S

~~FOR OFFICIAL USE ONLY~~

SPECIAL REPORT
COPY NO ______

DTIC
SELECTE
NOV 07 1995
S D
F

STRUCTURAL PLASTICS IN AIRCRAFT

DISTRIBUTION STATEMENT A
Approved for public release;
Distribution Unlimited

19951102 005

DTIC QUALITY INSPECTED 5

MARCH 1965

DEPARTMENT OF DEFENSE
PLASTICS TECHNICAL EVALUATION CENTER
ARRADCOM, DOVER, N. J. 07801

PLASTEC

PLASTICS TECHNICAL EVALUATION CENTER

PICATINNY ARSENAL
DOVER, NEW JERSEY

~~FOR OFFICIAL USE ONLY~~

PLASTEC

Authorized by the Office of Director of Defense Research and Engineering, the Plastics Technical Evaluation Center (PLASTEC) evaluates and disseminates technical information on current development, engineering, and application work in the field of plastics and reinforced plastics. It engages in materials surveys and other special assignments, and provides the Department of Defense with technical data and advice on research and development programs on plastics.

Army, Navy, and Air Force installations, and contractors and other suppliers of defense needs may request information from this center directly.

PLASTEC documents are distributed automatically to qualified organizations and individuals, according to the stated interest established in its Field of Interest Register (FOIR).

Unless otherwise stated below, PLASTEC reports are available to qualified requesters from the Defense Documentation Center (DDC), Cameron Station, Alexandria, Virginia, 22314; and to the general public through the Clearinghouse for Federal Scientific and Technical Information (CFSTI), Springfield, Virginia 22151.

Harry E Pebly Jr

Harry E. Pebly, Jr., Director

Copies not available from DDC or CFSTI.

LIMITATIONS:

- Initial distribution of this report is limited to those individuals or installations listed herein.
- Use of this report is limited to the recipient installation.
- Inquiries concerning the procurement of this report should be referred to PLASTEC.

```
*MSG DI4  DROLS PROCESSING - LAST INPUT IGNORED

*MSG DI4 DROLS PROCESSING-LAST INPUT IGNORED

*MSG DI4  DROLS PROCESSING - LAST INPUT IGNORED


--    1   OF     1
--                     ***DTIC DOES NOT HAVE THIS ITEM***
-- 1 -  AD NUMBER: D409138
-- 5 -  CORPORATE AUTHOR: PLASTICS TECHNICAL EVALUATION CENTER DOVER NJ
-- 6 -  UNCLASSIFIED TITLE:     STRUCTURAL PLASTICS IN AIRCRAFT (U).
--10 -  PERSONAL AUTHORS: CUTLER,M. ;NARDONE,J. ;SLOBODZINSKI,A. E. ;
--      SHIBLEY,A. M. ;
--11 -  REPORT DATE:      MAR   , 1965
--12 -  PAGINATION: XXXXXX    MEDIA COST:  $  6.00
--14 -  REPORT NUMBER:    PLASTEC-SR-1
--20 -  REPORT CLASSIFICATION:  UNCLASSIFIED
--22 -  LIMITATIONS (ALPHA):    CONTROLLED
--33 -  LIMITATION CODES:  1  24

-- END              Y  FOR NEXT ACCESSION                END

 Alt-Z FOR HELP3 ANSI
```

~~FOR OFFICIAL USE ONLY~~

SPECIAL REPORT

STRUCTURAL PLASTICS IN AIRCRAFT

by

ALLEN M. SHIBLEY
Plastics Technical Evaluation Center

and

ADOLPH E. SLOBODZINSKI
JOHN NARDONE
MARTIN CUTLER
Plastics and Packaging Laboratory

MARCH 1965

PLASTICS TECHNICAL EVALUATION CENTER
PICATINNY ARSENAL, DOVER, NEW JERSEY

~~FOR OFFICIAL USE ONLY~~

CONTENTS

LIST OF TABLES

* Numbers indicate Section (3-) and position within Section (-4)

LIST OF FIGURES

NOTE: Throughout the text of this report, the genetic spelling of 'fiberglas', as given in the American College Dictionary (Random House) is used.

SECTION 1. INTRODUCTION

The present survey attempts to answer a series of questions related to the fabrication of military aircraft using fiberglas composites as the major structural materials. Such a survey implies a critique of the aircraft and reinforced plastics industries. Most experiences in the aircraft industry have been based on metals technology. The reinforced plastics industry in contrast has developed distinctly different techniques. The aircraft manufacturer is accustomed to buying his raw materials to specified properties as "off-the-shelf" items. With reinforced plastics, he fabricates his own structural material and its properties depend on the care and technique expended in its manufacture.

The major consideration is the question of how plastic designs can be adapted to aircraft without sacrificing the potential of the material to the limitations of metal designs or to current assembly line practices. Two lines of approach are open - to substitute fiberglas composites for metals with minimum design change, or to incorporate newer concepts into structural designs which are more closely aligned to the characteristics of the material.

Scope of the Survey

The objectives of this report are:

- To consider the feasibility of constructing all structural parts of an aircraft from existing fiberglas composites.
- To consider alternate designs in which FRP is combined with metals or in which only certain structures would be FRP.
- To review possible advantages and disadvantages of such constructions in the light of specific aircraft mission requirements.
- To review present and earlier applications of FRP which are of significance.
- To give indications of expected problem areas from preliminary design to finished aircraft.
- To indicate programs required to further advances in the use of FRP as a structural material.
- To indicate programs in other areas of FRP which can be applied to aircraft technology.

Types of Aircraft

The sections on design, fabrication, materials and applications will emphasize the lighter aircraft, which can be defined as:

- V/STOL, V/TOL, STOL, COIN
- Rotary wing

- Maximum mach number - 0.8
- Maximum gross weight - 50,000 pounds

Present and earlier practices with FRP show a wider usage, extending to all types of aircraft. Many of the practices discussed for the lighter aircraft will be applicable to other types.

Information Sources

A literature search included a review of early NACA reports, the later NASA reports, and all relative PLASTEC documents. A special machine run was instigated in cooperation with the Defense Documentation Center. While these efforts yielded some pertinent information, the most useful material came from the various aerospace companies and much of this was of a proprietary nature. For this reason this report has been classified for official use only.

Companies Contributing

Because of time requirements, only a limited number of aerospace and other companies could be included in the survey. The attempt was made to get a representative cross-section of the industrial segments involved. Judgments as to which company to include were based on a number of factors - previous knowledge of the company, availability of personnel, travel distances, and current programs underway at the organizations. The fact that certain companies were included or excluded has therefore no particular significance as to their selection or relative merits. The types of organizations visited included:

- Manufacturers, fixed wing aircraft
- Manufacturers, rotary wing aircraft
- Manufacturers, reinforced plastics
- Manufacturers, filament wound structures
- Raw material suppliers
- Research organizations, private and universities
- Government agencies

General Approach

This report attempts to give as broad a coverage as possible without deviating into the areas of aerodynamics. Perhaps the emphasis has been greatest in the design facets such as material selection, analysis, structural concepts, and fabrication techniques. The intention has been to give as concise a picture as possible of the present state-of-the-art in regard to reinforced plastics in the aircraft industry.

SECTION 2. SUMMARY AND CONCLUSIONS

PLASTIC AIRCRAFT DESIGNS

Based on a review of present practices in the aircraft industry, of the materials currently available, and of the designs proposed, it is concluded that an "all-plastic" aircraft can be built. The all-plastic aircraft is considered to be one in which the wings, fuselage and empennage are made from reinforced plastics.

The question immediately arises as to what advantages are to be gained by building such an aircraft. Presumably, construction and flight testing of a plastic aircraft will yield information relating to the expected performance of fiberglas, allow closer approximations of costs and tooling requirements, and indicate any problems arising in the design or manufacture. It is believed that much of this information would not be forthcoming or would add little to what is already apparent. The major considerations, the reliability and reproducibility of a fiberglas aircraft, would remain unresolved.

It is clear that most of the current plastic aircraft designs are closely affixed to metals technology and thinking. When plastic aircraft designs are little more than duplications of metal counterparts, few advantages can be expected. It can be assumed that a fiberglas prototype built to a metallic aircraft design would be 5% to 10% heavier than an aluminum aircraft.

The properties attained with present fiberglas composites are adequate for normal aircraft use, but the variability, brought about by a multiplicity of factors, is too great for fabrication of highly efficient structures. Similarly the current hand lay-up manufacturing methods are satisfactory for non-critical parts or low production rates, but are not geared for controlled quality and reliability at higher production rates. Under these conditions performance data will be premature and will not reveal the full potential of the fiberglas composites. Unless production type tooling is used, estimates of tooling requirements and total cost are likely to be misleading.

Solution to design problems as dictated by customary metals procedures imposes weight penalties associated with excessive use of ribs and stringers, with the joining of components and in attachment of control surfaces. Use of woven glass fabrics as facings in wings and other structures leads to material inefficiencies in strength and fatigue.

Manufacturing procedures and specifications based on hand lay-up methods will not give satisfactory quality control, reliability, reproducibility or weight control.

PROPOSED PROGRAMS

The potential of the fiberglas resin composites is great enough to warrant government efforts to put these materials to practical use in aircraft structures as they have been in missile applications. Major requirements to achieve these ends include developments in raw materials, structural design and fabrication, as outlined:

- Raw Materials:

  Controlled fiber forming and finish application
  Controlled weaving of glass fabrics
  Controlled preimpregnation of fabrics, strands and unwoven fabrics
  Optimization of resin systems for various types of loading
  Development of improved sandwich core materials

- Design:

  Development of concepts which utilize directional properties of unwoven fabrics
  Elimination of major attachment problems through design
  Development of improved hard point attachments and techniques for reinforcing openings
  Adaptation of the latest developments in analytical techniques to aircraft structures

- Manufacturing:

  Development of improved continuous processes
  Adaptation of filament winding techniques to aircraft structures
  Development of improved pre-impregnating processes to reduce material handling and allow greater uniformity
  Adaptation of existing non-destructive testing techniques to quality control and field inspection

For early implementation of the proposed program, the design and development of a fiberglas wing is suggested. Preferably, several designs could be investigated. Each design could be closely related to material selection and manufacturing procedures so that reliability and quality control can be established. Back-up programs should include investigation of the variability in raw materials and improved fabrication techniques.

Under conditions of optimum materials, designs and processes, the major advantages of reinforced plastics will include: substantial weight saving, improved fatigue performance with no catastrophic failures, reduced number of total parts, possible complete structures in one assembly, lower costs, and an automated and controlled manufacturing process.

PROGRAMS IN OTHER AREAS

It is noted that the more recent government development programs related to filament wound missiles and deep submersible pressure hulls are not sufficient to meet the needs in reinforced aircraft structures. There has been some carry over, notably the development of the stronger S-glass, high temperature resins, and improved analytical procedures for composite materials. Since the successful culmination of the Polaris A3 development program there has been a significant reduction in the number of programs related to the glass reinforced materials. The developments suggested above will serve to put another segment of the reinforced plastics industry on a sounder engineering basis. Improvements with the fiberglas composites will be instrumental in the advancement of future composites such as those containing boron filaments, beryllium wire, silica filaments and the inorganic matrices.

APPLICATIONS

Fiberglas reinforced plastics have been successfully applied in radomes where their use is mandatory due to a combination of electrical and structural properties. Helicopter rotor blades and propellers, though not in extensive use, represent efficient designs and adaptation of the composite materials. The application of fiberglas sandwiches for skin panels in military and commercial aircraft has shown a rather rapid growth. Generally these developments have been dictated by a manufacturing need for lower costs in parts which are difficult to shape in metals. There is a trend to increase use in such secondary structures as tail booms, rudders and helicopter canopies. Aside from early developments and small commercial aircraft, there have been no fiberglas primary structures such as wings or fuselages. However, all of these applications, except for the rotor blade, make negligible contributions to the building of an all-plastic aircraft or primary structure.

MATERIALS

Most of the fiberglas in aircraft has been in sandwich structures. The facings are woven fabrics, particularly style 181 cloth. There are few instances where the unwoven fabrics have seen use. The newer weaves, including the high modulus fabrics, are beginning to find favor. Although S-glass has not been used in aircraft for economic reasons, it is being considered in all new parts and developments. Core materials are predominantly aluminum honeycomb, though in radomes and temperature sensitive areas fiberglas honeycomb cores find service. Resins are equally divided between polyesters and epoxies. The phenolics and silicones find limited application for high temperature service. In all major applications, reinforcements and resins are purchased as preimpregnated materials.

Improvements in materials since the begining of the reinforced plastics industry have been at a gradual pace, except for the improved strengths obtained with the advent of S-glass. A number of improved epoxy resins have been developed within the past five years, and improved glass finishes and coupling agents are now available. Two types of fluted core material have been developed but have not been used to any extent.

Development of advanced reinforcements and newer resins is receiving enough attention in present government programs. If fabrication of primary aircraft structures were undertaken, additional programs would be required to optimize existing materials and to control their manufacture.

MECHANICAL PROPERTIES

The fiberglas reinforced materials, in general, have demonstrated satisfactory mechanical properties for aircraft use. They are characterized by high tensile strengths, but compressive strengths, which are more important in aircraft, are somewhat lower. Investigations to improve compressive properties are being conducted only in relation to Navy programs for deep submersibles. The low modulus of the fiberglas composites has not appeared to be a handicap. Flexural rigidity is gained by increasing the section modulus as in a sandwich structure. There have, however, been several attempts to improve the modulus of the reinforcing fibers. Air Force programs are now in progress to develop higher modulus glasses as well as the continuous boron filaments.

The fatigue strength of the composites is in the order of from 25% to 30% of initial ultimate strength. Indications are that the S-glass composites have improved fatigue resistance, compared to E-glass, and will allow continuous use at higher stress levels. It is felt that studies of failure mechanisms in fatigue and compression and investigations into the micro-mechanical behavior at the glass/resin interface will lead to substantial improvements in the mechanical properties of the composite materials. Studies as to cumulative damage effects and fatigue life prediction have not been conducted with the fiberglas materials.

There are a number of problems associated with standard tests for plastics and the methods for obtaining reliable design data. It is noted that the aircraft companies have managed to circumvent these problems and have devised their own methods for substantiating design data. A considerable amount of structural testing is required for verification of preliminary designs and of allowable design stresses. The orthotropic analytical procedures appear promising as the most satisfactory methods for predicting the mechanical properties of the reinforced composites.

SECTION 3. APPLICATIONS OF FIBERGLAS COMPOSITES IN AIRCRAFT STRUCTURES

Historically, plastics have been applied in aircraft for at least thirty years. During the 1930's, paper, canvas and asbestos-base phenolic components were used for such nonstructural parts as fairings, fillets, pulleys and cable conduits. In most instances these parts were machined from rod, tube or laminate stock. Occasionally more complex pieces such as engine baffle plates were molded to shape. The fiberglas-polyester composites, which became available during World War II, were tested in wing and fuselage prototypes and found immediate application in airborne radomes. Since then, the development of improved resins, sandwich cores and reinforcements extended the usage of the composite materials. Table 3-1 lists aircraft production parts which have been fabricated from fiberglas composites.

A review of fiberglas applications has revealed certain definite trends:

- Fiberglas composites have become recognized and accepted materials for radomes and other radar transparent parts. The only limitation is at upper temperature ranges.
- Helicopter rotor blade development has reached a stage where fiberglas appears to be a practical structural material for present designs. For the advanced designs with varying blade cross-sections, fiberglas is the most suitable material from the standpoint of fabrication and cost.
- Fabrication and testing of fiberglas propeller blades has demonstrated their use to the extent that they can advantageously replace metal blades.
- Use of fiberglas for such secondary structures as fairings and leading or trailing edges, began as a simple replacement of metal parts usually motivated by manufacturing or cost considerations. This practice has proceeded at an increasing rate so that as much as 30 to 35 per cent of an airframe surface may be fiberglas in designs now under consideration.
- Improved design techniques have permitted construction of such parts as tail booms, tail assemblies, and helicopter canopies.
- The cases where fiberglas has been used as the primary structure material for wings or fuselage have been limited to prototype or small commercial aircraft.

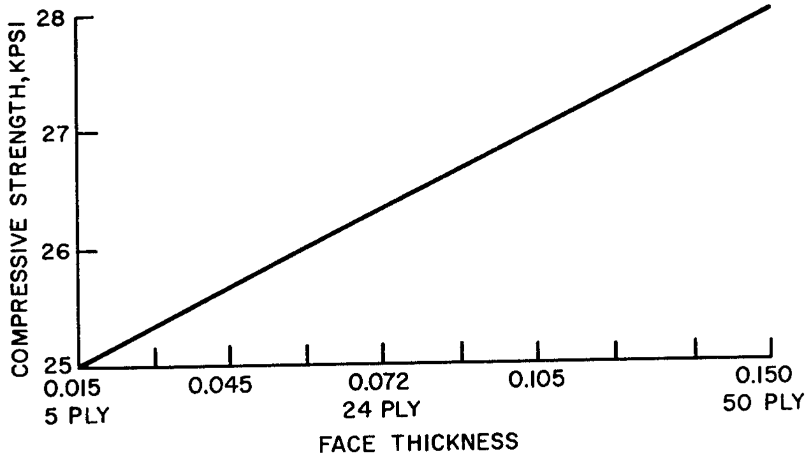


Figure 3-1. Compressive Strength of Sandwich Panels, 112 Cloth Facings, CCA Core

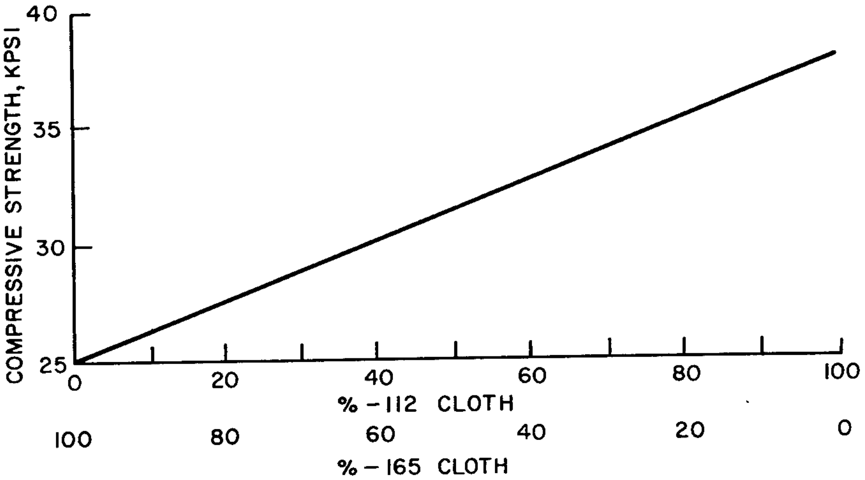


Figure 3-2. Compressive Strength of Sandwich Panels, Various Percents of 112 and 165 Cloth, CCA Cores

EARLY DEVELOPMENTS

The Vultee BT-15 aft-fuselage development began in 1943 at Wright-Patterson Air Force Base. This section was fabricated as a sandwich with glass cloth/Plaskon polyester facings and balsa wood core. Static tests were undertaken in 1943 and flight tests the following year. Results indicated that newer core materials were needed to realize suitable structures with the fiberglas facings (11, 12, 223).

