.40	0.0013	5052	EC-1595	Single-Step
18	3/16	0.40	0.0013	5052	None	Single-Step
19	3/16	0.20	0.0013	5052	EC-1595	Single-Step
20	3/16	0.20	0.0013	5052	AF-110B	Separately Bonded
21	3/8	0.40	-	HRP-CF11[2]	AF-110B	Separately Bonded
22	3/8	0.20	-	HRP-CF11	AF-110B	Separately Bonded
23	3/16	0.20	-	HRP-CF11	AF-110B	Separately Bonded
24	7/16	0.75	-	125-35-20[3]	None	Single-Step
25	7/16	0.75	-	60-20-40	None	Single-Step
26	7/16	0.50	-	125-35-20	None	Single-Step
27	7/16	0.50	-	60-20-40	None	Single-Step
28	7/16	0.25	-	60-20-40	None	Single-Step
29	3/4	3/4	-	KP-99-18[4]	A-12	Separately Bonded
30	3/4	3/4	-	KP-99-18	A-12	Separately Bonded
31	3/4	3/4	-	KP-99-18	A-12	Separately Bonded
32	1/2	3/4	-	KP-99-18	A-12	Separately Bonded
33	1/2	3/4	-	KP-99-18	A-12	Separately Bonded
34	1/2	3/4	-	KP-99-18	A-12	Separately Bonded
35	1/2	3/4	-	KP-99-18	A-12	Separately Bonded
36	3/8	1	-	KP-99-18	A-12	Separately Bonded
37	1/4	3/4	-	KP-99-18	A-12	Separately Bonded

[1] Hexcel Products, Inc. 5052-Aluminum, Hexagonal Cells with Perforated Walls.

[2] Hexcel Products, Inc. Heat Resistant Phenolic (HRP) Impregnated Fiberglass, Hexagonal Cells.

[3] Douglas Aircraft Company Aircomb Paper, Hexagonal Cells.

[4] Hexcel Products, Inc. Kraft Paper KP-99-18 (18 Denotes Percent Phenolic Impregnation), Hexagonal Cells.

[5] See "Fabrication Process and Evaluation" for Description.

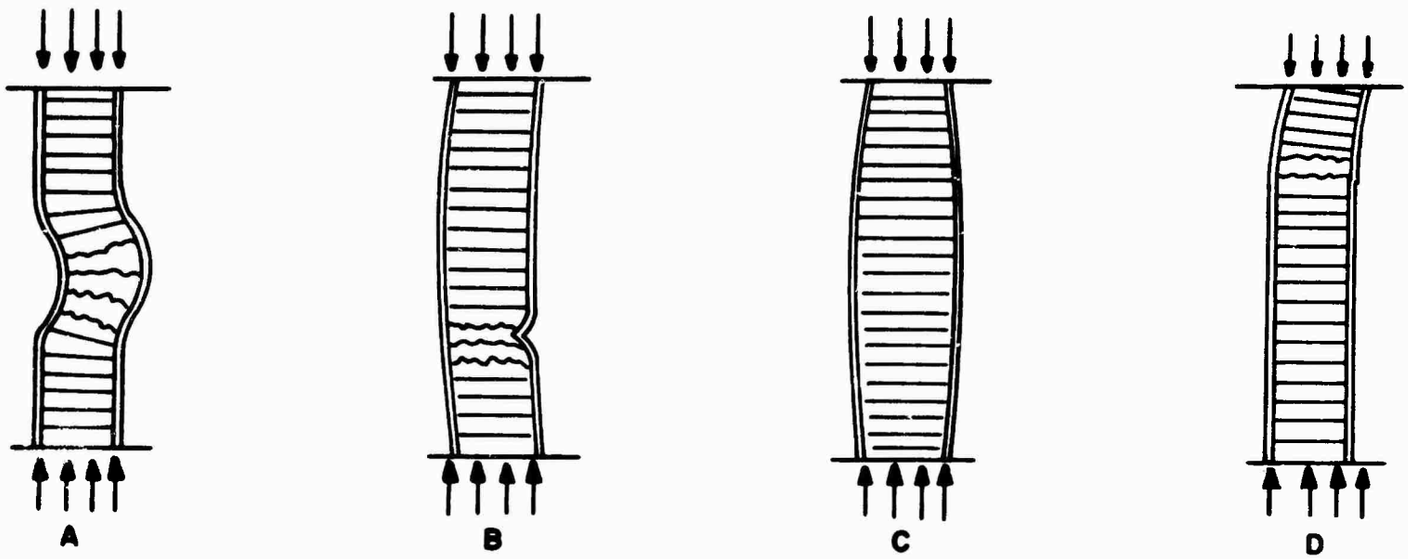

Figure 29. Types of Panel Collapse. (A and B occurred either in the absence of facing rupture or with numerous combinations of complete and partial rupture of the facings. C occurred as core-to-facing bond failure between one or both facings. D occurred as either complete or partial rupture of the facings.)

TABLE 4
SHEAR CRIMPING TEST DATA,
CLAMPED LOADED ENDS AND FREE SIDES

Specimen Group Identification	Facing Material, Thickness	Loading Configuration: Loaded Edge (in.)	Loaded Length (in.)	Load Orientation[1]	Type of Collapse[2]	Ultimate Stress: Average Stress (psi x 10^{-3})	Range of Stress (psi x 10^{-3})	Number of Specimens	Modulus (psi x 10^{-6}): Sandwich Average Modulus	Range of Modulus	Number of Specimens
6	F150-11[3], 3-Ply	6.0	5.5	=W, =R	2A	29.9	29.6-30.2	2	-	-	-
13	1002, 4-Ply	6.0	5.5	-, =R	1D, 2D	30.0	27.1-32.8	3	3.19	-	1
13	1002, 4-Ply	6.0	5.5	-, +R	2D	31.5	29.7-33.3	2	3.47	-	1
24[4]	181 OURI, 2-Ply	6.0	3.5	=W, =R	-	28.0	23.2-32.3	2	2.53	-	1
25[4]	181 OURI, 3-Ply	6.0	5.5	+W, +R	-	30.7	26.9-33.1	4	3.00	2.88-3.13	2
25[4]	181 OURI, 2-Ply	6.0	3.5	=W, =R	-	31.1	30.5-32.4	2	2.23	-	1
26[4]	181 OURI, 3-Ply	6.0	5.5	=W, =R	-	34.8	33.6-36.0	2	2.79	-	1
26[4]	181 OURI, 2-Ply	6.0	3.5	=W, =R	-	34.1	31.8-35.2	4	2.40	2.31-2.49	2
26[4]	181 OURI, 3-Ply	6.0	3.5	=W, =R	-	35.4	32.9-37.0	3	-	-	-
27[4]	181 OURI, 3-Ply	6.0	3.5	=W, =R	-	29.7	27.5-32.2	4	3.13	3.10-3.16	2
27[4]	181 OURI, 3-Ply	6.0	3.5	=W, =R	-	32.2	-	1	-	-	-

[1] This symbol is used to denote whether the Fabric weave (W) was perpendicular (+) or parallel (=) to the core ribbon direction(R).