Development of the AT-6C wing outer panel, also at Wright Field, was conducted over a period of years (223). Sandwich constructions were made from 112 glass cloth or a combination of 112 with a unidirectional 165 cloth and cellular cellulose acetate cores. The wing was flight tested for 1621 hours and 1645 landings following repair of minor surface cracks detected after 245 hours. The strengths obtained in sandwich panels are shown in Figures 3 - 1 and 3 - 2. Design allowables established are shown in Table 3 - 2.

TABLE 3 - 1

Aircraft Parts Fabricated from Fiberglas Composites

Radar Transparent Elements:

- Radomes
- Radome support frames
- Tail empennage
- Antenna

Fuselage Components:

- Canopies
- Cargo liners
- Doors
- Coverings and fairings
- Decking
- Wheel housing and doors
- Air ducts

Empennage Assemblies:

- Rudder tips and assemblies
- Rudder fairings and weather seals
- Tail booms and cones
- Tail spars and fins
- Leading and trailing edges
- Control surfaces

Wing Components:

- Aerodynamic fences
- Ailerons, flaps and spoilers
- Acoustical panels
- Fairings
- Closure panels
- Skins
- Landing gear shock struts
- Leading edges
- Trailing edges
- Wing tips
- Fuel tanks

Power Plant Assemblies:

- Engine air ducts
- Engine cowlings and inlet rings
- Propellers
- Rotors and rotor blades
- Engine exhaust ducts and diffusers
- Engine fairings
- Engine compressor blades

TABLE 3 - 2

Allowable Stress - AT-6C Wing Panels (223)

Property	112 Fabric - K psi	165 Fabric - K psi
Compression	20	35
Tension	30	35
Bearing	20	-
Vertical shear	15	-
Bond shear	1.5	-

TABLE 3 - 3

Typical Constructions for Airborne Radomes

Aircraft	Radome Type	Material
F-82	Nose	Solid laminate, fiberglas/polyester
F-86	Nose	Sandwich; fluted core, fiberglas/polyester facings
F107	Nose	Solid laminate
EC-121H	Belly	Sandwich; nylon phenolic honeycomb core, fiberglas/polyester facings
B-52	Nose	Sandwich; honeycomb core, fiberglas/ polyester facings
F-104	Nose	Filament-wound, E-glass/epoxy
W2F-1	Rotodome	Sandwich; honeycomb core; 181 glass cloth/epoxy facings
E-2A	Dorsal	Filament-wound, E-glass/epoxy
F-111	Nose	USAF, filament-wound, E-glass/ polyester
F-111	Nose	USN, sandwich; HRP honeycomb core, fiberglas/epoxy facings
F-111	Horizontal stabilizing	Solid laminate, 181 glass cloth/epoxy
B-58	Nose, search	Solid laminate
B-58	Doppler	Sandwich
DC-8	Doppler	Sandwich; fluted core

A wing for the Martin KMD-1 was designed, fabricated and tested at Cornell Aeronautical Laboratory under Bureau of Aeronautics contract (298). Here again , sandwiches of glass laminates and foam cores were used. Fabrication techniques, costs and production times were investigated. It was concluded, after static tests, that the plastic wing with additional reinforcement near the attachment fittings was comparable to the metal honeycomb wing.

RADOMES

The fiberglas radome represents one of the most successful applications of this material in the aircraft industry. Since the early glass cloth/polyester radomes in 1943, developments in fabrication and materials have continued. Radomes are made as "monolithic" glass cloth laminates or as sandwich structures with honeycomb, foam or fluted cores and glass cloth facings. Processing methods include hand lay-up, matched die molding, autoclaving vacuum bagging, and filament winding. Most of the available resin systems, such as the phenolics, phenyl-silanes, silicones, epoxies, and the TAC polyesters have been tried, although present emphasis remains with the polyesters and epoxies, due to their dielectric properties.

Critical parameters in radome design are the loss tangent and dielectric constant of the laminate measured at "X" band frequency. Variation of these properties with temperature and the strength of the laminate are secondary considerations. The major production requirement is that a reproducible electrical thickness of the radome wall be maintained. The electrical properties of the resin are controlling in determining this wall thickness. Improved electricals allow the laminate to be made thicker and stronger. With poor electricals, the laminate may be too thin to withstand the aerodynamic loads. Consequently, a trade-off is made between the electrical and structural requirements. Figure 3 - 3 illustrates the effect of dielectric constants on the wall thickness of a laminate for several incident angles and frequencies.

The fabrication of these precise wall thicknesses is implemented by strict control of resin content, resin distribution, and void content. Filament winding and autoclaving have been found to yield more uniform thicknesses and have been adopted more frequently in the latest radome designs. Regardless of processing, surface grinding is nearly always necessary. In a Grumman radome for the E-2A a filament-wound laminate, specific gravity has been found to range from 1.98 to 2.02 while holding thickness tolerances. Typical radome applications are listed in Table 3 - 3.

With supersonic flight, the composite properties at elevated temperatures have become limiting. The Air Force has sponsored several programs to raise the operations level of these materials. Of interest are the polybenzimidazole resins (PBI) which may be effective to 900° - 1000° F. A fiberglas-alumina laminate now being developed is reported as retaining properties up to 1400° F. Tables 3 - 4 and 3 - 5 list some resins being evaluated and their strength retentions.

TABLE 3 - 4

Resin Systems Evaluated for High Temperature Use (47)

RESIN TYPE	DESIGNATION	SUPPLIER
Diphenyl oxide derivative	Experimental Resin QX-2682-1	Dow Chemical
Diphenyl oxide derivative	DORYL	Westinghouse
Polyphenyl epoxide	KOPOX 170 & KOPOX 171	Koppers
Phenolic	BLS-3135	Union Carbide
Phenolic	101X	Ironsides
Glass Resins	Type 100	Owens-Illinois
Polyimide		DuPont
Aromatic Amide-imide	ARAMIDYL	Westinghouse
Polyimide	SKYGARD 700	Monsanto
Phenylene Sulfide		Dow
Polyisocyanurates		Monsanto
Phenyl Silane	EC-205	Evercoat
Phenyl-Aldehyde	4-76	Ironsides
Polyphenyl	DP29-27	Ironsides
Epoxy Novalac phthalocyanine cure		Shell

TABLE 3-5

Summary of Screening Tests of Heat Resistant Resins (47)

Material	Flexural Strength, KPSI	% Strength Retention At 600°F After Aging At 600°F			
		1/2 Hour	24 Hours	48 Hours	100 Hours
101X Phenolic	86.1	21.5	66.8	61.8	64.8
SKYGARD 700	61.4	61.9	65.2	64.2	44.3
DPO QX-2682-1	70.0	41.7	76.6	65.2	32.8
KOPOX 170	64.7	49.8	53.1	56.0	41.3
Phenyl Aldehyde 4-76	63.3	58.4	52.1	49.5	39.8
KOPOX 171	52.1	48.1	46.6	60.3	56.1
Phenyl Silane EC-205	81.3	36.1	10.0	---	---

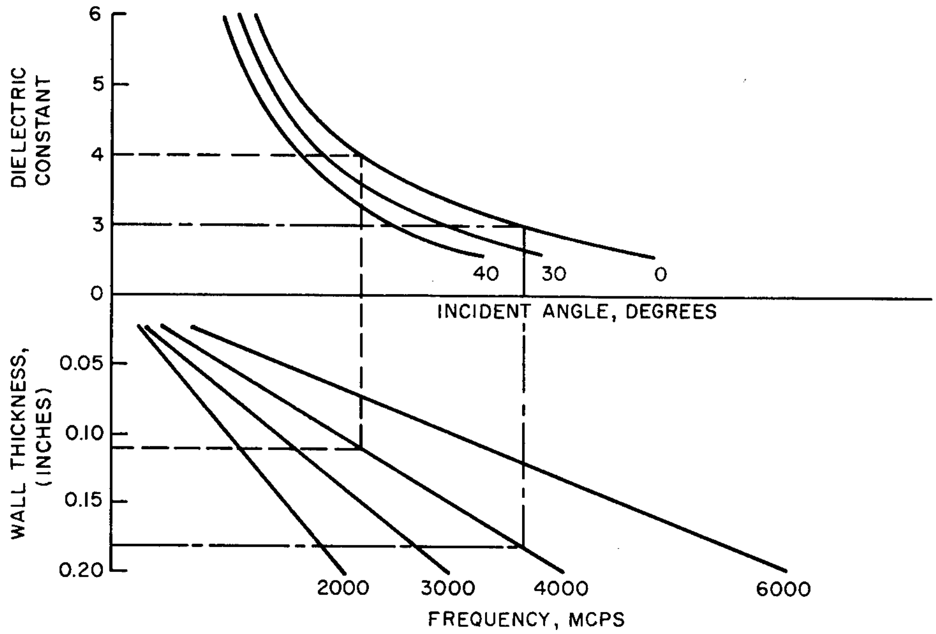


Figure 3-3. Thinwall Radome Design Nomogram

HELICOPTER ROTOR BLADES

Developments in fiberglas rotor blades have been taking place for over 15 years. An early Air Force program at Cornell Aeronautical Laboratory in 1950 was followed by a contract with Piasecki Helicopter for the H-21 blade. The development pattern has proceeded as a gradual replacement of metal or wood parts until arriving at the present "all plastic" rotor. Generally, these plastic blades contain metal either at the root end, as aluminum honeycomb, or as an abrasion strip. Sandwich structures are used extensively with either aluminum honeycomb or a polyurethane foam serving as core material. Companies engaging in these developments, beside Piasecki, include Parsons, Boeing-Vertol, Kaman and Dama Plastics. These latter three have all-plastic versions. Present production designs, although employing fiberglas skins, have metal spars. The all-plastic blades have been whirl tested successfully and in the case of Kaman, flight-tested. The significance of these fiberglas rotors lies in the design aspects which are discussed more fully in Section 8.

FIBERGLAS PROPELLER BLADES

Fiberglas propeller blades have been fabricated at Curtiss Wright and Hamilton Standard. The Curtiss blades are constructed as a monocoque shell molded over a steel shank and filled with polyurethane foam for added rigidity. The fabrication has been described in Modern Plastics (18). This process makes use of an inflatable Teflon mandrel. Prepreg cloth patterns are laid-up on the mandrel and the assembled lay-up is placed inside a clam-shell mold for pressurization during cure. Directional properties to fit expected load conditions are achieved by varying the amounts and directions of the styles 181 and 143 epoxy prepregs. Finishing operations include the addition of a foam-filled fairing around the shank, bonding of a stainless steel leading edge for erosion resistance, and the winding of hoop wraps around the shank for added strength in that region. The core is foamed in place. De-icing equipment can be included in the molding step when required.

Six designs, of five diameters ranging from 10 to 15.5 feet, have been fabricated. These blades are used on the X-19, the X-100 and the Canadair CL-84. The major advantages of the FRP blades as compared to hollow steel or solid aluminum alloy forgings are lower weight for equivalent strength and lower fabrication costs. It is noted that these advantages become greater with larger propeller size. For example, the weight saving at a 7-foot diameter is about 15 to 20 per cent. At a 15 foot diameter the weight reduction is nearly 50 per cent. Lower costs accrue from cheaper tooling and a simplified fabrication process. Other advantages are: corrosion resistance, easy repair, and low lead time for procuring tooling.

Endurance properties, particularly fatigue, are controlling in blade design, and the company has accumulated a backlog of data on these properties. Their testing program is based upon material evaluation followed by full blade vibratory tests, whirl tests and eventual flight tests. The testing program includes:

- Preparation of S-N curves and modified Goodman diagrams. Establishment of material design limits on the basis of 50 x 10^6 cycles.
- Coupon testing for endurance; long term creep rupture; tensile, flexural and compressive strengths and moduli; and impact resistance.
- Free-flapping vibration at 2 and 3 times design loads for as high as 104 x 10^6 cycles.
- Propeller testing on a gyroscopic test rig to simulate 1XP vibratory loads.
- Normal flight tests, hover and transition on the X-100.
- Comparative abrasion tests and compatibility with aviation fuels.
- Full-scale design-load and overload retention fatigue testing.
- Evaluation of blades with deliberately unbonded steel shanks.
- Lightning tests and development of blade grounding methods.

As a result the blades are designed for a "perpetual life." In laboratory and flight tests, the blades have withstood continuous stresses up to 12000 psi and tip speeds of 1200 ft/sec.

The original development program began in 1956 and has been active since. The company estimates that about 75% was company sponsored and the remainder on government development or purchase. Curtiss is presently working on the first phase of a Bureau of Weapons contract which will lead eventually to an all fiberglas shank. Present interests are directed to S-glass which it is believed will give further weight reductions; and to investigation of unidirectional prepregs which it is claimed have superior fatigue properties.

Other FRP applications at Curtiss include a foam filled FRP propeller spinner and X-19 parts, such as the nose radome, the tail radome, and fairings.

Hamilton Standard has had eight blade designs of which three are qualified. Four hundred blades have been manufactured and 11,000 hours have been accumulated in test. Their blades are developed for the XC-142 and P2-V among others. The largest of these is approximately 15-1/2 feet in diameter.

Style 181 cloth epoxy has been used with weight savings up to 45%. In some of their designs, the structural part of the blade is metal, with fiberglas forming the airfoil.

MISCELLANEOUS EXTERNAL PARTS

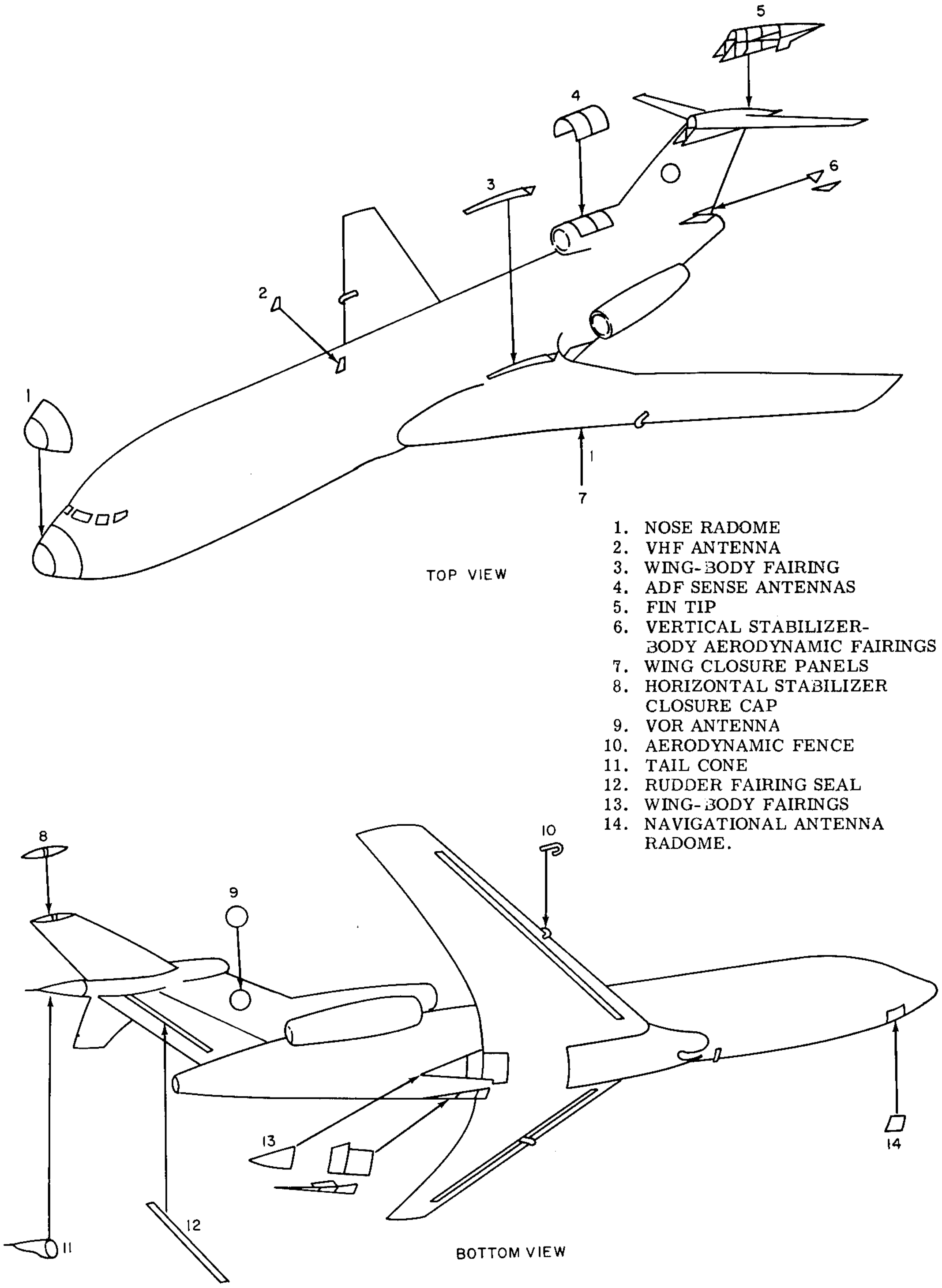
The use of fiberglas reinforced materials in aircraft has shown a marked increase within the past few years. The greater part of the growth has gone into secondary or accessory structures. Only a few parts involving this composite material could be considered primary or highly stressed. The increase is noted in commercial as well as military aircraft.

A major incentive has been the ease of manufacturing combined with lower costs. The lower costs are usually associated with shaped parts having compound curvatures or sharp radii, although similar trends have taken place with simple shaped skin panels. Here a reduction in the number of pieces that make up the panel has been effected .

In addition to the suitable dielectric and electrical characteristics already mentioned, the composite materials possess other properties which make them attractive for certain applications. These are:

- Sonic resistance: High damping in plastic structures has not been established. Due to monocoque construction envisioned for plastic structural design, it is doubtful that there will be any advantage.
- Aerodynamic resistance: Fiberglas panels do not "oil-can" under aerodynamic buffeting as do metal panels of the same weight.
- Impact resistance: In cargo hold areas or in fairings adjacent to wheel wells, fiberglas has withstood impact where metals are dented or cracked.
- Heat resistance: Certain reinforced plastics are used for heat insulation or in high temperature areas where the usual aluminum alloys lose strength.
- Aerodynamic drag: FRP lends itself to maintenance of a geometric aerodynamic shape better than does metal.
- Special applications: There are some cases where the reinforced panels are designed for aerodynamic loads, but are required to flex when subjected to greater loads. In other cases the higher tensile strengths of FRP have resulted in lighter weight ducting.

Indications of the amount of material involved is obtained from the following figures. The B-47 bomber has 1000 pounds of plastic compared to over 5000 pounds for the Boeing 727. Figure 3-4 shows plastic parts in the 727. Boeing has estimated the surface area for several aircraft to be:



1. NOSE RADOME
2. VHF ANTENNA
3. WING-BODY FAIRING
4. ADF SENSE ANTENNAS
5. FIN TIP
6. VERTICAL STABILIZER-BODY AERODYNAMIC FAIRINGS
7. WING CLOSURE PANELS
8. HORIZONTAL STABILIZER CLOSURE CAP
9. VOR ANTENNA
10. AERODYNAMIC FENCE
11. TAIL CONE
12. RUDDER FAIRING SEAL
13. WING-BODY FAIRINGS
14. NAVIGATIONAL ANTENNA RADOME.

Figure 3-4. FRP Structures on the Exterior of the 727 Airplane

KC-135 - 2-3% surface FRP
707 - 5% surface FRP
720 - 20% surface FRP
727 - 25% surface FRP

From the standpoint of number of parts, the F-111 has over 200, the Sikorsky CH-53 helicopter from 500 to 600 parts, and the Fairchild F-27 Friendship has 450 parts.

Boeing has also stated that in their version of the C-5A (CX-HLS), designs call for fiberglas sandwich with HRP fiberglas core in large sections of the wing and fuselage. See also Section 10, Table 10-1, for distribution of reinforced material in a number of aircraft. Some typical applications are briefly summarized in the following:

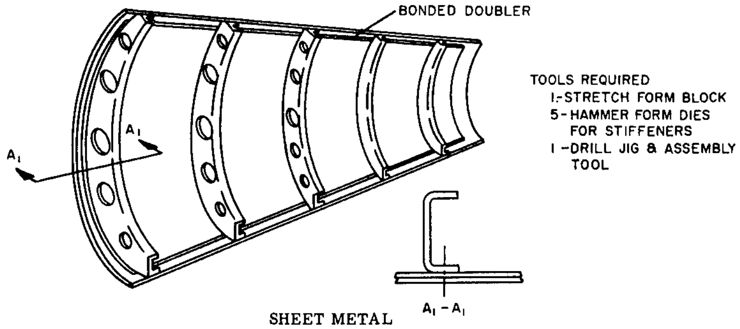
Skin panels: Panels for wings, fuselage or tail section are fiberglas sandwiches with either aluminum or fiberglas honeycomb core. Such applications are numerous. As a special example, a modification of the 720 wing utilized an FRP panel for the redesigned section. Designs closely parallel metal construction, but are somewhat stiffer and require less ribs or other stiffeners. Figure 3-5 compares a typical metal.

Tail cone, 707 and KC 135: This cone is subjected to severe mechanical and sonic vibrations. Originally of conventional stringer-stiffened sheet metal, the cone failed in as short a time as 8 hours. It was redesigned as a fiberglas fairing and core sandwich. Some aluminum was used in the composite, as the face of the production break, andfor flare racks. The assembly was completed by adhesive bonded and mechanically fastened joints. One of these cones has accumulated over 16,000 flight hours. Weight was reduced 50% and the cost reduced from $2600 to $1600 per unit (244).

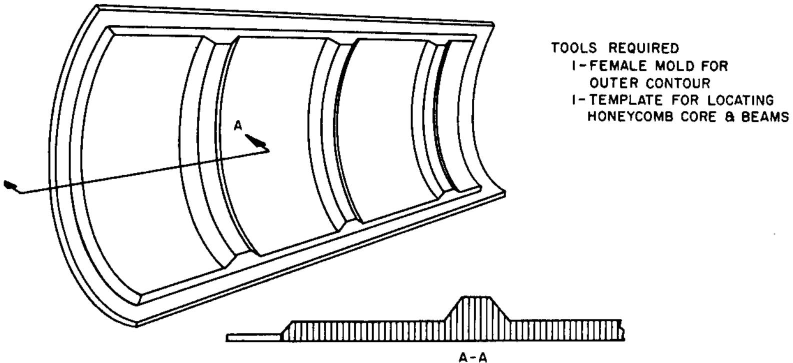
Leading edge C-141 horizontal stabilizer: This part is made up of 8 sections, totaling 50 feet in length. De-icing elements are molded into the structure. Other uses of FRP for leading and trailing edges are now rather common.

Fatigue resistance skins, B-58: These skins are composite sandwiches with fiberglas facings, aluminum and stainless steel honeycomb core. The sandwich is placed between two metal layers.

Grumman E-2A rudder: To improve the performance of their E-2A rotodome, Grumman has redesigned sections of the tail assembly. These parts include the two outboard fins; the two outboard rudders with tabs; a right inboard fin, rudder and tab; and a left inboard fin. Construction is fiberglas sandwich with style 181 cloth facings and fiberglas core. The sandwiches are reinforced with aluminum frames. The core is foam-stabilized in attachment regions. The assemblies have been flight tested and will be installed on the E-2A.



SHEET METAL



PLASTICS SANDWICH

Figure 3-5. Comparison of Sheet Metal Construction and Fiberglas Reinforced Plastics Sandwich Construction (69)

Aft fuselage - Neptune patrol bomber: This bomber of the early 1950's was modified by a fiberglas sandwich aft fuselage replacement. Several hundred were installed. Static tests went to 300% of design load. Tool costs were estimated to be 27% of the previous metal tooling. The approximate size was 20 feet long by 6 feet deep by 3 feet wide. Several of these are reported to be still in use (240).

Tail section, X-21, Northrup-Ventura: This structure is a vacuum-bagged hand lay-up of style 181/polyester prepreg. It is reinforced by hat-section ribs which are foam-stabilized.

Canopy, Sikorsky CH-53: The largest FRP part in the CH-53 is the single-piece canopy. It is constructed as an autoclaved laminate reinforced with foam-filled hi-hat ribs. It varies from 3 plies to as many as 17 plies in more highly stressed areas. Style 181 cloth has been used, but Sikorsky is tending towards newer weaves such as styles 7581 with 550 finish. Directional properties are obtained with 143 cloth and thin plies with 128 cloth.

Diffuser for piston motor: Sikorsky has also developed a replacement diffuser for earlier helicopter models. This is a fiberglas-phenolic molding and is subjected to vibration at 500°F. Previous steel and titanium diffusers have failed in a few hours. The plastic version is performing satisfactorily.

F-111 parts: The three radomes on this aircraft are listed in Table 3-3. Other typical fiberglas applications are described in Table 3-6.

Other airplane parts: The following parts are known to be plastic, but specific details have not been obtained.

Stabilizer, Martin P-4M Marlin Seaplane

Wing tip, Boeing B-52

Vertical tail section, Douglas DC-8

Spar, Douglas DC-8

Heater intake ducts, DeHavilland Buffalo

Wing tips, Buffalo

Door frame, Buffalo

DOW (WINDECKER) PLASTIC WING

The Dow Chemical Company has built a plastic wing and it is being tested on a Monocoupe 90 AL-115 airplane. It is built as a single integral structure to which the fuselage and landing gear are attached. Skins are solid glass cloth/epoxy laminates. The core is a rigid polyurethane foam, machined from a precast block. Spars run through slots in the foam core. Fuel storage tanks are contained within the foam. Provisions are also made for electrical wiring and control mechanism within the core. The design is similar in concept to the earlier Cornell Aeronautical Laboratory wing for the Martin KMD-1. These types of structures do not appear to be applicable to Military aircraft because of low shear and compressive strengths in the foam core sandwiches.

ALL-PLASTIC AIRCRAFT

The Taylorcraft Model 20, certified by the FAA in 1955, has been referred to as an all-plastic aircraft. Actually reinforced plastics were used mainly as coverings for the substructure. Fuselage covering, wing and stabilizer skins, engine cowlings, gas tanks, doors, wheel parts and others were of fiberglas construction. The fuselage covering was molded in halves and bonded to the steel substructure.

The Marvel and Marvellette are also frequently mentioned as examples of plastic aircraft. These are experimental aircraft in which reinforced plastic were utilized to facilitate the test procedures. No particular significance is given to the plastic structures in these planes.

Development of the Piper Aircraft plastic plane began in 1958. The plane has been flying since April 1962, and is presently equipped with its second set of wings. The company does not plan to go ahead in production. They will, however, continue development of FRP parts such as wing tips, cowls and wing tanks.

The aircraft was constructed as a fiberglas sandwich with paper honeycomb core. Fabrication was by hand lay-up. Costs turned out to be relatively high and some people within the company attributed this to the hand lay-up method. The polyester resin used did not perform well in regard to weatherability. It is believed that this deficiency can be remedied. The paper honeycomb core tended to pick up moisture and to vary in weight. This was due to some surface porosity and leakage through the sandwich close-outs. In retrospect, the company feels that aluminum honeycomb core would have been satisfactory and would not add materially to the total cost. It was not possible to obtain good estimates of production tooling costs. The project was run on an experimental basis and experimental tooling costs could not be correlated with production tooling.

Table 3-6. Typical Fiberglas Application in F-111 Aircraft

Application	Type of FRB	Facing Fabric	Resin	Honeycomb Core	Loading
Ram air duct	0.35" sandwich, 0.02" facing	181	Conolon 506, phenolic	8 lb aluminum	36 PSI ultimate internal pressure at 413°F
Boundry air duct	1.60" sandwich 0.04" facing	181	Conolon 506, phenolic	6 lb aluminum	28 PSI ultimate internal pressure at 291°F
Vertical stabilizer leading edge skin	Sandwich, varying thickness	120/181	Conolon 506, phenolic	5.5 lb HRP	20 PSI side pressure at 280°F
Vertical stabilizer tip	Solid laminate	181	Conolon 506, phenolic	----	12 PSI ultimate side pressure at 280°F
AF wing tip	Solid laminate with ribs	181	Epoxy	----	8 PSI internal pressure at 250°F
Radome support frame	0.100" solid laminate	181	Epoxy	----	500 lb in shear at 270°F

SECTION 4. RELATIVE MERITS OF FIBERGLAS STRUCTURES

The relative advantages or disadvantages of fiberglas structures are divided for convenience into three general categories - performance, manufacturing, or design. It is apparent that such advantages or disadvantages to be meaningful must apply to specific designs for specific components, and that in many cases trade-offs are indicated. The intention here is to summarize the more important considerations. Details are given in other sections. Since experiences with fiberglas in aircraft have been mainly with secondary unstressed structures, conclusions are based on practices in other areas of the plastics industry and on the anticipated potentials of the material as well as the current practices within the aircraft industry.

PERFORMANCE CHARACTERISTICS

Higher strength to weight in tension, compression, flexure and axial load fatigue have often been quoted as advantages of fiberglas composites. Usually these estimates are based on performance of filament wound structures and tend to be over-optimistic. The strength to weight will depend on the design concepts and when metal designs are duplicated in fiberglas, no weight savings can be expected. When the designs take greater advantage of the unique properties of the reinforced composites, weight savings are possible. As an approximation, weight savings of from 5% to 15% may be anticipated.

Actual experience with fiberglas components has shown conflicting results. Skin panels usually weigh the same in metal or fiberglas. In more complex structures such as tail booms, weight savings have been effected, although there are exceptions in both cases. Proposed designs for all-plastic aircraft have also shown conflicting weight estimates. These range from a weight decrease of 12.5% to an increase of 7.5%. In a COIN-type aircraft, structural parts represent about 50% to 55% of the total weight, and a 10% change in structural weight is equivalent to a 20% change in payload.

The fatigue characteristics of reinforced plastics have not been investigated in enough detail to predict performance in aircraft. It is likely that fatigue failure will not be catastrophic and may be detected in advance of failure. Woven E-glass/epoxy laminates have approximately the same specific fatigue strength at 10 million cycles as does 7075-T6 aluminum. Unwoven crossplied laminates made with E-glass have about a 25% higher fatigue strength than the aluminum, and crossplied unwoven laminates with S-glass show a 60% increase. Limited testing of fiberglas rotor blades and propellers is promising as to fatigue life. Fatigue tests on parts such as fin-to-rudder fairings have demonstrated superior fatigue performance for fiberglas.

Improved damping characteristics have been claimed for the fiberglas composites. The energy dissipation of a structure can be attributed to material damping and damping due to interactions at joints or interfaces. In the commonly used sandwich structures, most of the damping is due to the viscoelastic adhesives and low modulus core materials in the composite. Comparisons should then be on the basis of contribution of fiberglas facings or fiberglas core materials to the overall damping of a particular sandwich construction. Such data is limited. Some fiberglas materials have about a 10% higher damping coefficient in axial stress, depending on the resin system. Total damping for fiberglas structures may be expected to be slightly greater than for a similar aluminum sandwich.

Tests on skin panels at Boeing and others have shown that fiberglas panels have a greater resistance to sonically induced stresses. Sikorsky, using a monolithic fiberglas construction for the canopy of the CH-53, claims less vibration and improved damping in this structure.

The molding processes for fiberglas lead to smoother surfaces which will result in less aerodynamic drag. The absence of rivets and lack of oil canning further improves the surface. Data on the effect of such surfaces is limited. Figure 4-1 shows the variations in subsonic zero-lift drag coefficients and compares lifting surfaces and fuselage surfaces made of aluminum and fiberglas. The fiberglas is shown as approaching a theoretically smooth airfoil. The general conclusion reached is that it is easier and cheaper to attain an aerodynamically clean surface with the plastics than with the metals. With the conventional metal fabrication methods, the cost of producing an aerodynamically smooth surface can be prohibitive.

Three studies have been made of the radar transparency and radar cross-sections of fiberglas composites. It is concluded that little reduction in detection can be achieved by the use of fiberglas. The addition of radar absorbing materials into the resin system is a possible help, but greatest gains can be obtained by changes in structural configurations. It is also questioned whether any improvement would be of value in light of the short range, low altitude mission proposed for the COIN aircraft.

The effect of natural weathering on reinforced plastics has been based on coupon type testing, and such data as has been compiled can be misleading. Changes in strength on exposure depend on the resin systems and glass finish in the laminate, and on the region of exposure. Tests are run on tension, compression and flexure only and these are effected differently. Length of exposure is usually for three years so that data on current and presumably improved materials is not available. Epoxy resins, which are more likely to be used in critical strength areas, are least affected by weathering, as measured by strength degrations. Coupon data indicate that epoxies will show about a 10% decrease in tensile and flexural strength and no decrease in compressive strength after 3 years in the worst climatic region of exposure. With normally protected surfaces, little strength deterioration is to be expected.

The fiberglas composites have a relatively high resistance to impact. This property makes them less susceptible to denting and damage due to debris encountered during take-off or landing. Laminates have been used effectively in wheel wells or adjacent areas and in cargo compartment liners. Generally damage from impact with fiberglas is localized. The following data has been compiled by Boeing to compare relative impact strengths for several materials (27):

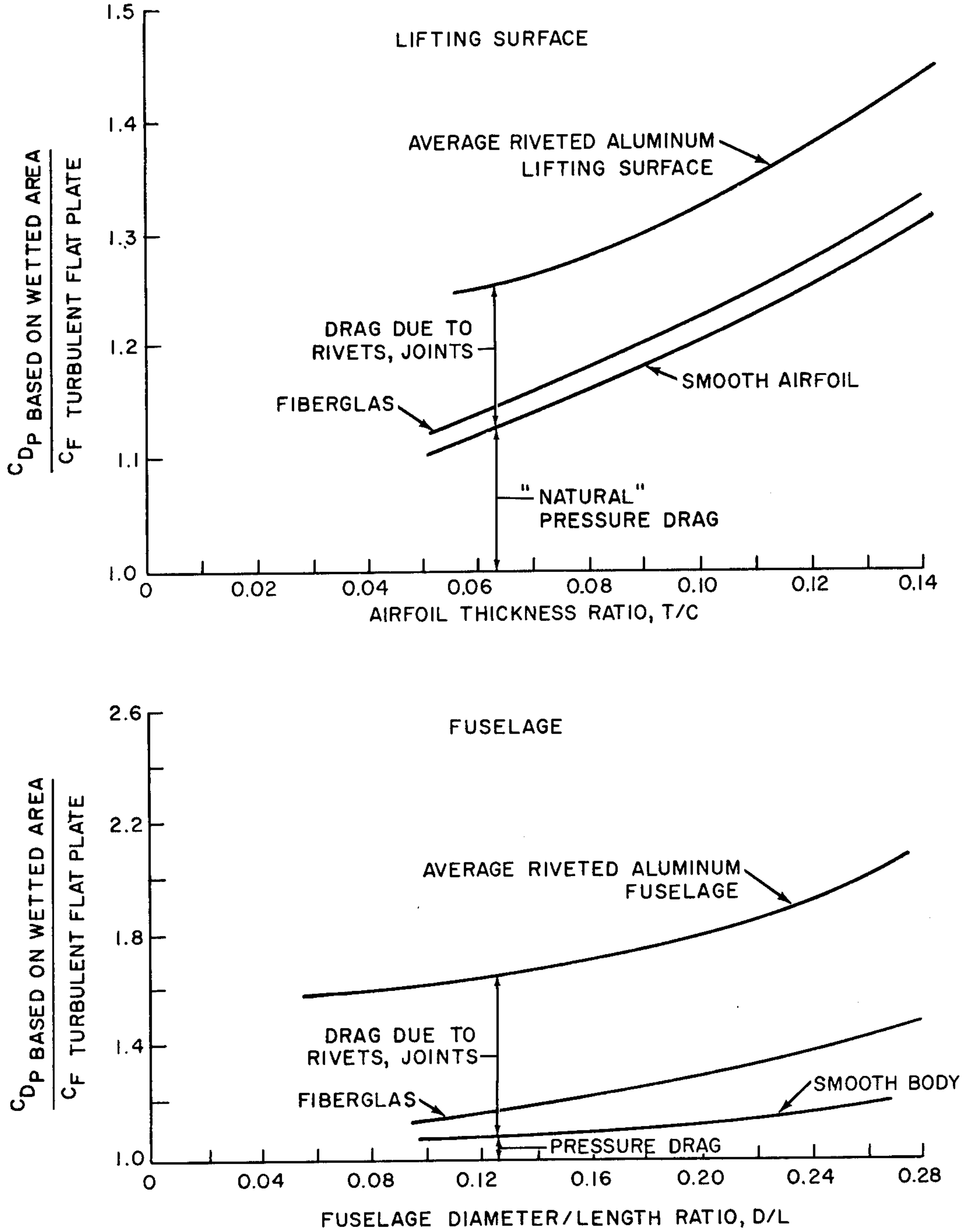


Figure 4-1. Zero-Lift Drag Multiplying Factors

Material	Relative Weight Index
Non woven reinforced	1.0
Woven fabric reinforced	1.32 - 1.65
Aluminum alloy	1.35
Stainless steel	3.89

Notch sensitivity and crack propagation are not problems with the reinforced composites as they are with the brittle metals. Crack propagation in a composite initiates in the resin phase and the glass fibers prevent further crack extension. Modifications in the resin system can also be made to increase resin toughness or flexibility and increase the stress level at which crack formation begins. With the composite materials, designs can be altered to meet situations where holes or stress risers occur. To begin with, adhesive bonded joints are more practical, thus eliminating the need for rivets or bolts. Doilies or similar local reinforcements can be used where openings exist. Such methods have been successfully employed inthe filament would motor cases. Experience with fiberglas rotor blades has indicated that problems associated with notch sensitivity can be avoided in the design.