[2] The number refers to the number of specimens of a given type of collapse and the letter refers to the type of collapse as presented in Figure 29.

[3] See "Fabrication Process and Evaluation" for Pre-Preg Identifications.

[4] Fabricated on Vacuum Press.

TABLE 5
FACE WRINKLING TEST DATA,
CLAMPED LOADED ENDS AND FREE EDGES

Specimen Group Identification	Facing Material, Thickness	Loading Configuration: Loaded Edge (in.)	Loaded Length (in.)	Load Orientation[1,4]	Type of Collapse[2]	Ultimate Stress (psi x 10^{-3}): Average Stress	Range of Stress	Number of Specimens	Modulus (psi x 10^{-6}): Sandwich Average Modulus	Range of Modulus	Number of Specimens
1	F150-11[3], 2-Ply	4.0	3.0	-, =R	5D	43.0	41.5-44.1	5	1.85	1.82-1.87	2
1	F150-11, 2-Ply	4.0	3.0	-, +R	5D	34.6	33.3-35.6	5	1.70	1.65-1.79	3
2	F150-11, 2-Ply	4.0	3.0	=W, =R	5D	43.0	41.2-45.3	5	2.03	2.02-2.05	3
2	F150-11, 2-Ply	4.0	3.0	+W, +R	5D	35.3	32.8-37.5	5	1.77	1.61-2.05	3
3	F150-11, 2-Ply	4.0	3.0	-, =R	3A, 2D	45.4	39.6-50.4	5	1.92	1.64-2.25	3
3	F150-11, 2-Ply	4.0	3.0	-, +R	5D	34.6	30.8-36.5	5	-	-	-
4	1002[3], 4-Ply	4.0	3.0	-, +R	10A	34.1	27.7-38.2	10	1.55	1.35-2.10	5
5	1002, 4-Ply	4.0	3.0	-, +R	5A	30.7	29.1-33.3	5	1.67	1.62-1.73	3
5	1002, 4-Ply	4.0	3.0	-, =R	4A	43.6	40.9-48.9	4	1.55	-	1
5	F150-11, 4-Ply	4.0	3.0	=W, =R	6A	37.5	35.7-39.2	6	1.48	1.09-1.68	3
5	F150-11, 4-Ply	4.0	3.0	+W, +R	6A	29.6	21.4-34.8	6	1.44	1.19-1.72	3
6	F150-11, 3-Ply	4.0	3.5	=W, =R	1D	30.3	-	1	-	-	-
8	1002, 4-Ply	4.0	3.0	-, =R	1A, 1C, 3D	42.1	37.7-48.4	5	3.40	3.26-3.67	3
7	1002, 4-Ply	4.0	3.0	-, +R	-	34.5	30.4-37.0	5	3.18	2.44-3.91	2
7	1002, 4-Ply	4.0	3.0	-, =R	-	52.0	49.1-54.8	4	3.07	2.36-3.77	2
10	1002, 4-Ply	4.0	3.0	-, +R	5A	24.0	20.9-28.7	5	2.58	2.12-3.28	3
10	1002, 4-Ply	4.0	3.0	-, =R	5A	40.0	35.0-44.8	5	3.62	3.57-3.67	2
11	F150-11, 4-Ply	4.0	3.5	=W, =R	3D	32.0	29.6-33.3	3	3.51	-	1
14	1002, 2-Ply	4.0	3.5	-, =R	1B, 2D	39.5	33.8-43.5	3	3.67	-	1
17a	F150-11, 3-Ply	4.0	3.0	=W, =R	8D	41.8	37.9-46.0	8	3.27	3.01-3.62	4
17a	F150-11, 2-Ply	4.0	3.0	=W, =R	1A, 1C, 1D	44.4	39.6-48.1	6	3.19	3.05-3.29	3
17b	1002, 2-Ply	4.0	3.5	-, =R	6D	41.5	38.7-44.7	6	3.40	3.26-3.54	3
19	F150-11, 3-Ply	4.0	3.5	=W, =R	1A	35.2	-	1	-	-	-
19	F150-11, 3-Ply	4.0	3.5	+W, +R	1A	41.7	-	1	-	-	-

[1] See Note 1 of Table 4.

[2] See Note 2 of Table 4.

[3] See Note 3 of Table 4.

[4] Dash Indicates Orientation Indeterminate.

TABLE 6
FACE WRINKLING TEST DATA,
HINGED LOADED ENDS AND FREE EDGES

Specimen Group Identification	Facing Material, Thickness	Loading Configuration: Loaded Edge (in.)	Loading Configuration: Loaded Length (in.)	Loading Configuration: Load Orientation[1]	Type of Collapse[2]	Ultimate Stress (psi x 10^{-3}): Average Stress	Ultimate Stress (psi x 10^{-3}): Range of Stress	Ultimate Stress (psi x 10^{-3}): Number of Specimens	Modulus (psi x 10^{-6}): Sandwich Average Modulus	Modulus (psi x 10^{-6}): Range of Modulus	Modulus (psi x 10^{-6}): Number of Specimens
6	1002[3], 4-Ply	6.0	5.5	-, +R	1A, 1C	21.6	20.8-22.3	2	-	-	-
6	1002, 4-Ply	6.0	5.5	-, =R[4]	2D	29.7	26.7-32.7	2	-	-	-
6	F150-11, 3-Ply	4.0	3.5	=W, =R[4]	3B	21.8	20.3-22.9	3	-	-	-
6	F150-11, 3-Ply	4.0	3.5	+W, +R[4]	3B	20.4	20.0-20.8	3	-	-	-
6	F150-11, 3-Ply	4.0	3.5	=W, =R[4]	3D	28.3	27.2-29.2	4	-	-	-
6	F150-11, 3-Ply	4.0	3.5	+W, +R[4]	1B	25.0	-	1	-	-	-
11	F150-11, 4-Ply	4.0	3.0	=W, =R	2D	14.2	11.4-17.0	2	4.43	-	1
13	1002, 4-Ply	4.0	3.5	-, +R	3D	27.0	22.2-31.2	3	3.13	-	1
13	1002, 4-Ply	4.0	3.5	-, =R	3D	29.8	28.5-31.2	3	3.24	-	1
13	1002, 4-Ply	6.0	5.5	-, =R	2D	26.7	26.3-27.1	2	-	-	-
13	1002, 4-Ply	6.0	5.5	-, +R	1D	22.3	-	1	-	-	-
14	1002, 2-Ply	4.0	3.5	-, =R	2D	37.5	37.5-37.6	2	3.37	-	1
18	1002, 4-Ply	6.0	5.5	-, +R	2A	21.4	21.1-21.8	2	-	-	-
19	F150-11, 3-Ply	4.0	3.5	=W, =R	2A	19.9	17.6-22.1	2	-	-	-
19	F150-11, 3-Ply	4.0	3.5	+W, +R	2A	18.5	18.0-19.0	2	-	-	-

[1] See Note 1 of Table 4.

[2] See Note 2 of Table 4.

[3] See Note 3 of Table 4.

[4] Visual inspection revealed damage to Core Cell Walls of some of the specimens in this group.

TABLE 7
GENERAL PANEL BUCKLING TEST DATA,
CLAMPED LOADED ENDS AND HINGED EDGES

Specimen Group Identification	Facing Material, Thickness	Loading Configuration: Loaded Edge (in.)	Loaded Length (in.)	Load Orientation[1]	Type of Collapse[2]	Ultimate Stress (psi x 10^{-3}): Average Stress	Range of Stress	Number of Specimens	Modulus (psi x 10^{-6}): Sandwich Average Modulus	Range of Modulus	Number of Specimens
9	F150-11[3], 3-Ply	9.0	8.0	=W, =R	3D	33.6	32.3-35.3	3	3.40	3.25-3.67	3
9	F150-11, 2-Ply	12.5	12.0	=W, =R	2D	24.8	24.6-25.0	2	4.16	4.15-4.17	2
11	F150-11, 4-Ply	9.0	8.0	=W, =R	2A	25.3	25.2-25.5	2	3.28	3.18-3.38	2
11	F150-11, 3-Ply	9.0	8.0	=W, =R	6A	23.7	23.0-24.5	6	3.16	2.95-3.34	6
14	1002, 4-Ply	9.0	8.0	-, =R	4A	38.5	29.7-46.6	4	3.44	3.01-3.90	3
14	1002, 2-Ply	9.0	8.0	-, =R	2D	32.5	30.1-34.9	2	3.48	3.42-3.53	2
17b[4]	1002, 2-Ply	9.0	8.0	-, +R	1A	25.8	-	1	3.19	-	1
19	F150-11, 4-Ply	9.0	8.0	=W, =R	3A	29.3	27.3-30.4	3	3.73	3.37-4.27	3
19[4]	F150-11, 2-Ply	9.0	8.0	=W, =R	1A	28.2	-	1	-	-	-
19[4]	F150-11, 2-Ply	12.0	11.0	=W, =R	2A	19.3	18.7-20.0	2	3.70	3.50-3.90	2
19[4]	F150-11, 2-Ply	11.0	8.0	=W, =R	2A	23.9	23.4-24.3	2	3.65	3.11-4.19	2
24	181 OURI, 2-Ply	6.0	5.5	+W, +R	-	27.5	-	1	2.57	-	1
24	181 OURI, 2-Ply	6.0	5.5	=W, =R	-	29.7	29.6-29.9	2	2.57	-	1
24	181 OURI, 2-Ply	5.0	5.5	+W, +R	-	26.8	26.6-27.1	2	2.54	-	1
25	181 OURI, 3-Ply	6.0	5.5	+W, +R	-	27.3	27.0-27.6	2	2.91	2.90-2.93	2
25	181 OURI, 3-Ply	6.0	5.5	=W, =R	-	28.4	27.9-28.9	2	2.91	2.90-2.93	2
25	181 OURI, 3-Ply	5.0	5.5	+W, +R	-	26.0	23.8-28.3	2	2.74	2.64-2.83	2
25	181 OURI, 2-Ply	6.0	5.5	+W, +R	-	26.0	-	1	2.40	-	1
25	181 OURI, 2-Ply	6.0	5.5	=W, =R	-	29.8	29.3-30.2	2	2.40	-	1
25	181 OURI, 2-Ply	5.0	5.5	+W, +R	-	26.0	23.9-28.2	2	-	-	-
26	181 OURI, 3-Ply	5.0	5.5	+W, +R	-	28.9	27.5-31.4	4	2.71	2.66-2.77	2
26	181 OURI, 2-Ply	6.0	5.5	=W, =R	-	33.6	31.8-35.1	4	2.10	2.05-2.17	3
26	181 OURI, 2-Ply	5.0	5.5	+W, +R	-	29.3	29.1-29.5	2	2.08	1.86-2.31	2
27	181 OURI, 3-Ply	6.0	5.5	=W, =R	-	30.6	-	1	2.94	-	1
27	181 OURI, 3-Ply	6.0	5.5	=W, =R	-	30.1	29.3-30.8	2	2.76	-	1

[1] See Note 1 of Table 4.

[2] See Note 2 of Table 4.

[3] See Note 3 of Table 4.

[4] Several specimens in these groups were more rigidly supported on the sides than indicated...see "Experimental Procedure."

TABLE 8
GENERAL PANEL BUCKLING TEST DATA,
HINGED LOADED ENDS AND HINGED EDGES

Specimen Group Identification	Facing Material, Thickness	Facing Thickness (in.)	Loading Configuration: Loaded Edge (in.)	Loaded Length (in.)	Load Orientation[1]	Type of Collapse[2]	Ultimate Stress (psi x 10^{-3}): Average Stress	Range of Stress	Number of Specimens
27[3]	181 OURI[4], 2-Ply	-	5.0	5.5	-W, -R	-	24.7	-	1
27[3]	181 OURI, 2-Ply	-	5.0	5.5	-W, -R	-	28.1	27.8-28.4	2
27[3]	181 OURI, 2-Ply	-	6.0	3.5	-W, -R	-	31.4	-	1
27[3]	181 OURI, 3-Ply	-	6.0	5.5	-W, -R	-	27.6	-	1
28[3]	181 OURI, 2-Ply	-	6.0	3.5	-W, -R	-	14.4	12.7-15.5	3
12	F150-11, 3-Ply	0.2508	11.5	9.3	-W, -R	-	9.4	9.4-9.4	2
12	F150-11, 3-Ply	0.2510	9.0	7.5	-W, -R	-	18.3	18.2-18.5	3
17b[5]	1002, 2-Ply	-	9.0	8.5	-, +R	3B	26.6	25.7-27.1	3
19[5]	F150-11, 3-Ply	-	9.0	8.5	+W, +R	2A	27.0	25.8-28.2	2
19[5]	F150-11, 3-Ply	-	9.0	8.5	-W, -R	1A	28.3	-	1
19[5,6]	F150-11, 3-Ply	-	9.0	8.5	-W, -R	1A	29.6	-	1
19[7]	F150-11, 2-Ply	-	9.0	8.5	-W, -R	2A	41.1	33.2-49.1	2
22[7]	F150-11, 2-Ply	0.2366	9.1	7.5	+W, -R	-	14.8	-	1
22[7]	F150-11, 2-Ply	0.2366	11.5	9.2	+W, -R	-	11.2	-	1
22[7]	F150-11, 3-Ply	0.2512	9.0	7.5	+W, -R	-	15.3	-	1
22[7]	F150-11, 3-Ply	0.2512	11.5	9.2	+W, -R	-	11.3	-	1
22[7]	F150-11, 4-Ply	0.2655	9.1	7.5	+W, -R	-	13.5	-	1
22[7]	F150-11, 4-Ply	0.2655	11.3	9.2	+W, -R	-	11.1	-	1
23[7]	F150-11, 2-Ply	0.2368	9.1	7.5	+W, -R	-	16.2	-	1
23[7]	F150-11, 2-Ply	0.2368	11.5	9.2	+W, -R	-	11.9	-	1
23[7]	F150-11, 4-Ply	0.2537	8.5	7.5	+W, -R	-	20.8	-	1
23[7]	F150-11, 4-Ply	0.2537	11.5	9.2	+W, -R	-	14.1	-	1
23[7]	F150-11, 3-Ply	0.2368	11.7	9.2	+W, -R	-	9.4	-	1
23[7]	F150-11, 3-Ply	0.2368	9.0	7.5	+W, -R	-	3.1	-	1
20	F150-11, 3-Ply	0.2540	11.5	9.2	-W, -R	-	9.3	9.3-10.2	2
20	F150-11, 3-Ply	0.2539	9.0	7.4	+W, +R	-	11.7	10.8-13.2	3
20	181 OURI, 2-Ply	-	8.5	8.5	-W, -R	-	9.9	9.6-10.2	2