Fiberglas laminates are being investigated for use as protective armor against small arms fire. The weight of these laminates and the types of glass fabric in the structure, however, are not comparable to aircraft laminates. The vulnerability of aircraft laminates or sandwiches, per se, has received little attention. North American Aviation has reported tests conducted with 30 caliber projectiles fired into fiberglas sandwiches. These show that penetration normal to the plane of the test samples resulted in clean holes and minimum delamination, but that delamination increases at low angles of penetration. Stresses at penetration are approximately equal to the ultimate allowable strength of the material and no catastrophic failures were found to exist (223). It is concluded that fiberglas composites offer some slight advantages as to vulnerability, but protective measures are required as with the metal structures.

What the effects of lightning will be on an "all-plastic" aircraft or on a primary fiberglas structure is not known. It is believed that in some instances damage will be severe. The practice with existing plastic parts has been to add conductors and conductive coatings to the structural surface. Similar procedures are contemplated for cases where use of reinforced plastics is more extensive. It is apparent that the problems of lightning effects and the elimination of precipitation static will need further investigations to determine resistance requirements of conductive coatings, the extent of conductor use, and the effect of configuration on lightning strikes.

Reduced maintenance is anticipated for fiberglas aircraft. This is based on a smaller number of parts, no corrosion problem, and less damage from impact or denting.

The field repair of reinforced plastic parts is not expected to present any serious problems. Methods already exist in several segments of the plastics industry to handle repair of damaged parts. Usually these methods rely upon relatively primitive equipment and would not be applicable to highly stressed areas. Improved equipment and techniques will be required so that more extensive and stronger repairs can be made. Development programs in this area are suggested.

MANUFACTURING

A major advantage in the manufacture of fiberglas components is the fact that the number of parts forming an assembly can be greatly reduced. Typical examples are the Boeing 707 tail cone, the Sikorsky CH-53 canopy, and the Boeing-Vertol and Kaman all-plastic helicopter rotor blades. A second advantage is that complex shapes which are difficult to form in metals can be more easily formed in fiberglas. Tooling costs for fiberglas have been cheaper and tools can be built in shorter lead times. As a result of lower tool costs and fewer parts required, overall costs for components have been lower. It is recognized that the cheaper tooling and lower costs apply to present low production rates for aircraft and costs may be reversed at higher production rates. Costs for expendable tooling for fiberglas, however, would be expected to remain low.

The manufacturing techniques most frequently used for fabricating aircraft parts are vacuum bagging, pressure bagging, or autoclaving. While these methods have been adequate for present production, they leave much to be desired as to reduced handling, automation and reproducibility of parts.

Establishment of adequate reliability controls in production is expected to be a major problem. Improvements and innovations in the manufacturing processes are therefore of great concern. Of the present methods available in the reinforced plastics industry, filament winding is best suited to automated control and has produced the highest quality fiberglas parts to date. This process, however, will require modifications to handle aircraft components. Improved methods of pre-impregnating glass cloths and unwoven fabrics can be of importance in supplementing the existing molding methods. Variations in raw materials, particularly the glass reinforcements, are considered as excessive. Continuation of their use will impose a heavy weight penalty or will result in reduced reliability. Variations pertain to the monofilaments as drawn, bundled strands or yarns and eventually the woven or unwoven fabrics and prepreg materials. While imperfections at each stage may not be cumulative, their effects on overall strength are unknown and they create problems as to weight control of finished structures.

DESIGN

Although many aircraft components have been successfully fabricated from fiberglas, its full capabilities have not been realized in these applications. It is apparent that the advancement of fiberglas composites to the point where they can be profitably used in primary or total structures is dependent on the development of new design concepts. Design of most existing parts has been simplified by the fact that these parts are not subject to high stresses and specific performance requirements are not severe. The early and presently proposed designs for primary structures have more or less followed contemporary metal designs. The only exception, perhaps, is the all plastic helicopter blade which begins to approach the possibilities inherent in the composite structures. It is proposed that development of newer design concepts be given full consideration. Implementation of such programs can be limited to the design, fabrication and testing of wing structures. Greatest emphasis should be placed on the reliability and reproducibility of any part designed. Joining of fuselage and wing structures is considered an integral part of the design. Close coordination of the design with the manufacturing process and the control of raw materials is essential. Concurrent with design studies, optimization of existing materials is indicated.

Consideration of composite materials in this survey has been restricted to the fiberglas composites, simply because these are the more advanced. It is implied that improvements and better understanding of the fiberglas composites will be applicable to other filamentary type composites. Conversely, as other composites are developed they can be adapted to existing processes for the fiberglas. Some work has been done with steel wire-epoxy composites. This material showed lower specific strength than the fiberglas-epoxy. It did have a higher modulus and may have some applications as local reinforcement in aircraft structures.

One of the criticisms levelled at fiberglas design is that joining and attachments are problems. It is believed that existing and proposed methods for attachment of ailerons, flaps, wing tips, rudders and similar structures are adequate. The joining of primary components may cause some problems. Such joining has been studied only in proposed plastic aircraft designs and in the few earlier trials of plastic wings and aft-fuselage. Component joining, as noted, requires development in the overall structural design.

A second criticism of fiberglas designs relates to the lack of reliable design data. It is noted that the aircraft companies have developed their own methods for obtaining design data and establishing design allowables. In some instances they have relied on Military Handbooks 17 and 23 data, or similar data is generated for various materials. Typical Military Handbook data is shown in Table 4-1. Table 4-2 lists data used by North American for laminates and sandwich panels. Table 4-3 lists Whittaker-Narmco design allowables for several thin ply laminates. Establishment of design allowables by a government agency or on contract is not considered as necessary or useful at the present time.

Table 4-1. Typical Military Handbook 17 Design Allowables

Parameters	Epoxy/181	Polyester/181
Tensile Strength, KPSI	45	38
Tensile Modulus, PSI x 10^6	2.9	2.6
Compressive Strength, KPSI	45	30
Compressive Modulus, PSI x 10^6	3.3	2.9
Flexural Strength KPSI	65	45
Flexural Modulus, PSI x 10^6	3.2	2.5
Bearing Strength, KPSI		
D/T = 1	45	---
D/T = 4	37.8	---
Shear Strength, KPSI	14	---

Density - epoxy - 0.07 lbs cu. in.
polyester - 0.065 lbs cu. in.

Table 4-2. Typical Design Data, S-Glass Epoxy (225)

Type	Compression KPSI	Tensile KPSI	Modulus PSI x 10^6	Shear Modulus PSI x 10^6
181- Sandwich	31	80	3.8	1.2
Uniply - Sandwich	40	86	4.0	1.0
181 - Laminate	63	100	3.8	1.0
Uniply - Laminate	93	98	4.2	0.7
143 - Laminate	71	95	4.0	1.0

Table 4-3. Typical Design Stresses (S-901/81/HTS Fabric and 1009 HTS unidirectional Tape) (174)

Lay-up	Thickness (mils)	Tension, Ultimate (KPSI)	Modulus, Longitudinal (x 10^6 PSI)	Modulus, Transverse (x 10^6 PSI)	Compression, Ultimate (KPSI)	Shear, Ultimate (KPSI)	Modulus (x 10^6 PSI)	Poisson's Ratio, Longitudinal	Poisson's Ratio, Transverse	Density (lb/ cu. in.)
1-ply Tape	4	165.0	6.71	1.04	92.0	7.8	0.389	0.264	0.041	0.069
1-ply Fabric	8	27.0	2.25	2.25	27.0	24.0	1.4	0.12	0.12	0.067
1-ply Tape 2-ply Fabric	20	73.0	3.74	1.85	48.6	20.8	1.06	0.168	0.094	0.067
1-ply Tape 1-ply Fabric	12	96.0	4.48	1.64	59.5	18.6	0.894	0.192	0.080	0.067
2-ply Tape 1-ply Fabric	16	119.0	5.22	1.44	70.3	15.9	0.726	0.216	0.067	0.068
3-ply Tape 1-ply Fabric	20	130.0	5.59	1.34	75.7	14.3	0.642	0.228	0.061	0.069

SECTION 5: COMPOSITE MATERIALS FOR AIRCRAFT

The established practices in the aircraft industry as to choice of raw materials are reviewed here. Although some of the older cloths and resin systems are still used extensively, primary interest is in the more recent innovations in fiber and resin technology. The material categories which are covered include the reinforcements, resin systems, sandwich cores and adhesives. Many of the early government programs dealing with reinforcing fibers, finishes and resins placed strong emphasis on aircraft applications. Since that time interest has been shifted more to missile and high temperature uses. The newer aircraft materials have for the most part been an outgrowth of these latter programs. The adaptation of S-glass fiber to unidirectional or woven cloths for aircraft structures is an outstanding example. Concurrently with basic material developments, the material suppliers have made modest improvements in woven cloths, finishes, resins and preimpregnation techniques.

FIBERGLAS MANUFACTURE

Although all services have been interested in fiberglas for many years, it is noted that no program has been conducted to improve the fiber drawing operation. A number of programs involving the drawing of glass filaments have been undertaken but these have been more concerned with improving fiber properties through changes in glass compositions. In only one program, conducted by Owens-Corning, has a study of the production process been made. (236) This investigation covered such aspects of roving manufacture as bushing hole size, bushing temperature, filament diameter, type of size, forming tube operation, and strand gathering and drying. The objective was to relate filament strengths or deterioration in strengths to any of the manufacturing steps. Owens-Corning concluded that consistent with economic practice:

- The difference between average virgin fiber strength and average strand or roving strength is 20% for both E-HTS and S-HTS fiber (see Table 5-1 for summary of data).
- None of the 20% difference can be attributed to manufacturing.
- Control of glass compositions is adequate.

TABLE 5-1

Strength Data - Virgin Fiber and Roving (236)

Test	Tensile Strength (KPSI)	No. Of Samples	Standard Deviation (KPSI)	Coefficient of Variation %
		E-HTS		
Virgin Fiber	499	667	58	12.0
Forming Strand	404	3690	22	5.4
12-End Roving	398	2650	12	3.0
Cylinder Glass Strength	372	790	12	3.3
		S-HTS		
Virgin Fiber	665	243	79	12.0
Forming Strand	545	152	27	5.0
12-End Roving	549	96	12	2.2
Cylinder Glass Strength	477	73	16	3.3

Despite these conclusions further work to improve and control processing is indicated, if only to reduce variations in virgin fiber strength. Owens-Corning states that fibers are drawn from a batch mix or marbles, but does not show how either affects properties. The evidence that bushing hole-size, temperature and winding speed had no effect was not conclusive. Data is given that yardage weight and the amount of sizing put on the fibers varies, and that there is a tendency for cycling.

Normal production bushings have 204 holes for simultaneous drawing of filaments into a single strand. Present developments are to increase the number of holes to 408, 816 or possibly 1632 in order to speed production. The filament diameters which are applicable to reinforced plastics are shown in Table 5-2. Diameters E and G are more commonly used, but there is a trend to go to the K diameters for some woven cloths. These two developments imposed upon an already existing variation in fiber properties again point out the need for process control.

To obtain optimum and reliable aircraft constructions it would seem more logical to go to smaller diameters, lower filament count, and single end strands in which variations are minimized.

TABLE 5-2

Fiber Diameter Range and Designation

Designation	Fiber Diameter, ins.
D	0.00020 - 0.00025
E	0.00025 - 0.00030
DE	0.00029 - -
F	0.00030 - 0.00035
G	0.00035 - 0.00040
K	0.00050 - 0.00055

GLASS TYPES AND COMPOSITIONS

Of the glass types currently in production, E-glass and S-glass are of interest to the aircraft industry. E-glass, originally developed as an electrical grade, has been the predominant type for structural applications in this country. It appears it will continue in such uses in the foreseeable future. Most woven fabrics are made from E-glass. The first major improvement in E-glass came in 1960 when HTS (an epoxy type) finish was applied to improve strengths. The E-HTS was developed for filament wound motor cases and gave from 5% - 10% increase in burst strengths. It has since been supplanted by the superior S-HTS and is no longer in production.

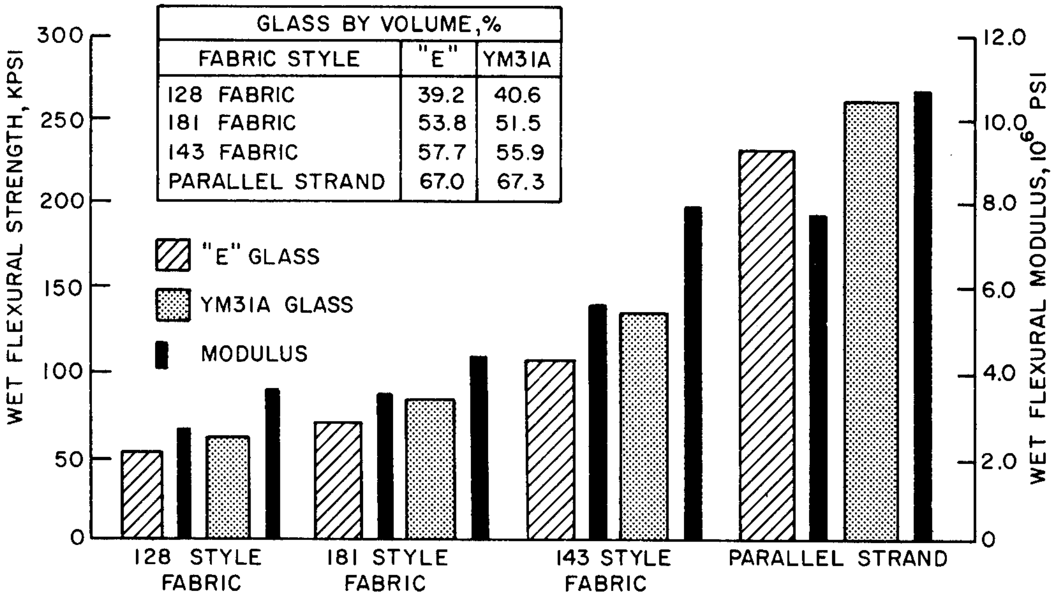
The S or S-994 glass was developed by Owens-Corning under Air Force contract. Except in rare instances, it is marketed with an HTS finish. Again, this fiber has been mostly applied to filament winding. It is currently used on both stages of Polaris A-3, the third stage Minuteman, and both stages of Sprint. In these motor cases it has shown an increase of from 20% - 25% in burst strengths. Compared to E-glass, the S-glass fibers are characterized by higher strengths at room and elevated temperatures, a slightly higher modulus and a slightly lower density. Currently it is being evaluated for aircraft structures as woven or unidirectional cloths.

YM-31A is the most familiar of the high modulus glasses. It also was an Owens-Corning development on Air Force contract. It has not been accepted by industry, either because of its toxic beryllia content or because its composites have not shown exceptionally high strength-to-weight or moduli-to-weight. (See Figure 5-1.) High modulus formulations were developed by Imperial Glass under Navy contract, but these, too, do not seem to have found favor. Without significant increases in moduli-to-weight, it is doubtful whether such glasses can add much in the way of rigidity to an aircraft sandwich construction.

Table 5 - 3 lists the compositions for E, S and high modulus glass.

Table 5 - 4 gives comparative properties of several glass fibers.

Table 5 - 5 compares E and S glass in wound structures.



GLASS BY VOLUME, %		
FABRIC STYLE	"E"	YM31A
128 FABRIC	39.2	40.6
181 FABRIC	53.8	51.5
143 FABRIC	57.7	55.9
PARALLEL STRAND	67.0	67.3

Figure 5-1. Flexural Properties of "E" and YM31A Laminates

TABLE 5 - 3

Approximate Glass Compositions (234)

Component	Percent Component		
	E	S	M(31A)
SiO_2	54.5	64.32	53.7
Al_2O_3	14.5	24.8	-
CaO	17.0	-	12.9
MgO	4.5	10.27	9.0
BeO	-	-	8.0
B_2O_3	8.5	-	-
ZrO_2	-	-	2.0
Na_2O	0.5	0.27	-
TiO_2	-	-	8.0
Fe_2O_3	0.4	0.21	0.5
Li_2O_2	-	-	3.0
Ce_2O_2	-	-	3.0

TABLE 5 - 4

Comparative Properties of Glass Filaments (240)

Parameters	Type Glass			
	E	S(994)	M(31A)	D(556)
Virgin Strength, KPSI				
75° F	500	665	500	350
600° F	425	600	-	-
Density, lbs /cu in	0.092	0.090	0.104	0.070
Modulus Elasticity psi x 10^6	10.5	12.4	15.9	7.5
Dielectric Constant, 10^{10} cycles	6.1-6.4	5.5	-	4.0-4.1
Loss Factor, 10^{10} cycles	0.0055	0.0015	-	0.0010
Index Refraction	1.547	1.523	1.635	1.47

TABLE 5 -5

Comparison of E and S Glass in Wound Structures

Test Method	Tensile Strength (KPSI)					
	Nominal Composite Thickness (in.)	E-HTS		S-HTS		% Difference
		Average x 10^3	Std Dev x 10^3	Average x 10^3	Std Dev x 10^3	
Strand	---	427	10	540	25	26
NOL Ring	0.060	333	9	443	26	33
Elliptical Ring	0.020	422	40	527	17	25
Elliptical Ring	0.010	439	41	540	34	23
4-in. Chamber Hoop	0.008	373	6	492	15	32
4-in. Chamber Hoop	0.0015	355	8	476	6	34
4-in. Chamber Hoop	.023	348	8	457	6	31
4-in. Chamber Hoop	0.030	309	3	389	9	26

NEW FILAMENT DEVELOPMENTS

Current or recent programs to develop improved reinforcements which may be useful in future aircraft structures are briefly summarized.

- Owens-Corning Program. The Air Force is continuing development on contract AF 33(615)-1370, (1964). Target objectives are 1 x 10^6 psi tensile, 18 x 10^6 psi modulus and maximum density of 2.50.
- Hollow Glass Fibers. These have been produced at General Electric and Pittsburgh Plate Glass under BuWeps contract. PPG has also put the fibers into production. Fibers are formed from an E-glass formulation. Their main advantage lies in having greater resistance to bending and buckling at lower weight.