[1] See Note 1 of Table 4.

[2] See Note 2 of Table 4.

[3] Boundary Conditions Were Hinged Loaded Ends and Free Edges.

[4] See Note 3 of Table 4.

[5] Several specimens in this group were more rigidly supported on the sides than indicated...see "Experimental Procedure."

[6] Specimens Developed a Two-Half-Wave Buckle.

[7] The failure loads for these specimen groups were accurately established by the measurement of side deflection.

TABLE 9
INTRACELLULAR BUCKLING TEST DATA

Specimen Group Identification	Facing Material Thickness	Facing Thickness (in.)	R_1 (Nominal) (in.)	R_2^1 (Measured) (in.)	(f/R_2)	Loading Configuration: Loaded Edge (in.)	Loaded Length (in.)	Load Orientation	Stress At Collapse[3] (psi, x 10^{-3})	Sandwich Modulus[3] (psi x 10^{-6})	Number of Specimens	Dimpling Stress (psi x 10^{-3})	Number of Specimens
29	143 OURI, 4-Ply	0.0332	0.375	0.371	0.0895	6.0	5.0	-W, -R	16.5 -	4.90 -	1	-	-
29	143 OURI, 4-Ply	0.0332	0.375	0.367	0.0905	6.0	5.0	+W, +R	- 11.4 -	- 2.21 -	1	-	-
30	143 OURI, 3-Ply	0.0249	0.375	0.358	0.0696	6.0	5.0	-W, -R	18.9 17.0 15.3	5.86 5.54 5.27	4	15.3	1
30	143 OURI, 3-Ply	0.0249	0.375	0.357	0.0697	6.0	5.0	-W, +R	10.5 9.6 8.5	2.46 2.15 1.85	3	-	-
31	181 OURI, 2-Ply	0.0175	0.375	0.376	0.0465	6.0	5.0	-W, -R	15.2 14.6 13.4	4.23 3.74 3.16	4	10.2	2
31	181 OURI, 2-Ply	0.0175	0.375	0.385	0.0455	6.0	5.0	+W, +R	12.0 10.7 9.6	4.20 3.24 2.16	4	9.6	1
32	143 OURI, 4-Ply	0.0324	0.250	0.229	0.1415	6.0	5.0	-W, -R	32.2 27.1 25.9	6.55 6.03 5.68	4	-	-
32	143 OURI, 4-Ply	0.0324	0.250	0.207	0.1565	6.0	5.0	+W, +R	- 16.5 -	- 2.06 -	1	-	-
33	143 OURI, 3-Ply	0.0251	0.250	0.243	0.1033	6.0	5.0	-W, -R	27.7 25.7 23.4	5.98 5.66 5.13	3	-	-
33	143 OURI, 3-Ply	0.0251	0.250	0.241	0.1041	6.0	5.0	+W, +R	14.7 13.6 12.1	2.10 1.94 1.70	3	-	-
34[2]	181 OURI, 2-Ply	0.0220	0.250	0.241	0.0913	6.0	5.0	-W, -R	8.4 8.2 8.0	3.19 2.99 2.84	3	-	-
35	181 OURI, 2-Ply	0.0233	0.250	0.265	0.0879	6.0	5.0	-W, -R	15.9 15.5 15.1	2.76 2.70 2.66	3	-	-
35	181 OURI, 2-Ply	0.0233	0.250	0.262	0.0889	6.0	5.0	+W, +R	12.1 10.7 8.3	2.85 2.65 2.31	4	-	-
36[2]	181 OURI, 2-Ply	0.0205	0.1875	0.1695	0.1209	6.0	5.0	-W, -R	22.0 21.2 20.5	3.29 2.99 2.68	2	-	-
37	143 OURI, 4-Ply	0.0253	0.125	0.156	0.1622	6.0	5.0	-W, -R	- 25.0 -	- 3.93 -	1	-	-
37	143 OURI, 4-Ply	0.0253	0.125	0.113	0.2239	6.0	5.0	+W, -R	- 17.9 -	- 2.06 -	1	-	-

[1] Obtained By Averaging Four Actual Cell Size Measurements Per Specimen.

[2] Specimens Failed As a Result of Poor Core-To-Facing Bond.

[3] Values Are Listed Vertically in the Order of High, Average, and Low.