  Table 5 - 6, compiled by GE, compares hollow fibers with other structural materials.

TABLE 5 - 6

Structural Efficiencies for Various Materials

Material	D Density	S_{tu} Ultimate Tensile	S_{tu}/D	S_{cu} Ultimate Compressive	S_{cu}/D	E Modulus	$E^{1/2}d$
	lb/in^3	ksi	$x\ 10^6$ in	ksi	$x\ 10^6$ in	10^6 psi	$x\ 10^3$
Steel	.296	280	0.95	280	0.95	30	18.5
Molybdenum	.368	200	0.54	200	0.54	50	19.2
Titanium	.163	230	1.41	230	1.41	16	24.5
Aluminum	.100	90	0.90	90	0.90	10	31.6
Solid FRP	.082	180	2.19	120	1.46	8.5	35.5
Hollow FRP	.035	67	1.88	93	2.63	3.1	49.8
LA 141	.049	22	0.44	22	0.44	6.5	52.3
Beryllium	.067	78	1.16	78	1.16	42	97.0

- Silica Fibers. These fibers have been investigated at Whittaker - Narmco, also on BuWeps contract. Potentially, they have high tensile strengths and resistance to high temperatures. In work done at Rolls Royce in England, tensile strengths of 1 x 10^6 psi have been reported. Narmco has reported strengths of 500,000 psi but with high variability. Using a core-sheath technique, they have obtained a composite fiber having a tensile strength of 200,000 psi and a modulus of 20 x 10^6.
- Large Diameter Fibers. These are a Whittaker-Narmco development with BuShips funding. Optimum fiber diameter for compressive loading has been found to be 5 mils. These fibers can be resin-coated at the bushing and wound directly as a composite.

- Aerojet 4H-1 Fiber. Development of this formulation is a joint company-Air Force funded effort. The fiber is reported to have a virgin strength of 730,000 psi, a modulus of 14.0 x 10^6 psi and a specific gravity of 2.55. Plans are under way for pilot runs with a 204 hole bushing.
- Boron Fibers: Development of this fiber has received wide attention as one of the most promising future reinforcing materials. Original work started at Texaco Experimental a number of years ago and is still continuing on Air Force contract. Actually it is a composite fiber since the boron is deposited on a substrate, usually tungsten. Present programs at Texaco and others are to improve processing methods, develop finishes, and to evaluate the fibers in composites. Estimates as to when it will be available in quantity have varied, some being as high as 5 - 10 years at a high cost. The Air Force believes that its efforts will produce a reasonably priced reinforcement in quantity within a shorter period. Reported tensile strengths have been as high as 500,000 psi with a modulus of close to 60 x 10^6 psi. (See Table 5 - 7 for properties of boron and other fibers.) The problem in boron composites is to reconcile a high modulus fiber with binders in which the modulus cannot be expected to reach over 1 x 10^6 psi. Suggested first uses in aircraft structures are as localized reinforcements where high rigidity is required.

TABLE 5 - 7

Comparison of Fiber Physical Properties*

Property	Specific Gravity	Young's Modulus of Elasticity (10^6)	Specific Modulus (Sp. gr. x10^6)	Density lbs/cu in	Tensile Strength (Single virgin fiber) (KPSI)
E Glass	2.55	10.5	4.10	0.092	500
S-994 Glass	2.485	12.4	5.00	0.090	700
Quartz	2.20	10.6	4.82	0.079	517
Boron	2.3	58.0	25.20	0.083	400
Beryllium	1.62	44.0	27.20	0.065	190
High Carbon Steel	7.8	30.0	3.85	0.282	600
Titanium	4.7	15.0	3.20	0.170	270
Tungsten	19.2	50.0	2.61	0.695	700

* Tanis, C., 20th SPI, Chicago, February 1965

FIBERGLAS FABRICS

Woven fiberglas cloths have been widely used in the aircraft industry since the earliest applications and it appears they will continue to be used in the future. Also available are the non-woven glass materials, supplied as unidirectional, cross-plied or biased prepregs. These materials, as might be expected, show higher strengths than the woven cloths. Fabricators, however, have been reluctant to use them. The reasons given are that the woven cloths are easier to handle and fibers do not move during cure. In optimized thin aircraft laminates, it seems that the non-woven reinforcements will be superior not only in strength but in thickness and weight control as well. In structures such as helicopter rotor blades, non-woven prepregs have been successfully molded. For optimum structures, at least, molding techniques are required to accommodate these non-woven materials. It is granted that in many non-critical applications the woven fabrics can serve a useful purpose.

Glass fabric reinforcements are selected on the basis of strength, weight, thickness and weave pattern. Table 5 - 8 lists the fabrics used in the aerospace industry along with their properties in both warp and fill direction. Figure 5 - 2 illustrates the more common weaves for structural cloths.

The plain weaves, such as 112, are the firmest and most stable of the woven cloths. They are available in light to heavy weight fabrics.

The satin weaves require more threads per inch to retain stability and are supplied in medium and heavy weights. This weave allows the fabric to be easily draped and accounts for the popularity of 181-8H satin weave, the most commonly used cloth. Crowfoot satins, such as 143 cloth, have less drape and find use in many applications.

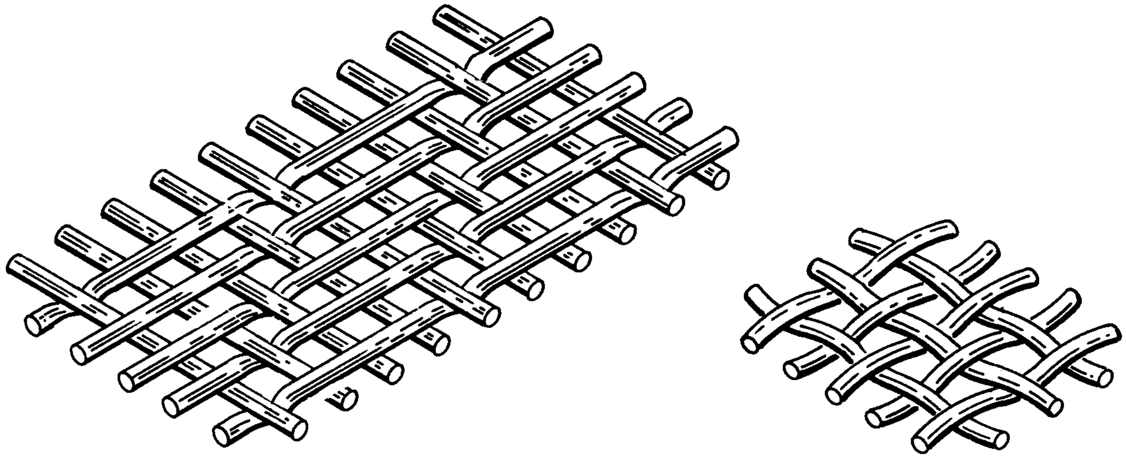
The high modulus weaves (patented by J. P. Stevens) consist of structural and binder yarns. The structural yarns are not interlaced or crimped and begin to approach the non-woven cloths in properties. The material still retains good drape characteristics with minimum fiber distortion. These cloths can be woven with unidirectional reinforcements only or with varying amounts of cross-directional fillers. In a square weave fabric approximately 93% by weight is made up by structural yarns.

The high modulus type weave appears to be the most significant advance made in the woven cloths. S-glass in this type of weave can be expected to give further improvement. A few improvements have been noted in the other types of woven glass. In the newer cloths shown in Table 5-8, single heavier yarns are used rather than two, three or four plies of lighter yarns. The 75-1/0 single yarns, although twisted, are not subjected to a subsequent plying operation. The single twist may allow easier resin penetration than the plied yarns. Table 5-9 compares the properties of the newer styles 7581, 918 and 909 (J. P. Stevens) with style 181 for two different finishes.

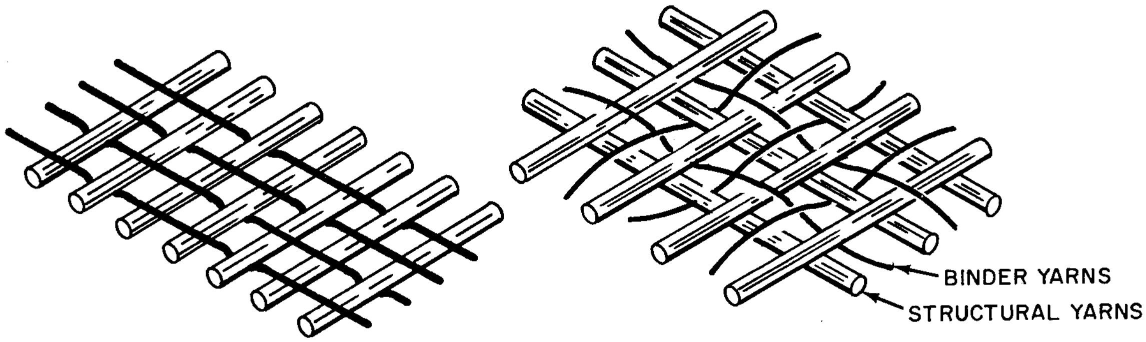
Hess Goldsmith has marketed a style 481 which is similar to 7581, and a style 442, unidirectional cloth, both with 75-1/0 yarns. Table 5-10 compares properties of style 481 cloth with style 181. In this case, the binder is an epoxy resin.

Glass textile terminology indicates the denier, number of plies and number of yarns in a ply but does not show the filament diameter or number of filaments in the yarn. The usual practice in selecting filament diameters for yarns is shown below (234). However, newer weaves with 75's have been using DE filaments.

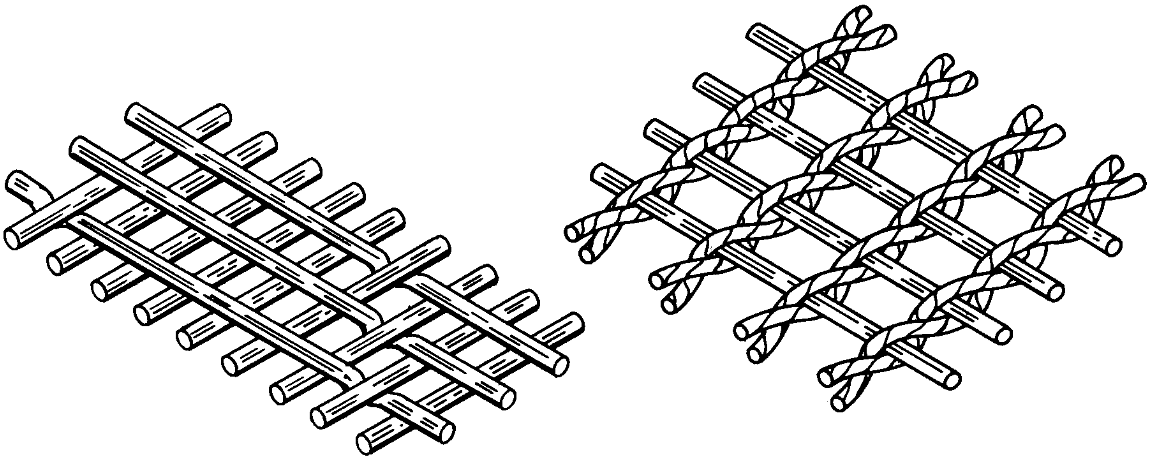
Yarn	Diameter
450's	D
225's	E, DE
150's	G
75's	K(DE)



TWILL PLAIN



UNIDIRECTIONAL HIGH MODULUS



SATIN LENO

Figure 5-2. Basic Types of Weaves

TABLE 5 - 8

Fabrics for Aircraft Structures (239, 275, 276, 277)

Style	Oz/ Sq Yd	Thickness Mils	Thread Count		Tensile Strength		Warp Yarn[1]	Weave
			Warp	Fill	Warp	Fill		
112	2.1	3	40	39	82	80	450-1/2	Plain
116	3.2	4	60	58	125	120	450-1/2	Plain
120	3.2	4	60	58	155	120	450-1/2	Cr. Satin
123	6.0	7	42	32	250	200	225-1/3	Plain
143	8.8	9	49	30	611	56	225-3/2	Cr. Satin
162	12.2	15	28	16	450	350	225-2/5	Plain
164	12.7	16	20	18	500	450	225-4/3	Plain
181	8.9	8.5	57	54	340	330	225-1/3	8-H Satin
183	16.8	18	54	48	650	620	225-3/2	8-H Satin
184	25.9	27	42	36	950	800	225-4/3	8-H Satin
909	9.7	10	--	--	380	380	--	High Mod.
1581	9.0	6.5	56	54	350	325	150-1/2	8-H Satin
1582	13.5	13.4	60	56	440	400	150-1/3	8-H Satin
1584	24.5	24.5	44	35	950	800	150-4/2	8-H Satin
Newer weaves								
151[2]	7.3	9	48	44	--	--	150-1/2	5-H Satin
442	10.2	11	120	20	990	93	75-1/0[3]	8-H Satin
481	8.8	9	57	54	590	356	75-1/0[3]	8-H Satin
7581	9.0	8.5	56	54	350	325	75-1/0[3]	8-H Satin

[1] Fill same as warp except for 143 - 450-1/2

[2] S-glass, all others E-glass

[3] DE - Dia. filaments

TABLE 5 - 9

Laminate Properties - Styles 7581, 918, 909 vs. Style 181 with Polyester Resin (273, 274, 276)

Finish:	--	--	Volan		S-550	
Style:	**181**	**7581**	**918**	**909**	**181**	**7581**
Flexure, Dry, KPSI	77.1	87.6	78.2	79.3	88.1	94.4
Wet	58.9	69.0	63.0	60.0	78.7	83.8
Flex. Mod. Dry, psi x 10^6	3.1	3.0	3.1	3.1	3.2	3.1
Wet	2.9	2.8	2.8	2.9	3.0	2.9
Compression, Dry, KPSI	51.4	60.6	48.2	50.5	62.9	62.1
Wet	36.7	47.3	40.3	41.2	58.2	52.7
Tensile Dry, KPSI	48.6	57.0	42.0	41.0	48.7	59.4
Wet	47.4	55.1	40.0	39.0	48.1	54.9
% Resin	38.0	36.5	36.0	38.0	36.1	37.0

TABLE 5-10

Laminate Properties - Style 481 vs. Style 181 with UCC-ERL-2256/MDA Epoxy*

Style:	481 - I 550	181 - I 550
Flexure, R. T., KPSI	97	87
2 hr. boil - KPSI	90	74
160° F	87	71
300° F	57	52
Flex. Mod. R. T., psi x 10^6	3.2	3.2
2 hr. boil	3.3	3.2
160° F	3.5	3.4
300° F	3.1	3.0
Tensile, R. T., KPSI	68	52
2 hr. boil	66	48
Compression, R. T., KPSI	62	59
2 hr. boil	60	58
% Resin	34.1	34.0

* Hess Goldsmith Data

In the older weaves, the efficiency of smaller diameter fibers is not apparent, most likely because of filament damage in the twisting operations. With the newer weaves, strength improvements can be made by use of smaller diameter filaments as shown in Table 5 - 11.

TABLE 5 - 11

Effect of Filament Diameter on Strength of Fabric and Laminate (156)

Construction:	42 x 32	42 x 32	42 x 32
Yarn:	75-1/0 ECK	75-1/0 ECG	75-1/0 ECDE
Filament Diameter, ins.	0.00051	0.00037	0.00025
Filaments/yarn	204	408	816
Greige - Tensile	284/228	315/260	380/317
Volan - Tensile	100/75	105/91	180/145
Thickness	0.007	0.008	0.007
Wt/sq. yd	5.9	5.9	5.8
Dry Flexural, KPSI	70	77	80
Wet Flexural, KPSI	58	64	67
Dry Compression, KPSI	51	58	61
Wet Compression, KPSI	40	45	53
% Resin	28.5	29.0	28.6
Resin - Epon 1001		Finish - Volan A	

COUPLING AGENTS FOR FIBERGLAS FABRICS OR ROVING

As the glass filaments leave the bushing, they are coated with a "sizing" to protect them from abrasion and to allow for subsequent handling and weaving. With the laminating cloths, it is necessary to remove the sizing after a weaving since it inhibits glass resin bonds. The size is replaced by a coupling agent or "finish" to develop wet strength retention in the laminate. Generally, silane or chrome complexes are used for this purpose, some finishes being more suitable to particular resins. The more common finishes are summarized in Table 5 - 12. Glass finishes used with epoxy, polyester and phenolic resins are listed in Table 5 - 13.

TABLE 5 - 12

Finishes for Fiberglas Fabrics or Rovings

Finish	Description	Compatible Resin
111	Heat clean, part of size burned off	Melamine
516	Improved 111	Melamine
112	Heat clean, almost all size burned off	Silicone
112-neutral p^H	112 plus wash to neutral p^H	Silicone
114 (Volan)	112 plus saturation with methacrylate chromic chloride	Polyester
Volan A	Basically 114 hydrolyzed to remove chlorine and provide free hydroxyl Superior wet strength than 114	Polyester, epoxy phenolic
I-550	Modified Volan, soft finish	Polyester, epoxy
A-1100	112 plus saturation with an amine-vinyl silane	Epoxy, phenolic melamine
Garan, 136, 301	112 plus saturation with silane solution to leave a vinyl-silane	Polyesters, silicones
172	Similar to Garan. Different solution for saturation	Polyester
A-174, Z-6030	112 plus saturation with modified methoxy silane	Polyester
Z-6040, Y-4086 Y-4087	112 plus saturation with modified methoxy silane	Epoxy
NOL-24	112 plus saturation with solvent solution of halosilane	Polyester, epoxy phenolic
801	Applied at bushing. E-glass rovings	Epoxy
HTS (901)	Applied at bushing. Epoxy resin plus coupling agent	Epoxy
L HTS	Modified HTS for E-glass	Epoxy

TABLE 5 - 13

Glass Finishes for Use with Various Resins

Finish		Resin System	
	Epoxy	Polyester	Phenolic
Chromic			
114 (Volan)	x	x	x
Volan A	x	x	x
I-550 & S-550	x	x	x
Silane			
A-1100 (Y2967)	x		x
Garan		x	
172		x	
A174-Z6030		x	
Z-6040, Y4086, Y4087	x		
NOL 24	x	x	x

TABLE 5 - 14

Improvement in Polyester Laminates (24)

Year:	1950	1955	1963
Cloth:	181-114	181-136	181*
Flexural, dry, KPSI	60	60-62	87-103
75 days at 75° F, K psi	44	52-58	-
2-hr. boil, KPSI	-	52-56	81-100
Tensile, dry, KPSI	-	46-48	59-69
30 days at 75° F, KPSI	-	44-46	-
2-hr. boil, KPSI	-	-	59-64
Compression, dry, KPSI	-	41-44	50-60
30 days at 75° F, KPSI	-	30-42	-
2-hr. boil, KPSI	-	-	53-60

*Methacrylate silane finish

Although the mechanism of coupling agent action is not known and is subject to controversy, the results obtained indicate general improvement in other physical properties besides wet strength retention. Strength increases may be due to better wet-out of the cloths by the resin. A general trend is shown in Table 5 - 14.

Recently developed are the finishes I-550, S-550, Z6030, Z6040, and J.P. Stevens' 910 for polyesters and 920 for epoxies. Comparative data is shown in Tables 5 - 15, 5 - 16, 5 - 17 and Figure 5 - 3.

The theoretical aspects of the glass-resin bond or interface have been the subject of intensive research. It is believed by many that this phase of the laminate structure can be governing and offers the greatest potential for property improvements. The interface is not discussed here, not because of its lack of importance, but simply because it is adequately covered in present researches. Specific programs on application of interface theory to aircraft materials is not anticipated.

TABLE 5 - 15

Comparison of Finishes/Paraplex P-43/181 Cloth

Finish:	(I-545)[1] (Z-6030)	Volan	Garan	A-172	MIL-P 8013-C
Dry Flexural	82.7	72.0	60.3	68.7	50.0
Wet Flexural	81.9	55.2	57.7	61.9	45.0
Dry Compression	59.9	51.6	51.2	51.9	35.0
Wet Compression	60.9	36.5	46.2	48.7	30.0
Dry Tensile	47.2	61.3	47.0	50.0	40.0
Wet Tensile	48.5	59.6	49.3	58.2	38.0
No. of Plies	12	12	12	12	12
% Resin	37.4	38 ± 2	38 ± 2	38 ± 2	38 ± 2
No. of Laminates	8	50	25	15	-
Rate of Resin Wet-Out:					
Initial Wet-Out	5 secs	4.5 secs	5 secs	5 secs	-
Final Wet-Out	10 secs	16 secs	-	14 secs	-
Drape Stiffness:					
Warp	2.6	1.2	1.9	1.7	-
Fill	2.1	1.1	2.0	1.6	-

[1] I-545 J.P. Stevens, Z-6030 Dow Corning
Hess Goldsmith Data

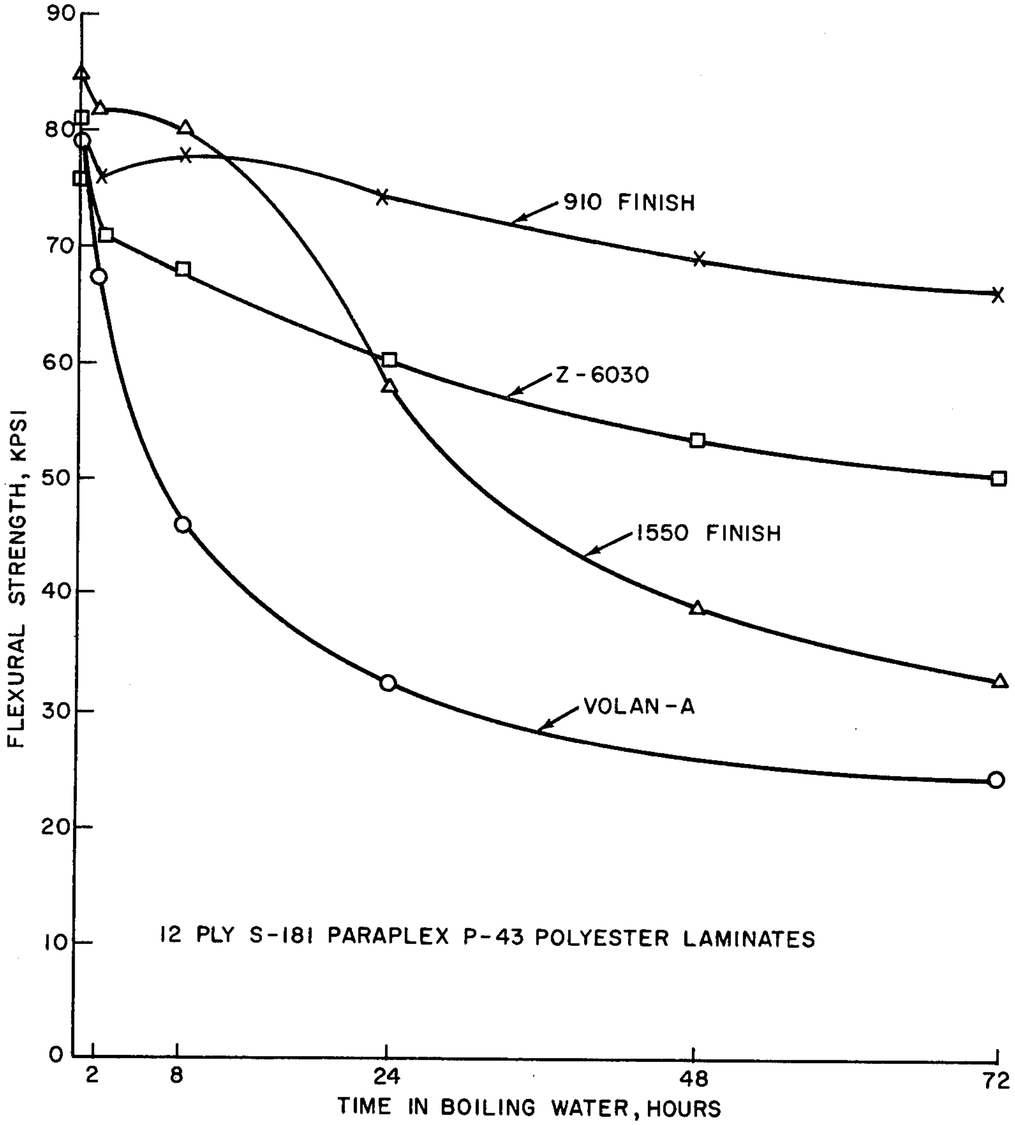


Figure 5-3. Influence of Finish on Long Term Wet Strength

TABLE 5 - 16

Comparison of Finishes/Epon 828/181 Cloth (280)

Finish:	920	Volan A	A-1100 Soft
Flexural Strength KPSI			
Dry	83.0	75.2	67.9
2-Hour Boil	80.9	72.8	61.9
8-Hour Boil	73.5	58.1	58.7
24-Hour Boil	73.9	44.7	52.4
48-Hour Boil	71.9	38.3	49.2
72-Hour Boil	72.8	38.7	44.5
Flexural Modulus psi x 10^6			
Dry	3.23	3.01	2.48
2-Hour Boil	3.24	3.06	2.48
8-Hour Boil	3.00	2.82	2.48
24-Hour Boil	2.96	2.80	2.41
48-Hour Boil	2.92	2.81	2.40
72-Hour Boil	3.12	2.92	2.24
Compressive Strength KPSI			
Dry	63.8	56.6	59.5
2-Hour Boil	62.5	51.8	50.4
Tensile Strength KPSI			
Dry	53.0	50.1	51.7
2-Hour Boil	50.9	44.7	48.7
% Resin Content	39.8	39.1	41.7
Number of Plies	12	12	12

TABLE 5 - 17

Effect of Finish on Adhesion of Epoxy to Glass (156)

	Finish:	Volan A	I-550	A-1100	Z-6040	Y-4086
Flexural, Dry KPSI		84	102	84	105	115
Wet	2-hr.	71	94	71	91	108
	24-hr.	46	92	65	85	98
	48-hr.	42	73	61	79	85
	72-hr.	39	65	60	74	75
% Retention	2-hr.	84.0	92.0	85.0	86.2	94.9
	72-hr.	46.1	63.7	71.6	70.5	65.2
Compression, Dry KPSI		59	65	69	73	65
Wet	2-hr.	45	61	63	56	58
	24-hr.	31	46	43	43	43
	48-hr.	23	36	33	38	42
	72-hr.	21	32	33	34	39
% Retention	2-hr.	76.5	94.0	91.5	77.3	89.0
	72-hr.	35.4	50.0	47.2	46.6	59.8
% Resin		30	29.1	29.2	27.5	28.0
Samples		25	10	25	2	2
Bond Str., lbs.						
Warp		3783	3716	3483	3383	3316
Fill		3350	3216	3050	3116	3050

12 layers - 181 with Epon 1001

FABRICS FROM S-HTS GLASS YARNS

S-glass cloth has been made available in three styles - 181, 143 and a new 151. Only limited data has been reported and this for the style 181 cloth. The style 151 is presently undergoing evaluation at a number of aircraft companies. Comparative data for the S-glass version of 181 is shown in Table 5 - 18. While the data reveals significant increases in tensile and compressive strengths for the S-glass, the survey has not uncovered a single use of the material in aircraft structures. In a few more critical designs it is receiving consideration. If the material exhibits superior fatigue properties as has been claimed, it can be expected to find use in rotors and propeller blades. The general reaction to S-glass cloth is that its higher price is hard to justify, particularly in parts which are not highly stressed. In future optimized designs it would seem more efficient to make use of S-glass fibers as unidirectional, biased or cross-plied unwoven cloths or in the high modulus type weaves.

The S-glass cloth has been furnished with an HTS finish in almost all cases. Some of the problems associated with the weaving and subsequent molding of the cloth have been due to this finish. The HTS finish, whether on roving or cloth, has exhibited some deterioration in handling properties and strength characteristics which vary with the storage history. Figure 5 - 4 illustrates this effect as a fall-off in shear strength with storage time. In the Polaris A3 program, low temperature storage was made mandatory, thus resolving the problem. Another deficiencv noted in the HTS finish has been a decreased strength at temperatures in excess of 400° F.

The fact that S-HTS yarns have been woven without heat cleaning or solution application of other finishes is of significance. This approach to weaving has a decided advantage in that damage or deterioration in properties incurred in a heat clean operation is eliminated.

This method, improved to remove aging problems, should be applied to the weaving of other type cloths as well.

TABLE 5 - 18

Comparison of Epoxy Laminates Made from S-Glass and E-Glass Style 181 Fabric

Source	Owens-Corning			Grumman		Naval Air Engineering Center		North Amer
Glass	S-HTS	S-HTS	E-Volan	S-HTS	E-Volan	S-HTS	E-Volan	S-HTS
% Resin - weight	28.5	33.6	28.0	29.8	31.9	--	--	--
Thickness - Mils	120	120	115	125	125	125	125	125
Specific Gravity	1.89	1.89	1.91	--	--	--	--	--
Tensile - KPSI								
0° - dry	97.7	87.2	55.8	97.1	56.5	94.9	68.6	103.8
45° - dry	39.4	36.2	25.2	--	--	--	--	--
0° - wet	94.4	88.0	58.6	90.0	55.0	94.8	57.7	93.4
Tensile Modulus - psi x 10^6								
0° - dry	3.15	3.37	3.16	3.98	3.41	--	--	--
45° - dry	1.56	1.67	1.80	--	--	--	--	--
0° - wet	3.15	3.48	3.31	4.10	3.45	--	--	--
Compression - KPSI								
0° - dry	67.4	62.1	54.2	--	--	59.1	35.8	66.4
45° - dry	31.6	28.8	35.4	--	--	--	--	--
0° - wet	64.6	61.1	46.8	--	--	54.4	37.1	65.1
Compression Modulus - psi x 10^6								
0° - dry	4.60	4.01	4.22	--	--	--	--	--
45° - dry	2.26	2.12	2.64	--	--	--	--	--
0° - wet	4.17	3.88	4.08	--	--	--	--	--
Interlaminar Shear - psi								
0° - dry	3040	3950	3330	--	--	3700	2053	3728
45° - dry	2590	3410	3160	--	--	--	--	--
0° - wet	3200	4260	2380	--	--	3251	1780	3865
Edgewise Shear - KPSI								
0° - dry	--	18.9	14.3	--	--	--	--	--
45° - dry	--	24.9	17.7	--	--	--	--	--

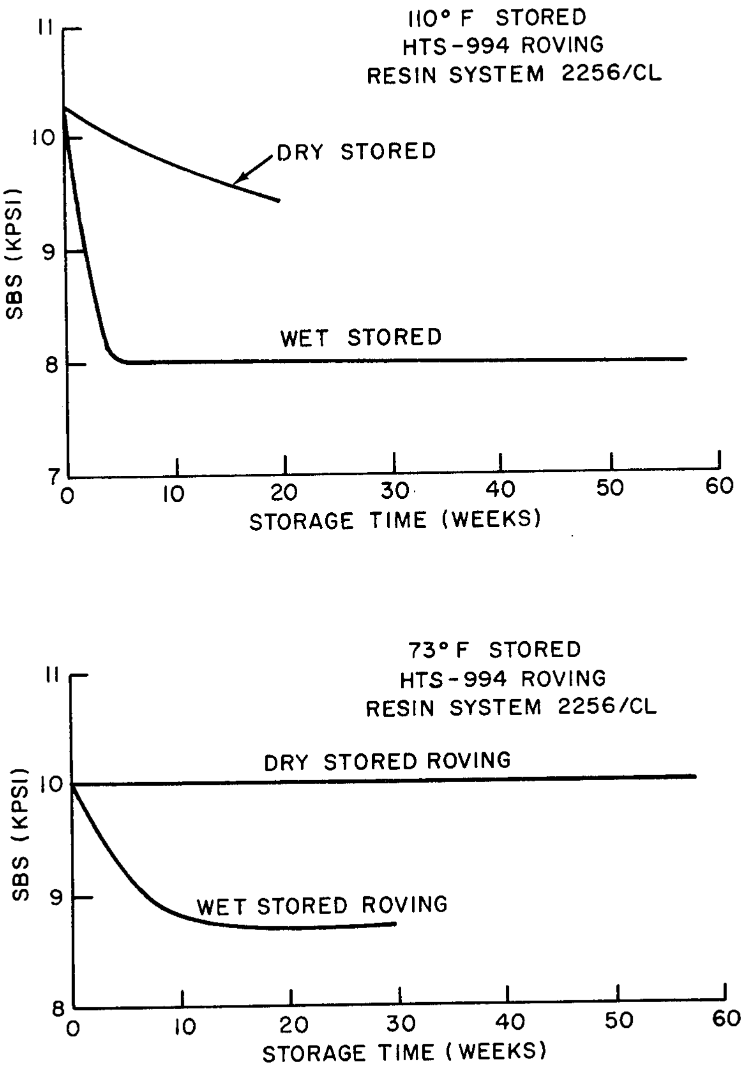


Figure 5-4. NOL Beam Shear Strength

RESIN SYSTEMS FOR FIBERGLAS COMPOSITES

There has been a tendency among aircraft fabricators to minimize the effects of the resin system on the properties of a composite. As a result resins have been selected simply on the basis of their ease of handling and relative costs. With the advent of refined filament winding techniques, the development of orthotropic analytical methods and studies of failure mechanisms, the function of the resin has assumed greater importance. Certain types of failures as an interlaminar shear or cyclic loading are believed to originate in the resin. In cases where resin crazing occurs, the useful life of a structure may be ended, even though it still maintains integrity. Crazing is known to allow moisture penetration with subsequent "stress corrosion" of the fibers or in deterioration of facing to sandwich core bonds. The resin is also the limiting factor at elevated temperatures and in time dependent properties such as creep, aging and weathering. The characteristics to be considered in selecting a resin, then, will include -

- Composite Properties. Tensile, compressive and shear strengths; modulus and elongation; heat distortion; aging, weathering and resistance to moisture penetration.
- Handling Properties. Wet-out of fibers; variation of viscosity with time and temperature; pot-life of catalyzed resin.
- Curing Properties. Time and temperature for cure and post cure; resin flow during cure; resin shrinkage during cure.

Polyesters and epoxies are the most widely used resins in the aircraft applications. The phenolics, phenyl-silanes and silicones find more limited use in applications where heat resistance or electrical properties at higher temperatures are of importance. The choice between a polyester or an epoxy has sometimes been an arbitrary one. It is believed, however, that in optimum structures epoxies offer more advantages. Polyesters, which are cheaper than the epoxies, will still be used where strengths are less critical or in electrical and radome applications. The polyesters are discussed briefly and the epoxies are covered in more detail in this section. Polyesters and epoxies are compared in Table 5 - 19.

Developments in newer polymers are directed to higher temperature applications for both composites and adhesives. Of significance are the polybenzimidazoles (PBI) being studied on Air Force contract, Narmco variations of PBI known commercially as 'Imidite' and the du Pont polyimides. The PBI resins are already being evaluated in sandwich panels to be used at supersonic speeds.

TABLE 5 - 19

Comparison of Polyester and Epoxy Resins

Resin	Polyester	Epoxy
Cost/lb. 181 prepreg	2.10	2.60
Cast Resin Properties, KPSI		
Compressive	22.5	22.0
Tensile	5.5	14.4
Shear, Interlaminar	5.7	9.1
Shear, Johnson	7.4	10.0
Composite Cylinder, KPSI		
Burst, 0.30 in. wall	62.0	89.0
Burst, 0.50 in. wall	62.0	79.8
Laminates, Style 181		
Tensile, KPSI	45.5	56.7
Tensile Modulus, psi x 10^6	2.7	3.6
Compressive, KPSI	36.4	45.9
Compressive Modulus, psi x 10^6	3.2	3.3
Fatigue, Stress at 10^7 cycles, KPSI	12.0	15.0

Aerojet and Forest Products Data

Polyesters: The general purpose polyesters, comprising a variety of monomers, retain a balanced combination of properties such as easy handling, short cures, good mechanical and electrical properties and lower cost. Certain monomers are added for specific purposes; for example diallyl phthalate is frequently used to aid in B-staging prepregs and triallyl cyanurates for high temperature strength. A major deficiency has been the high shrinkage which takes place during cure. Some of the newer polyesters are said to have less shrinkage. For a comprehensive treatment of polyesters, see reference 234.

Epoxy Resin Systems: Conventional epichlorhydrin-bisphenol A, cured with MPDA, is the most widely used epoxy system for aircraft laminates. (See Table 5 - 20 for epoxy resin types and curing agents). Development of newer epoxy types has increased markedly in the last few years and a variety of systems are now available, having a range of such properties as flexibility, cure cycles, and heat resistance. Many of these have been evaluated in filament winding and in some instances the older conventional system is being replaced. The epichlorohydrin bisphenol-A formulations are generally limited to about 300° F. Epoxy novolacs have been tested at temperatures of 500° F and still retain some strength as high as 750° F. Some of the epoxidized polyolefins or cyclo-aliphatic derived epoxies offer advantages of easier handling, high heat distortion temperatures, and improved electrical properties and weather resistance.

The aliphatic amines with conventional epoxies or novolacs yield room temperature curing systems, but with low heat distortion temperatures and lower strengths. Modified aliphatic polyamines impart varying degrees of flexibility and improved impact strength. Aromatic amines give laminates with better heat resistance. The anhydrides have advantages over the amines in that heat distortion temperatures are higher, exotherms are lower and they are less toxic. The latent curing agents are used for longer pot lives and moderate curing cycles. Reactive diluents are sometimes added to the resin system to lower viscosity, improve wet-out of fibers and control resin content.