TABLE 10
SHEAR PROPERTIES,
HEXAGONAL CELL HONEYCOMB CORE

Specimen Group Identification	Load Orientation[2]	Ultimate Strength[1] Avg. Shear Strength (psi)	Range of Shear Strength (psi)	Number of Specimens	Modulus[1] Avg. Core Modulus (psi)	Range of Core Modulus (psi)	Number of Specimens	Published Data Density[3] (lb/ft³)	Shear[3] Strength (psi)
1	+R	190	-	1	29,200	28,500-30,200	5	4.3	157
	-R	-	-	0	74,300	67,200-83,900	5	4.3	300
2	+R	195	183-202	4	29,400	27,300-31,500	5	4.3	175
	-R	354	347-363	3	66,500	63,000-72,400	5	4.3	300
3	+R	221	218-226	5	29,000	27,500-30,200	5	4.3	175
	-R	439	431-445	4	78,000	70,800-88,400	5	4.3	300
4	+R	36	35-37	4	4,800	4,200-5,400	5	1.6	40
	-R	55	53-59	4	14,000	12,500-15,400	3	1.6	75
5	+R	43	41-46	8	7,800	4,900-10,400	9	1.6	40
	-R	64	63-67	9	17,700	14,000-26,200	9	1.6	75
6	-R	110	100-125	6	15,600	13,400-16,900	6	1.6	75
7	+R	62	59-66	4	6,600	6,100-7,200	4	1.6	40
	-R	105	100-108	4	17,600	16,500-18,200	4	1.6	75
8	+R	56	50-62	3	7,200	6,700-7,600	3	1.6	40
	-R	99	97-101	3	17,900	14,700-21,200	3	1.6	75
9	+R	82	81-83	2	11,300	10,100-12,600	2	1.6	40
	-R	139	124-152	5	26,500	24,700-28,500	5	1.6	75
10	+R	76	73-78	4	8,700	8,500-9,000	3	1.6	40
	-R	125	124-126	3	22,300	18,500-24,300	3	1.6	75
11	+R	308	-	1	25,200	-	1	1.6	40
	-R	265	213-324	3	22,700	19,500-25,000	3	1.6	75
12	+R	90	86-94	7	11,100	10,300-12,000	7	1.6	40
	-R	146	141-150	7	21,600	18,400-24,900	7	1.6	75
13	+R	90	85-95	4	14,600	13,400-15,600	4	2.3	50
	-R	145	142-149	6	30,200	28,400-30,800	6	2.3	125
14	+R	200	175-217	3	42,600	39,900-46,300	3	1.6	50
	-R	183	-	1	40,000	-	1	1.6	125
16	+R	127	122-129	3	19,900	18,800-21,500	3	3.1	100
	-R	216	207-224	2	34,800	31,600-38,000	2	3.1	180
17b	+R	139	135-145	5	20,300	18,700-22,800	5	3.1	100
	-R	247	235-255	9	40,500	37,100-47,600	10	3.2	180
18	+R	153	148-158	2	14,600	13,600-15,600	2	3.1	100
	-R	249	247-251	3	39,000	28,400-55,100	3	3.1	180
19	+R	196	187-205	2	25,200	24,000-26,300	2	3.1	100
	-R	397	321-539	8	56,800	43,800-65,400	9	3.1	180
20	+R	199	186-225	10	27,000	23,900-36,000	10	3.1	100
	-R	298	282-306	12	47,400	35,600-70,200	12	3.1	180
21	+R	84	84-85	3	2,500	2,400-2,700	3	2.2[4]	-
	-R	179	174-184	4	8,400	8,000-8,800	4	2.2	-
22	+R	157	149-171	6	4,100	3,300-5,400	6	2.2	-
	-R	266	253-280	8	8,700	7,200-9,600	8	2.2	-
23	+R	319	304-333	9	9,300	7,800-10,400	9	4.0	-
	-R	554	526-575	13	18,000	14,800-21,000	18	4.0	-
24	+R	177	134-223	34	9,400	5,800-12,900	40	-	146[5,6]
	-R	251	188-342	34	24,600	13,700-42,300	47	-	282

[1] Tested by the Plate Shear Method (MIL-STD 401A).

[2] See Note 1 of Table 4.

[3] Except where noted, these data were obtained from Hexcel Products Inc., Brochure "D" and "E" dated 1959.

[4] Hexcel Products Inc. TSB-112, Dated March, 1960.

[5] Douglas Aircraft Company Brochure Titled "Aircomb", Dated September, 1963.

[6] Shear Modulus Equal 7200 psi Obtained From Douglas Aircraft Company Brochure Titled "Aircomb, Test and Technical Data," Dated October, 1963. Also, for group 25, G_{Cbs} (+R) = 4,300 psi and G_{Cbs} (-R) = 6,300 psi and for group 27, G_{Cbs} (-R) = 6,900 psi.

TABLE 11
FLATWISE TENSION TEST DATA,
HEXAGONAL CELL HONEYCOMB CORE

Specimen Group Identification	Core Failure			Core-to-Facing Failure		
	Strength (psi)	Range of Strength (psi)	Number of Specimens	Strength (psi)	Range of Strength (psi)	Number of Specimens
1	-	-	-	870	820-930	5
2	890	850-910	6	950	-	1
3	-	-	-	877	740-1,080	7
5	245	220-290	13	220	-	1
6	301	220-340	13	-	-	-
7	320	270-350	7	-	-	-
9	345	300-380	14	-	-	-
10	340	310-380	6	-	-	-
11	360	310-410	7	-	-	-
12	393	355-425	8	-	-	-
13	360	310-430	6	332	260-480	8
14	495	400-560	13	490	-	1
16	540	480-590	11	480	370-540	3
17b	-	-	-	610	570-670	6
18	-	-	-	410	300-460	5
19	722	580-990	21	690	650-720	2
20	763	710-815	12	-	-	-
21	-	-	-	389	360-415	4
22	-	-	-	531	370-900	20
23	-	-	-	911	840-1,040	8

TABLE 12
FLATWISE COMPRESSION PROPERTIES,
HEXAGONAL CELL HONEYCOMB CORE

Specimen Group Identification	Ultimate Strength (psi)	Range of Ulti. Stress (psi)	Number of Specimens	Average Modulus (psi x 10^{-3})	Range of Modulus (psi x 10^{-3})	Number of Specimens	Published Strength[1] (psi)
9	198	183-213	2	37.5	37.0-38.0	2	155
14	320	253-353	4	67.8	50.4-81.0	3	210
16	328	300-355	3	79.3	65.9-92.7	2	310

[1] Hexcel Report No. 191-1, Dated February 29,1956 (Specimen Height 0.625 Inch).

TABLE 13
ELASTIC MODULUS OF ALUMINUM CORE

Loaded Direction[1]	Nominal Cell Size (in.)	Core Thickness (in.)	Thickness of Cell Wall (in.)	Number of Specimens	Modulus (psi)
+R	3/8	0.40	.0013	1	8
=R	3/8		.0013	1	8
=R	3/16	0.75	.0013	1	16
+R	3/16		.0013	1	23
+R	3/16		.0025	1	48
=R	3/16		.0025	1	169

[1] See Note 1 of Table 4.

TABLE 14

PROPERTIES OF FACING LAMINATES

Specimen Group Identification	Fabric Style	Loading Direction[1]	Fabric Plies	Facing Thickness (in.)	Fabrication Conditions[2]			Compressive Properties[3]				Tensile Properties[4]			
					Time (Min.)	Temperature (°F)	Pressure (psi)	Ultimate Strength (psi x 10^{-3})	Number of Specimens	Laminate Modulus (psi x 10^{-6})	Number of Specimens	Ultimate Strength (psi x 10^{-3})	Number of Specimens	Laminate Modulus (psi x 10^{-6})	Number of Specimens
12	181 (F150-11)[4]	+W, -W	3	0.0260	60	350	59	40.8 36.2 33.7	3	3.68 3.26 3.12	3	53.2 47.4 42.2	6	4.58 3.21 2.73	6
20	181 (F150-11)	+W, -W	3	0.0256	60	350	99	36.8 34.1 28.8	8	3.67 3.41 2.99	4	59.6 54.5 51.7	5	3.99 3.27 2.99	5
20	181 (F150-11)	+W, -W	4	0.0329	60	350	72	41.3 35.3 31.1	6	4.14 3.72 3.25	6	62.9 56.2 44.5	6	3.27 3.18 2.94	6
21	181 (F150-11)	+W, -W	3	0.0171	60	350	70	27.8 26.2 25.0	6	3.85 3.64 3.44	3	55.1 50.8 47.6	6	3.53 2.90 2.59	6
21	181 (F150-11)	+W, -W	3	0.0255	60	350	70	35.0 31.5 28.3	6	4.22 3.84 3.20	6	50.0 46.6 42.9	6	3.23 3.03 2.86	6
22	181 (F150-11)	+W, -W	3	0.0247	60	350	70	34.5 31.8 29.2	6	4.48 3.95 3.44	4	53.7 50.2 48.8	6	3.23 3.07 2.95	6
22	181 (F150-11)	+W, -W	4	0.0319	60	350	70	46.0 40.2 36.6	6	4.03 3.72 3.37	6	61.2 57.0 54.7	6	3.32 3.23 3.07	6
23	181 (F150-11)	+W, -W	4	0.0324	60	350	70	41.8 40.0 33.9	6	4.63 3.77 3.07	5	58.4 54.4 50.6	6	3.27 3.12 2.91	6
23	181 (F150-11)	+W, -W	3	0.0262	60	350	59	40.9 38.7 36.8	8	3.70 3.58 3.47	4	56.5 50.6 44.0	6	3.40 3.09 2.86	6
23	181 (F150-11)	+W, -W	2	0.0168	60	350	70	23.1 20.6 19.3	6	3.30 3.13 2.96	5	43.5 40.5 37.2	6	3.06 2.93 2.46	6
22	181 (F150-11)	+W, -W	2	0.0177	60	350	70	24.8 23.9 22.4	6	3.40 3.16 2.84	6	65.2 50.6 43.6	6	3.62 3.17 2.69	6
29	143 (OURI)	+W, -W	4	0.0339	90	185	59	23.6 18.4 14.9	8	1.91 1.54 1.10	4	100.0 87.8 79.1	6	5.31 4.92 4.69	6
30	143 (OURI)	+W, -W	3	0.0253	90	185	59	22.5 22.0 21.1	8	1.97 1.68 1.25	4	105.7 91.6 82.1	6	5.43 5.16 4.81	6
32	143 (OURI)	+W, -W	4	0.0329	90	185	59	21.3 20.0 13.9	8	1.84 1.74 1.67	4	98.3 91.2 88.5	6	5.38 5.08 4.58	6
33	143 (OURI)	+W, -W	3	0.0256	90	185	59	23.0 22.3 21.7	8	1.88 1.76 1.62	4	95.4 86.2 80.3	6	5.54 4.97 4.15	6
34	181 (OURI)	+W, -W	2	0.0206	90	185	61	32.0 31.5 30.0	6	3.43 3.04 2.82	5	45.0 43.8 42.0	6	3.37 2.56 2.01	6
35	181 (OURI)	+W, -W	2	0.0184	90	185	61	35.0 30.8 27.0	6	3.81 3.58 3.27	6	55.0 48.8 40.0	6	2.88 2.59 2.30	6
35	181 (OURI)	+W, -W	2	0.0184	90	160	13	34.0 29.4 26.0	6	2.97 2.67 2.39	6	37.0 35.0 31.0	4	2.19 2.00 1.85	4
36	181 (OURI)	+W, -W	2	0.0212	90	185	61	33.2 30.1 27.0	6	3.42 2.98 2.28	6	42.3 40.4 37.0	6	2.53 2.33 2.14	6
-	181 (OURI)[5]	+W, -W	3	0.0240	90	180	70	44.4 41.0 38.8	24	4.17 3.75 3.45	6	50.5 50.9 46.8	6	3.00 2.94 2.85	3
37	143 (OURI)	+W, -W	4	0.0261	90	180	59	23.3 22.3 20.4	8	1.84 1.74 1.50	4	94.9 86.1 78.4	6	5.12 4.83 4.58	6
-	143 (OURI)	+W, -W	4	0.0335	90	180	59	24.0 22.6 20.9	8	1.95 1.80 1.70	4	102.1 92.5 85.3	6	5.38 5.10 4.52	6
-	143 (OURI)	+W, -W	3	0.0258	90	180	59	25.4 22.5 19.3	8	1.88 1.58 1.40	4	95.9 89.4 79.5	6	5.64 4.96 4.74	6

[1] Order: Compression, Tension. See Note 1 of Table 4 For Definition of Symbols.

[2] All Facings Post-Cured 2 Hours @ 300°F.

[3] Values Are Listed Vertically in the Order of High, Average, and Low.

[4] See Note 3 of Table 4.

[5] Data From Reference 1.

APPENDIX II

SAMPLE CALCULATIONS

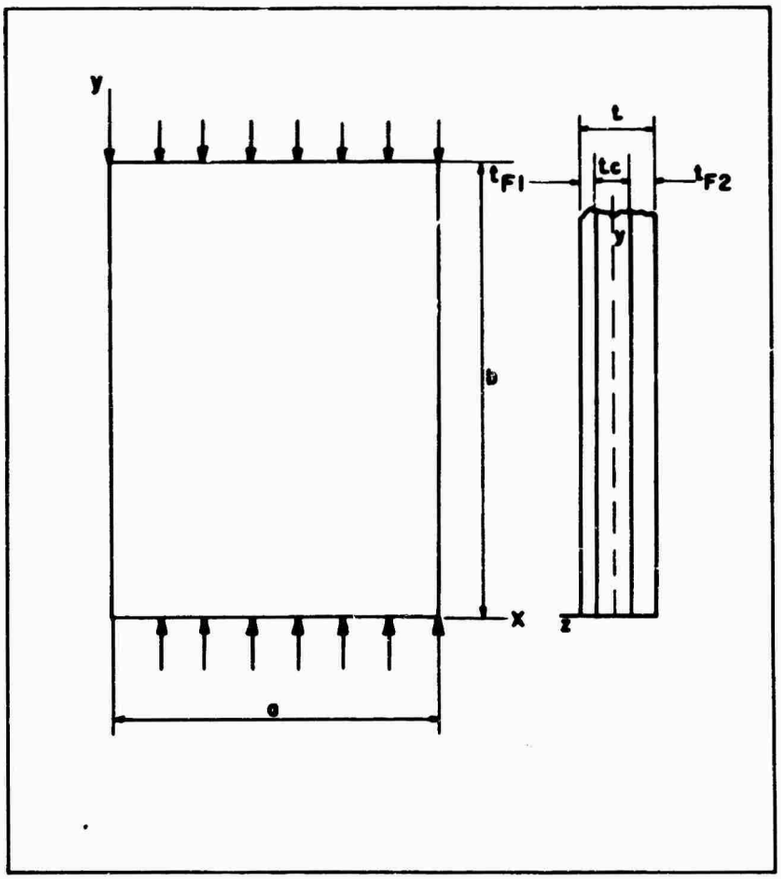

Figure 30. Coordinate System Used in Sandwich Analysis.