Most of the newer systems have resulted from research carried out by the resin suppliers. A few programs have been government sponsored, of which two are of interest at this point.

Union Carbide is presently conducting a research program on Navy contract. Although eventual application of resin systems being developed is in deep submergence, the approach is of significance in that an attempt is being made to correlate physical properties with molecular configurations. Results to date show a relationship of tensile strength, compressive strength and heat distortion to the distance between active carbon sites in the molecular structure. Figures 5-5 and 5-6 relate compressive strength and modulus to these distances. Similar results apply to tensile strength and modulus. One of the resins being developed has a compressive strength of 50,000 psi and a modulus of $1x10^6$ psi, which represent the highest values reported for any resin system (287).

Aerojet-General has had several contracts with the Navy in which resin studies were involved. The latest completed in August 1964 was part of the Polaris program. The objective was to determine the effect of resin parameters or resin modifications on physical properties of resin systems and finished composites. Regression equations or response surfaces showed the relationship between five resin parameters and eighteen resin physical properties. The relation between resin physical properties and composite properties was obtained from pressure vessel burst tests and short beam shear tests of

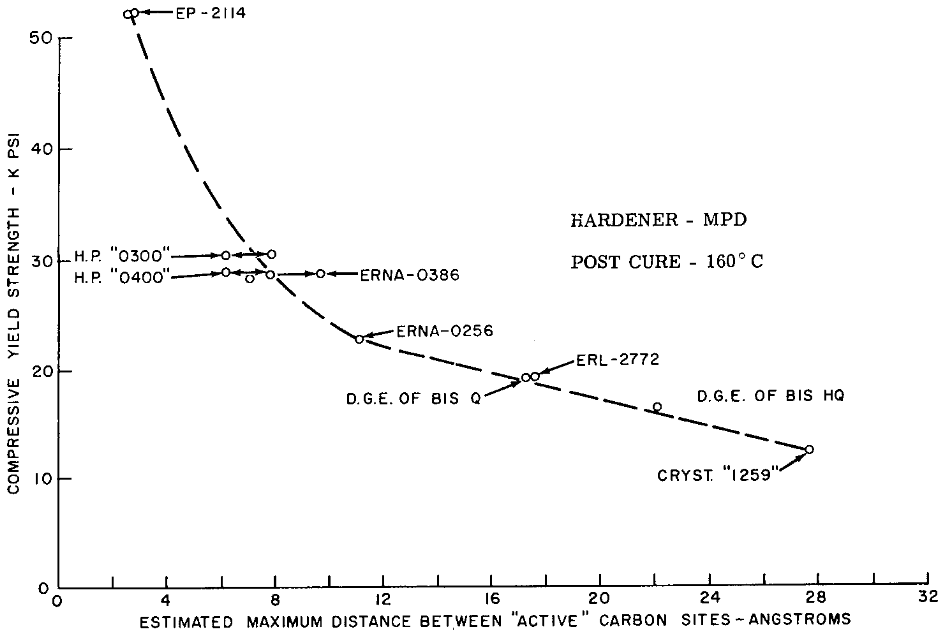


Figure 5-5. Correlation Properties of Epoxy Resin with Molecular Structure; Compressive Strength

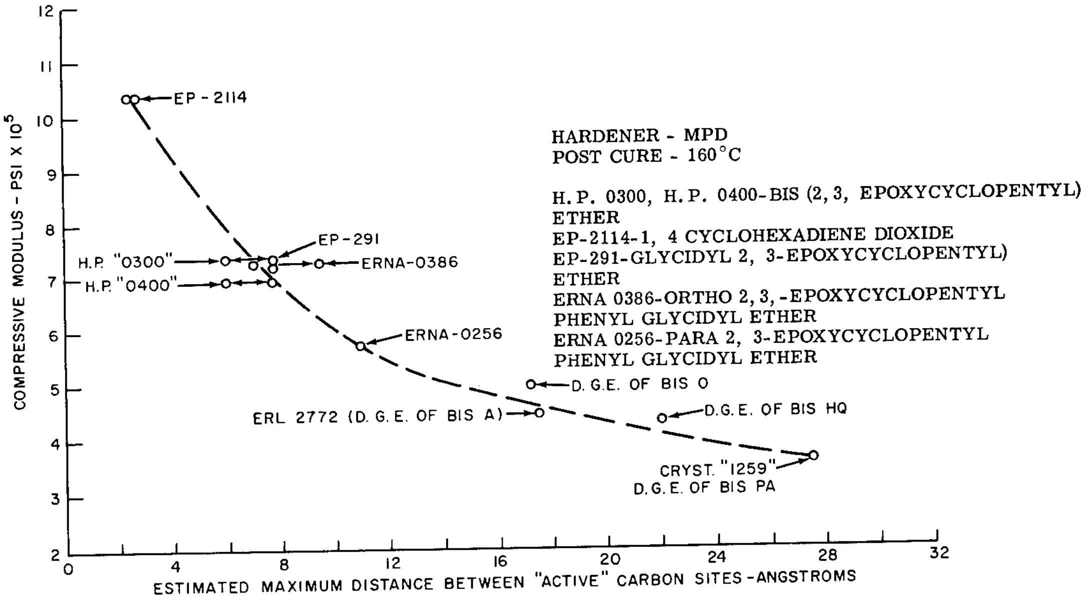


Figure 5-6. Correlation Properties of Epoxy Resin with Molecular Structure; Compressive Modulus

NOL-type rings. Results indicated that the type and amount of hardener are the most important parameters affecting resin physical properties, particularly for amine hardeners. Composite shear strength showed a statistically significant correlation with fourteen of the eighteen resin properties. Burst pressures, however, showed little correlation with resin properties. A decrease in burst strength was noted with increased resin viscosity and was attributed to variation in wetting of fibers and in resin content (3). (See Figure 5-6A.)

PREIMPREGNATED FIBERGLAS MATERIALS

The use of prepregs for fabrication of fiberglas structures has become standard practice within the aircraft industry, and only occasionally are constructions made with the wet lay-up method. The advantages cited for prepregs are their ease of handling and better control of resin content and distribution. It can also be added that from a viewpoint of standardization and control of processing, it would be simpler to apply such measures to prepregs than to wet lay-ups.

In this connection two government programs are of interest. One, at Aerojet-General, had as its objective the upgrading of prepreg S-glass roving for the Polaris. The other was run as a cooperative effort with Owens-Corning, the roving supplier, and U. S. Polymeric, the maker of the prepreg. A second contract at U. S. Polymeric was conducted to improve the storage life of S-glass roving. A resin system, designated as E-717, was developed which allowed extended storage at 40° F as compared to 0° F for older systems. It is concluded that similar investigation of the processing of woven and unwoven prepregs would be helpful in realizing a more uniform product.

Prepreg suppliers have indicated that epoxies are increasing in demand and that at present they are used about as frequently as the polyesters. The epoxy novolac type is coming more into favor, as it is less critical in either matched metal die or autoclave molding. Its flow characteristics prevent resin washout and allow closer control of resin content.

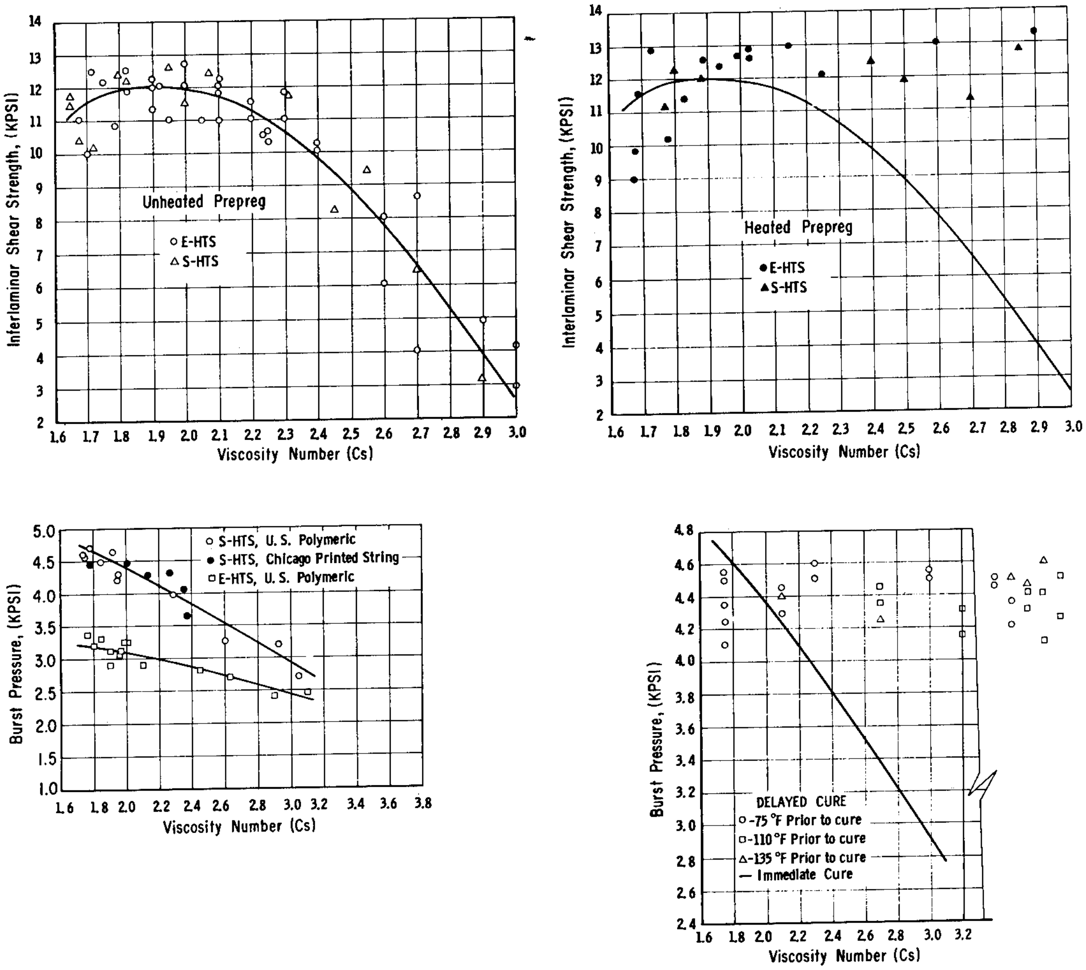


Figure 5-6A. Influence of Processability on Interlaminar Shear and Burst Strength

TABLE 5 - 20

Epoxy Resin Types and Curing Agents

Resin Types

1. Diglycidyl ether of bisphenol A (conventional)
2. Epoxy novolac
3. Epoxidized polyolefin
4. Cyclo-aliphatic
5. Other types
    - Bisphenol A - epoxide, halogenated
    - Bisphenol A - epoxide, high functionality
    - Bisphenol A - epoxide, reactive diluent
    - Triglycidyl derivative of para-amino phenol
    - Resorcinol diglycidyl ether

Curing Agents

1. Aliphatic amines
    - DETA - diethylene triamine
    - TETA - triethylene tetra amine
    - \- modified polyamines
2. Aromatic amines
    - MPDA - metaphenylene diamine
    - MDA - methylene dianiline
    - DDS (DADPS)- diamino diphenyl sulfone
    - eutectic mixtures of MPDA and MDA
3. Latent Curing Agents
    - BF_3MEA - boron trifluoride-monoethylene amine
    - DICY - Dicyandiamide
4. Anhydrides
    - MNA - methyl nadic anhydride
    - HHPA - hexahydrophthalic anhydride
    - TMA - trimellitic anhydride
    - PMDA - pyromellitic dianhydride
    - MA - maleic anhydride
    - HET - chlorendic anhydric
    - DDSA - dodecenyl succinic anhydride

CORE MATERIALS FOR SANDWICH CONSTRUCTIONS

Core materials are available in various forms, both metallic and nonmetallic. They include honeycombs, corrugations, waffles, foams, wood and the glass cloth fluted cores. Those which appear to be applicable to the fiberglas constructions are considered here.

The aluminum honeycomb cores are the most widely used in the aircraft industry, whether the sandwich is all metal or has fiberglas facings. The fiberglas honeycombs have been selected only where their special properties are needed as in radomes or for heat insulation. The fiberglas fluted cores, which are relatively new, have been suggested as possible airframe materials.

The advantages of the aluminum core are inherent in its fabrication process which allows greater automation and closer control of cell size. It can be shipped in an unexpanded (HOBE) form or as pre-expanded core. Costwise, it is cheaper than fiberglas. The fiberglas cores, on the other hand, have not shown reproducible properties. They are limited as to minimum cell size and are shipped as expanded core. Its advantages are better heat resistance than aluminum, good dielectric properties and lower heat transfer coefficients. It can be more readily formed to contoured shapes.

Strengthwise, the potential of fiberglas core does not seem to have been realized. Contributing factors are large variations in resin and reinforcement contents of the cloths making up the core. As a result core densities, cell shapes, compressive strengths and shear properties vary widely. Typical compression variations are shown in Table 5 - 21. Comparative properties of core materials are listed in Figure 5 - 7 and Table 5 - 22. Figure 5 - 8 gives the directional notations for honeycomb cores. A similar notation is used for other core materials.

Besides cell size and ribbon thickness, honeycombs come in various shaped cells as depicted in Figure 5 - 9. The hexagonal cell is most common. The multiwave cell, which was once quite popular, is no longer being fabricated. The staggered hexagon and cruciform cells are most easily formed to relatively severe single curvatures and moderate double curvatures. The chevron, dovetail and arrow cells, though uncommon, were designed to form to single or compound curvatures. Overexpanded hexagon is more frequently used for cylinders or moderate double curvatures.

TABLE 5 - 21

Variation in Fiberglas Honeycomb Core[1]

Nominal	$\overline{X}$	$\overline{X}\pm3\sigma$ at
Core Density	Compressive Strength, psi	99.7% Confidence
4 lbs/cu ft	485	350 - 600
6 lbs/cu ft	965	530 - 1400

[1] Douglas Aircraft - PDL 35605, 12 Dec. 1963

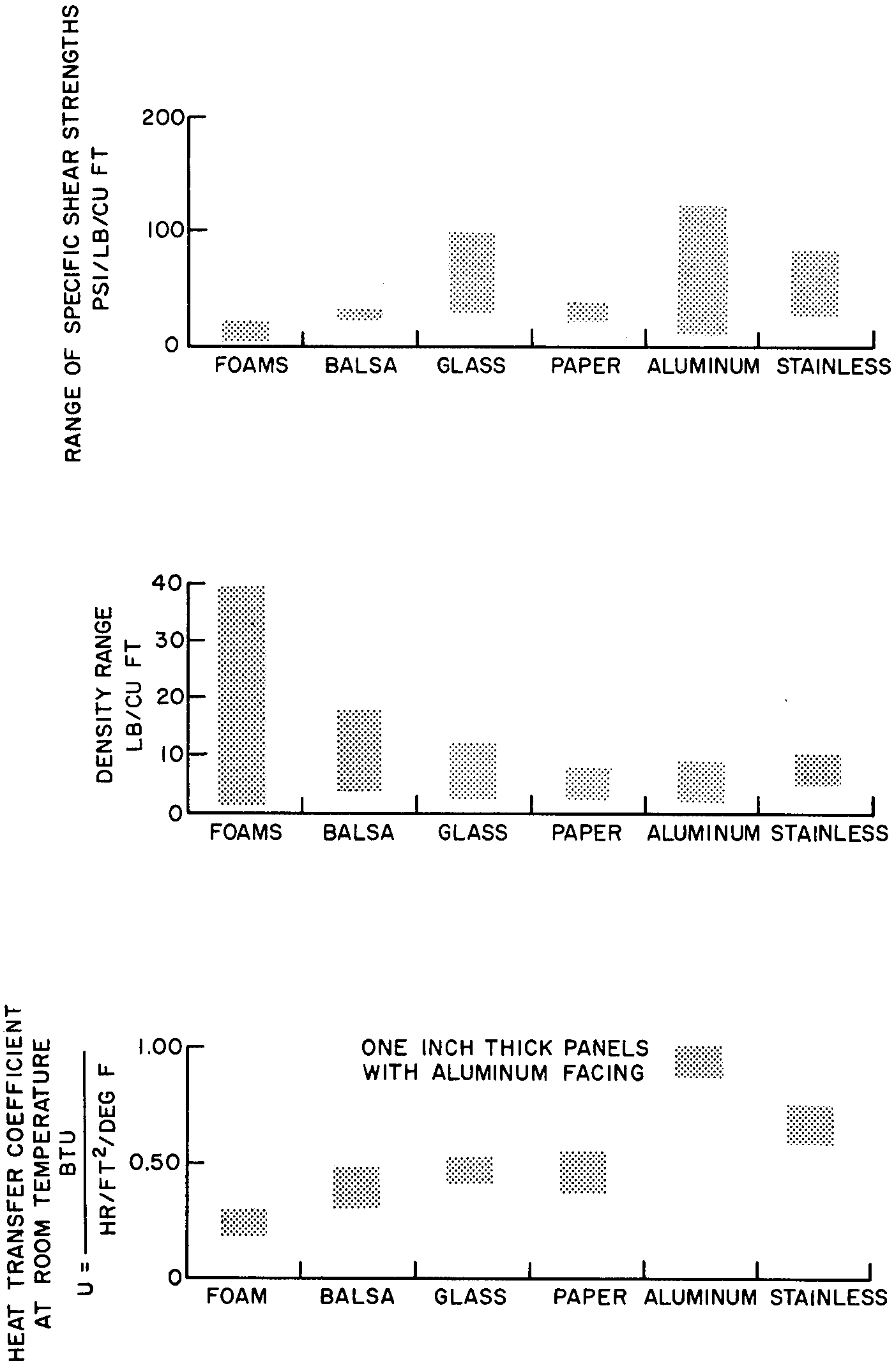


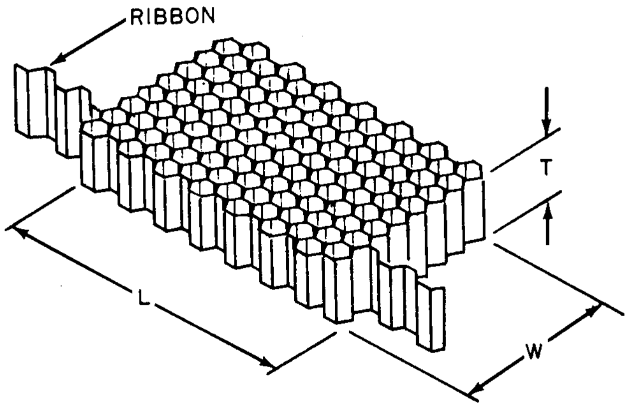
Figure 5-7. Properties of Various Core Materials

TABLE 5 - 22

Compressive and Shear Properties of Various Core Materials[1]

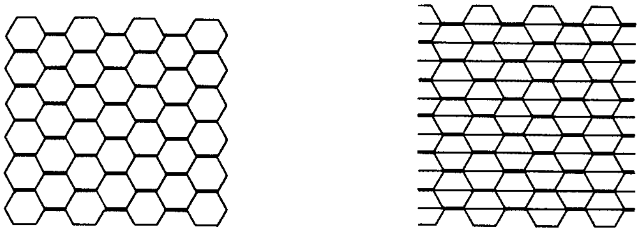
	Density lbs/ cu ft	Compression		Shear			
		Strength, psi	Modulus K psi	Strength, psi		Modulus K psi	
Direction:	--	T	T	TL	TW	TL	TW
Aluminum Foil Honeycomb							
1/8 cell-1 mil foil	4.8	590	--	320	210	59	38
3/16 cell-2 mil foil	5.8	720	--	400	250	87	40
1/4 cell-2 mil foil	4.5	390	191	240	170	42	25
1/4 cell-2 mil foil	4.8	440	--	260	140	48	39
Glass Fabric Honeycomb-112 Cloth							
3/16 cell-Nylon Phenolic	9.2	1510	151	610	280	32	16
3/16 cell-Nylon Phenolic	6.3	790	118	460	240	23	11
3/16 cell-Polyester Phenolic	6.4	650	98	340	140	22	11
1/4 cell-Polyester	8.3	890	100	320	200	18	11
Fluted Core							
Polyester/301	10 approx	370	--	480	200	57	13
S-Glass/181	--	482	--	--	--	-	-
Urethane Foam							
Alkyd	10.3	240	16	180	180	4	4

[1] Source - Proposed revision to ML HDBK 23.