1. General Buckling

   a. Analysis According to Reference 8.

$$f_{Fcr} = \frac{\pi^2}{4} \frac{t_{F1}\, t_{F2}}{a^2} \left[\frac{t + t_c}{t - t_c}\right]^2 \frac{(E_{Fa}\, E_{Fb})}{\lambda_F}^{1/2} K \qquad (7)$$

material properties for specimen group 22

E_{Fb} (+W) = 3.95 $(10)^6$ psi

$$E_{Fa} \ (=W) = 3.95 \ (10)^6 \ (1.055)^* = 4.17 \ (10)^6 \text{ psi}$$

$$G_{Caz} = 0.41 \ (10)^4 \text{ psi}$$

$$G_{Cbz} = 0.87 \ (10)^4 \text{ psi}$$

$$t = 0.2512 \text{ inch (measured)}$$

$$t_c = 0.20 \text{ inch}$$

$$t_{F1} = t_{F2} = 0.0247 \text{ inch}$$

$$a = 9.04 \text{ inch}$$

$$b = 7.48 \text{ inch}$$

$$E_{45} = 2.20 \ (10)^6 \text{ psi } (E_{45} \text{ values obtained from reference 13})$$

$$\mu_{Fab} = \mu_{Fba} = 0.13$$

calculations

$$b/a = \frac{7.48}{9.04} = 0.8274$$

$$\alpha = \left[\frac{E_{Fa}}{E_{Fb}}\right]^{1/2} = \left[\frac{41.7 \ (10)^6}{3.95 \ (10)^6}\right]^{1/2} = 1.0272 \qquad (14)$$

$$\frac{1}{G_{Fab}} = \frac{4}{E_{45}} - \frac{1 - \mu_{Fab}}{E_{Fa}} - \frac{1 - \mu_{Fba}}{E_{Fb}} \qquad (17)$$

$$\frac{1}{G_{Fab}} = \frac{4}{2.20 \ (10)^6} - \frac{1 - 0.13}{4.17 \ (10)^6} - \frac{1 - 0.13}{3.95 \ (10)^6}$$

$$G_{Fab} = 0.7198 \ (10)^6 \text{ psi}$$

---

*The factor 1.055 was calculated from the data given on page 7 of reference 8--$E_F \ (=W)/E_F \ (+W) = 1.055$.

$$\lambda_F = 1 - \mu_{Fab}\,\mu_{Fba} = 1 - (0.13)^2 = 0.9831$$

$$\gamma = \frac{G_{Fab}\,\lambda_F}{(E_{Fa}\,E_{Fb})^{1/2}} = \frac{0.7198\ (10)^6\ (0.9831)}{\left[4.17\ (10)^6\ (3.95)\ (10)^6\right]^{1/2}} \qquad (16)$$

$$\gamma = 0.1743$$

$$\beta = \alpha\mu_{Fab} + 2\gamma = 1.0272\ (0.13) + 2\ (0.1743) \qquad (15)$$

$$\beta = 0.4821$$

$$A = 1 - \beta^2 + \gamma\left[\frac{\alpha b^2}{a^2} + 2\beta + \frac{a^2}{\alpha b^2}\right] \qquad (11)$$

$$A = 1 - (0.4821)^2 + 0.1743\left[\frac{1.0272\ (7.48)^2}{(9.04)^2} + 2(0.4821) + \frac{(9.04)^2}{(1.0272)\ (7.48)^2}\right]$$

$$A = 1 - (0.4821)^2 + 0.1743\ (3.0893)$$

$$A = 1.3061$$

$$V = \frac{t_c\ t_{F1}\ t_{F2}}{t - t_c}\,\frac{\pi^2}{a^2}\,\frac{(E_{Fa}\,E_{Fb})}{\lambda_F\,G_{Cbz}}^{1/2} \qquad (12)$$

$$V = \frac{(0.20)\ (0.0247)^2}{0.2512 - 0.20}\,\frac{3.1416}{(9.04)^2}\,\frac{\left[4.17\ (10)^6\ (3.95)\ (10)^6\right]^{1/2}}{0.9831\ (0.87)\ (10)^4}$$

$$V = 0.14155$$

$$r = \frac{G_{Cbz}}{G_{Caz}} = \frac{0.87\ (10)^4}{0.41\ (10)^4} = 2.1220 \qquad (13)$$

$$K_M = \frac{\frac{\alpha b^2}{a^2} + 2\beta + \frac{a^2}{\alpha b^2} + VA\left[\frac{ra^2}{b^2} + 1\right]}{1 + V\,\frac{ra^2}{b^2}\left[\frac{\alpha b^2}{a^2} + \gamma\right] + V\left[\frac{a^2}{\alpha b^2} + \gamma\right] + V^2 rA\,\frac{a^2}{b^2}} \qquad (10)$$

$$K_M = \frac{3.0893 + 0.14155\ (1.3061)\left[\frac{2.1220(9.04)^2}{(7.48)^2} + 1\right]}{1 + 0.14155\ \frac{(2.1220)\ (9.04)^2}{(7.48)^2}\left[\frac{1.0272\ (7.48)^2}{(9.04)^2} + 0.1743\right] + \frac{(1)}{0.14155\left[\frac{(9.04)^2}{1.0272\ (7.48)^2} + 0.1743\right] + (0.14155)^2\ (2.1220)} \frac{(1)}{(1.3061)\ \frac{(9.04)^2}{(7.48)^2}}}$$

$$K_M = 2.2732$$

$$K_F = \frac{1}{3}\left[\frac{t_{F1}}{t_{F2}} + \frac{t_{F2}}{t_{F1}} - 1\right]\left[\frac{t - t_c}{t + t_c}\right]^2\left[\frac{\alpha_b^2}{a^2} + 2\beta + \frac{a^2}{\alpha_b^2}\right] \qquad (9)$$

$$K_F = \frac{1}{3}\left[\frac{0.2512 - 0.20}{0.2512 + 0.20}\right]^2 \quad (3.0893)$$

$$K_F = 0.0124$$

$$K = K_M + K_F$$

$$K = 2.2732 + 0.0124 \qquad (8)$$

$$K = 2.2856$$

$$f_{Fcr} = \frac{\pi^2}{4}\frac{(0.0247)^2}{(9.04)^2}\left[\frac{0.2512 - 0.20}{0.2512 + 0.20}\right]^2 \frac{\left[4.17\ (10)^6\ (3.95)\ (10)^6\right]^{1/2}}{0.9831} \quad (2.2856) \qquad (7)$$

$$f_{Fcr} = 14{,}400 \text{ psi}$$

b. Analysis According to Reference 7 (same data used).

$$P_{crs} = \frac{P_{cr}}{1+\eta} \tag{1}$$

$$C = E_{Fa}\mu_{Fab} + 2\lambda_F G_{Fab} \tag{4}$$

$$C = 4.17\ (10)^6\ (0.13) + 2\ (0.9831)\ (0.7198)\ (10)^6$$

$$C = 1.9574\ (10)^6$$

$$T = \frac{\pi^2}{2\ \lambda_{Fa}{}^2}\ (E_{Fa}\ b^2/a^2 + E_{Fb}\ a^2/b^2 + 2C) \tag{3}$$

$$T = \frac{(3.14)^2}{2\ (0.9831)\ (9.04)^2}\left[\frac{4.17\ (10)^6\ (7.48)^2}{(9.04)^2} + \frac{3.95\ (10)^6\ (9.04)^2}{(7.48)^2} + 2\ (1.9574)\ (10)^6\right]$$

$$T = 0.7701\ (10)^6$$

$$K' = G_{Cbz} + G_{Caz}\ (b^2/a^2) \tag{6}$$

$$K' = 0.87\ (10)^4 + \frac{0.41\ (10)^4\ (7.48)^2}{(9.04)^2}$$

$$K' = 1.1507\ (10)^4$$

$$\eta = \frac{t_c\ t_F\ T}{K'} = \frac{0.20\ (0.0247)\ (0.7701)\ (10)^6}{1.1507\ (10)^4} \tag{5}$$

$$\eta = 0.3306$$

$$P_{cr} = (t^3 - t_c^3)\ \frac{T}{6} = \left[(0.2512)^3 - (0.20)^2\right]\ \frac{0.7701\ (10)^6}{6} \tag{2}$$

$$P_{cr} = 964.24\ \text{lb./in.}$$

$$P_{crs} = \frac{964.24}{1 + 0.3306} = 724.67\ \text{lb./in.}$$

$$f_{Fcr} = \frac{P_{crs}}{2\ t_F} = \frac{724.67}{2\ (0.0247)}$$

$$f_{Fcr} = 14{,}700 \text{ psi}$$

2. Face Wrinkling (Analysis According to Reference 3)

a. Calculation of Parameter K

$$K = \frac{\sqrt{E_{Cb}\ E_{Cz}}}{2\ G_{Cbz}} - \mu_{Cbz}\sqrt{\frac{E_{Cb}}{E_{Cz}}} \qquad (20)$$

material properties for specimen group 10

$E_{Cb} = 8$ psi

$E_{Cz} = 2.67\ (10)^7\ (0.0013/0.375) = 9.26\ (10)^4$ psi*

$E_{Fb} = 3.70\ (10)^6$ psi

$G_{Cbz}\ (+R) = 8.70\ (10)^3$ psi

$\mu_{Cbz} = 0.30$**

$t_c = 0.40$ inch (nominal)

$t_F = 0.04$ inch (nominal)

$\mu_{Fab} = 0.13$

$L = 3/8$ inch (assumed)

---

* E_{Cz} value obtained by the equation $E_{Cz} = 2.67\ (10)^7\ (t/s)$ given on page 6 of reference 4 where t is the foil thickness and s is the cell diameter.

**Assumed to be that for aluminum, the basic material (see top of page 4 of reference 3.

calculation

$$\kappa = \sqrt{\frac{8\ (9.26)\ (10)^4}{2\ (8.70)\ (10)^3}} - 0.30\ \sqrt{\frac{8}{9.26\ (10)^4}}$$

$$\kappa = 0.05$$

b. Parameter b Calculation

$$f_{Fcr} = \frac{B}{L^2}\ \frac{t_c + aL^4}{t_c + bL} \qquad (19)$$

material properties for specimen group 5

$E_{Fb} = 3.70\ (10)^6$ psi $\qquad \mu_{Fab} = 0.13$

$E_{Cz} = 9.26\ (10)^4$ psi $\qquad L = 3/8$ inch (assumed)

$t_c = 0.75$ inch (nominal) $\qquad f_{Fcr}$ (+R) = 30,700 psi

$t_F = 0.04$ inch (nominal)

calculations

$$\lambda_F = 1 - \mu_{Fab}^2 = 1 - (0.13)^2 = 0.9831$$

$$B = \frac{\pi^2}{12}\ \frac{t_F^2\ E_{Fb}}{\lambda_F} = \frac{\pi^2}{12}\ \frac{(0.04)^2\ (3.70)\ (10)^6}{0.9831} = 4952.73$$

$$a = \frac{24}{\pi^4}\ \frac{E_{Cz}\ \lambda_F}{E_{Fb}\ t_F^3} = \frac{24}{\pi^4}\ \frac{9.26\ (10)^4\ (0.9831)}{3.70\ (10)^6\ (0.04)^3} = 94.54$$

$$30{,}700 = \frac{4952.73}{(0.375)^2}\ \frac{1.0 + 94.54\ (0.375)^4}{1.0 + 0.375\ b} \qquad (19)$$

$$b = 6.1118$$

c. Stress Calculation (Specimen Group 10)

$$f_{Fcr}\ (+R) = \frac{4952.73}{(0.375)^2}\ \frac{0.40 + 94.54\ (0.375)^4}{0.40 + 6.1118\ (0.375)} \qquad (19)$$

f_{Fcr} (+R) = 29,700 psi (Test: 24,000 psi)

3. Shear Crimping (Analysis According to Reference 3)

$$f_{Fcr} = \frac{t_c}{2t_F} G_{Cbz} \qquad (21)$$

material properties for specimen group 25

t_c = 0.75 inch

t_F = 0.03 inch (nominal)

G_{Cbz} = 4,300 psi

calculation

$$f_{Fcr} = \frac{0.75\ (4,300)}{2\ (0.03)}$$

f_{Fcr} = 53,800 psi

4. Intracellular Buckling (Analysis According to Reference 9)

$$f_{Fcr} = (E_{Fb}/3)\ (t_F/R)^{3/2} \qquad (22)$$

material properties for specimen group 31 (Table 9: 181-style, 2-ply)

E_{Fb} (+W) = 3.06 $(10)^6$ psi* $\qquad$ $R = R_2$ = 0.376 inch

E_{Fb} (=W) = 3.23 $(10)^6$ psi**

t_F = 0.0175 inch

calculation

$$f_{Fcr}\ (=W) = \frac{3.23\ (10)^6}{3} \left[\frac{0.0175}{0.376}\right]^{3/2}$$

f_{Fcr} (=W) = 10,820 psi

---

* Values assumed equal to the averages obtained from specimen groups 34, 35, and 36.

** See footnote on page 68.

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Contributions to the acceptance of fiberglass-reinforced plastic (FRP) as an aircraft structural material were made through verification of existing theoretical strength relationships by the fabrication and testing of sandwich panels in the laboratory. The four basic failure modes were investigated for sandwich plates and plate columns loaded in edgewise compression. These were general buckling, face wrinkling, shear crimping and face dimpling. To achieve these modes, it was necessary to vary not only the specimen size and boundary conditions but also, in many cases, the dimensions and composition of the constituent materials.

In the development of a suitable structural sandwich, a number of advances were made in the realm of fabrication. These include the development of a multi-ply pre-preg, the establishment of a precure phase in the resin cure cycle as a control of resin flow, and the use of the separately-bonded type of sandwich construction. The effect of adhesive filleting on the core strength and the effect of laminate thickness on facing strength properties were also isolated.

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