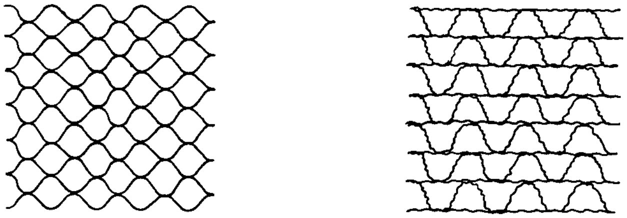
T = THICKNESS, OR DEPTH.
L = RIBBON OR LONGITUDINAL DIRECTION
W = TRANSVERSE DIRECTION PERPENDICULAR
TO RIBBON

Figure 5-8. Honeycomb Core



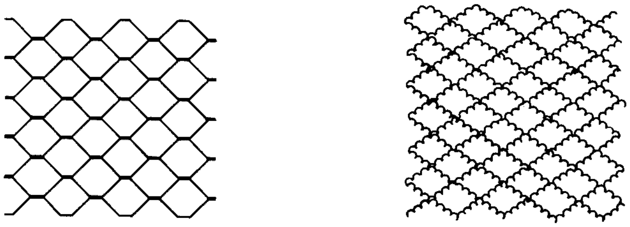
HEXAGON

REINFORCED HEXAGON



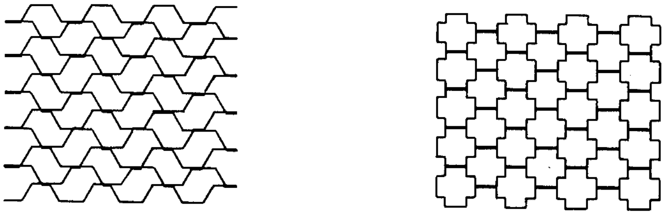
SINE WAVE

MULTIWAVE



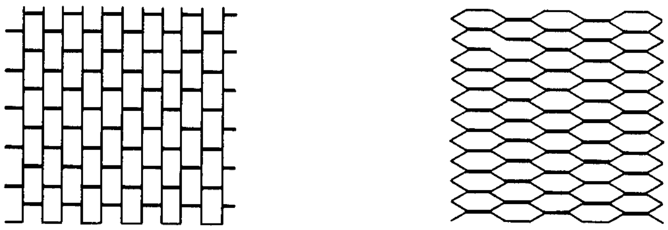
SQUARE

CORRUGATED SQUARE



STAGGERED HEXAGON

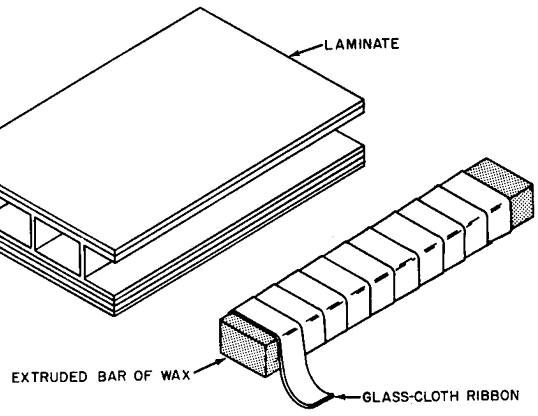
CRUCIFORM



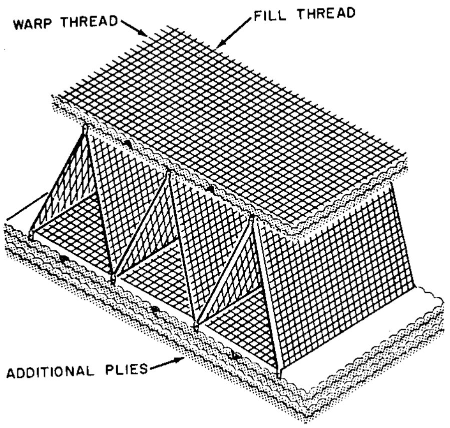
RECTANGULAR OR OVER-EXPANDED HEXAGON

UNDEREXPANDED HEXAGON

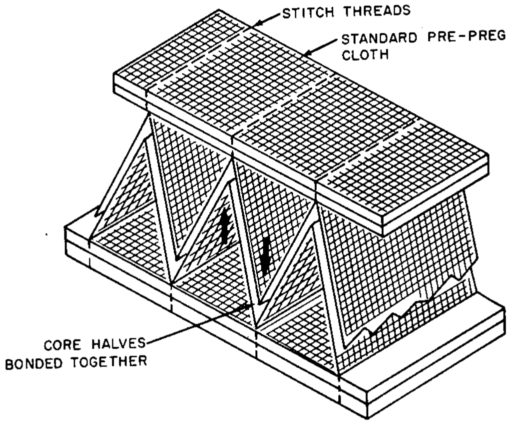
Figure 5-9. Honeycomb Cell Configurations



A - Lost Wax



B - Woven



C - Sewn

Figure 5-10. Types of Fluted Cores

In recent years considerable interest has been given to fluted core materials for structural sandwiches. The fluted cores have been made by using the lost wax process, direct weaving of cores and by sewing prepreg fiberglas fabrics into the required shapes. The three techniques are illustrated in Figure 5-10. The channels are either rectangular or triangular. Other shape flutes are possible, but those shown are most common for each type. The weaving of rectangular flutes, however, is a more complicated process. The sewn fluted core has greater diversity in respect to channel shapes. For example, it could easily be transformed into a "high-hat" construction.

The woven fluted core (Raypan) is made by continuous weaving of yarns into an integral structure of two facings separated by the core. Additional facings can be added during the molding operation. Higher shear strengths are obtained by varying the cell size as with the honeycomb cores. The advantage of the integral weave is that core-facing bonds cannot be failed without breaking glass fibers. Although the weave creates stress concentrations where the flute fibers join the facings, peel tests have revealed that these cores are substantially higher than adhesive bonded cores. Molding of fluted cores creates a problem in that mandrels have to be inserted and removed and that control of resin content in the flutes is difficult. In some cases foam mandrels have been used and left in after molding. For complicated curvatures it is necessary to resort to special weaves which are more costly. Table 5 - 23 lists data for Raypan sandwiches made from S-glass.

The sewn fluted cores have been proposed by General Dynamics/Convair for airframe structures, since they overcome some of the disadvantages of the Raypan core. This core is made in two halves. One prepreg facing and half the flute material are sewn on a special machine. The mandrels (polyurethane foam, polypropylene or other materials) can be added one at a time and incorporated into the sewn half. The two halves are mated and molded together. Advantages of this type are: better resin control with prepreg, unrestricted orientation of cloth or number of plies in either skins or flutes, easy forming of corners and splices, possible variations in flute geometry, controlled contours with the mandrel addition, and high peel strength.

The synthetic foams which were used in several early structures, have been displaced as sandwich core materials because of lower shear and compressive strengths. They still find secondary uses as thermal insulation panels or for stabilizing the honeycomb cores in critical areas. The polyurethane isocyanate based foams are the most prominent for such aircraft applications.

TABLE 5 - 23

Data Summary Raypan S-Glass Sandwiches[1]

Source	NAEC	North American
Tensile Flatwise, psi		
R. T.	318	440
160° F, 1/2 hr at 160° F	303	428
-65° F, 1/2 hr at -65° F	326	421
Compression Flatwise, psi		
R. T.	482	506
2-hr water boil	365	454
160° F, 1/2 hr at 160° F	293	469
-65° F, 1/2 hr at -65° F	502	602
Compression Edgewise, psi[2]		
R. T.	5080	--
2-hr water boil	4920	--
Tensile Lengthwise, psi[2]		
R. T.	10200	--
Panel Shear, Ultimate, lbs.		
R. T.	8860	9870

[1] Naval Air Engineering Center-NAEC-AML-1956, June 1964

[2] Based on full specimen thickness

TABLE 5 - 24

Adhesive Characteristics

Adhesive Type	Peel Strength	Shear Strength	Temperature Range - ° F
Phenolic	Medium	Medium	-100 +300
Vinyl Phenolic	Medium	Medium	-100 +250
Rubber Phenolic (Nitrile)	High	High	-300 +500
Epoxy	Low	Medium	-100 +150
Modified Epoxy	Medium	High	-300 +250
Epoxy Phenolic	Low	High	-300 +500
Epoxy Nylon	High	High	-300 +200
PBI	--	--	+1000

ADHESIVES FOR SANDWICH CONSTRUCTIONS

Adhesives for bonding sandwich skins to cores are in widespread use throughout the aircraft industry. They have been investigated in a number of government development programs and no particular material problems are foreseen for fiberglas bonded constructions.

Adhesives selected to bond core and facings are subjected to approximately the same loading conditions as the core itself. In addition they must be consistent with the processing procedure for fabricating the sandwich. High peel strength, a desirable property in an adhesive, need not be a factor since panels can be designed to eliminate most peel forces. High peel strength, however, usually is indicative of resistance to impact and crack propagation. Shear, fatigue, creep and heat resistance are other factors influencing adhesive selection.

Some adhesives are available as partially cured films, usually supported by a light weave scrim cloth. In this form, weight, thickness and distribution of the adhesive are more easily controlled. The carrier fabric also serves to increase the bond strengths.

The general adhesive characteristics are summarized below and in Table 5 - 24.

- Rubber Base - These are usually in solvent solution and are cured by removal of solvents.
- Thermoset Resin with Elastomeric Polymer - Typical are the vinyl-phenolics. These can be solvent solution or supported or unsupported films.
- Epoxy or Modified Epoxy - These are thick liquids or pastes without solvents.
- Epoxy-Phenolic - These are developed for high temperature service. Fillers and carriers are used; solvents avoided. They are supplied as extruded films, supported films or as pastes.
- Combination Tapes - These consist of supported films of thermoset-elastomer modified adhesives, with a liquid epoxy film on one side only. The epoxy, next to the honeycomb, provides resin fillets in the core cells and results in higher strengths.

The more recent adhesive programs have been directed to high temperature uses, in line with general trends towards increased service temperatures for aircraft structures. Typical is the work done at Whittaker-Narmco on PBI based adhesives (291).

SECTION 6. MECHANICAL PROPERTIES OF FIBERGLAS COMPOSITES

STANDARD TEST PROCEDURES

Standard procedures for the mechanical testing of reinforced plastics have been established by the ASTM. Equivalent tests are described in FED-STD-406 for laminates and MIL-STD-401 for sandwiches.

A number of tests, which are variations of ASTM methods, are also used in the plastics industry. Many of the methods employed in these standards and in the non-standardized tests have been adapted directly from metals technology. These methods, based on the testing of small coupons, are used for such diverse purposes as the obtaining of design data, quality control of incoming materials, in-process control, and the evaluation or comparison of reinforcements or resins for research or development studies.

There is general agreement that these tests are useful for quality control, materials evaluation and some development studies. The validity of data from these tests for design purposes is currently being questioned. Some consider that the results are unreliable since test conditions are not related to the use conditions. Others consider that the test methods, on the whole, would be useful for preliminary design if the specimens were made thinner and the materials were tested under a more comprehensive spectrum of loading conditions, including tension, compression, shear and fatigue. Programmed test conditions, simulating those which the aircraft are expected to encounter, could then be imposed on prototypes or bench specimens with greater economy.

DERIVATION OF DESIGN DATA

Data published on the mechanical properties of the fiberglas composites is based almost entirely on the ASTM or similar test methods. Typical is the data included in Military Handbooks 17 and 23. Details such as resin content, glass finish and cure cycles are often not reported so that verification of the material properties is not possible. In many instances the materials tested are either obsolete or out-dated and several years can elapse before newer materials are tested and reported. Stated bluntly, reliable design data is not found in published sources. As a result, each company generates its own data and establishes its own design allowables. This data, as with other design information, is considered proprietary. Typical design allowables are listed in Tables 4-1, 4-2, and 4-3.

The methods for obtaining design allowables vary considerably. In some cases coupon-type testing is retained. Data from Military Handbooks 17 and 23 serve as standards, augmented by additional testing. Correlations are attempted with standardized structures. In other instances, coupon data is used only for preliminary design. Besides the common engineering properties such as tensile and compressive strength, Goodman diagrams, stress concentration factors and other design aids are developed. Certain companies rely on bench tests, closely simulating production conditions, to arrive at the allowable design stresses. Prototype, scale-model and full-scale testing is necessary to verify allowable stresses as well as other features of the design. Results of such structural tests are not reported.

DEFICIENCIES IN TEST METHODS

Reinforced plastics are basically anisotropic. The resin and coupling agent are weak in comparison to the glass fibers. The viscoelastic nature of the resin and coupling agent influences the sensitivity of the composite to stress conditions. This behavior accounts for some of the differences noted in the responses of test coupons and of finished structures to applied loads. Other differences are by assumption. In uniaxial tests it is assumed that a uniaxial stress exists because loads are applied in this way. To calculate stress it is assumed that loads are distributed uniformly over the original cross sections. Strain deformations are also assumed to be uniform. The evidence is that these conditions do not exist.

Test specimens are usually thicker than the laminates normally used in aircraft structures, and properties tend to vary with thickness. Specimens are loaded in a manner which does not simulate aircraft practice. There are no adequate panel or torsion shear tests. Fabrication processes impose further variations, many of which are not taken into account during testing. It has been noted that more than 200 such process variables can exist. Other deficiencies and variations in test method are apparent in the following considerations of specific material properties.

TENSILE PROPERTIES

The unidirectional laminates have higher tensile strengths and moduli than any of the reinforced materials. Maximum strengths are obtained with glass loadings of about 80% by weight, although in aircraft use higher resin contents are normal. Weakness in the cross-fiber direction is offset by plying at ±5° or at greater angles. Strength in the fiber direction, as might be expected, is less affected by the type of resin than are other properties. Testing of the unidirectional laminates show that considerable variation is caused by specimen shape as well as thickness. Typical results are shown in Table 6-1. Reported values for S-glass as nigh as 160,000 to 170,000 psi are not uncommon. For E-glass, tensiles are in the order of 120,000 to 130,000 psi. The woven fabrics with lower strengths show less extremes in directional properties. Optimum tensile strengths are produced by composites in the range of from 65% to 70% glass content. Data reported for S-glass style 181 fabric lists tensile strengths of from 85,000 to 100,000 psi. For E-glass high modulus weaves, tensiles of from 60,000 to 70,000 psi have been obtained.

Stress-strain relationship for the reinforced plastics vary with the type of reinforcement, the percentage and type of resin, and the direction of loading. With unidirectional laminates, the stress-strain curve tends to be relatively straight from initial loading to failure. In other cases the curve can show four distinct regions - an initial straight section, a knee, a second straight section, and a curved section at failure. Preloading, which is resorted to in some stress-strain determinations, affects the characteristics of the curve. Reductions in cross-sections and the point of resin crazing are usually not included in the tests. Under these conditions, selection of a modulus is arbitrary. For the unidirectional materials, moduli as high as 7 to 8 million psi have been reported, but for aircraft laminates, the value is more likely to be between 5 and 5.5 million psi. Woven fabrics will range from 2.5 to 3.5 million psi in the warp directions and may be as low as 800,000 psi in directions as 45° to the warp. The relatively low modulus of the reinforced composites has not been a particular handicap since rigidity is gained by sandwich constructions and the stiffness to weight compares favorably with other materials.

COMPRESSIVE STRENGTH

Compressive tests are possibly least satisfactory for use with reinforced plastics. Tests are conducted as column, flat plate or edgewise compression and in none of these is loading similar to practical conditions found in aircraft structures. Failures may be buckling, bearing or debonding. Results for unidirectional laminates in column compression have varied as much as from 85,000 to 175,000 psi for the same material as the geometry of the test specimen was changed.

The most recent studies of compressive strength have been conducted under Bureau of Ships programs. Although this work is directed toward the use of structural reinforced plastics in underwater pressure hulls, many of the findings have direct bearing on aircraft laminates. Results at the Naval Applied Science Laboratory indicate a close correlation between compressive strength and interlaminar shear. It follows that any improvements of glass-to-resin bond result in higher compression. The limiting strength in a composite may be the yield strength of the resin. Further correlation was found between shear strength and the percentage of voids, in which case the compressive strength varied inversely with the voids content. Prepreg materials had lower voids and higher strengths than the wet layups. For unidirectional laminates, maximum compressive strengths were obtained with glass contents of from 80% to 85% by weight. E-HTS and S-HTS fibers gave the highest strengths of the materials tested. Although the S-glass is a stronger fiber, there were cases in which the E-HTS was stronger in the composite. This behavior was attributed to better resin adhesion to the E-HTS glass. As in tensile strengths, the highest compression strengths were encountered with the unidirectional materials. Typical values for several materials are shown in Table 6-4A.

SHEAR STRENGTH

It is recognized that in the common tests for shear, combined stress action occurs so that true shear strength in any plane is not determined. No single shear test has received acceptance, and methods are being modified in an effort to obtain the principal shear stresses. The unidirectional materials, particularly, present problems because of weakness in the direction at right angles to the fiber. In edgewise or interlaminar shear, the strength of the composite appears to depend on the resin strength. Maximum shear strength is at a higher resin content than is required for maximum tensile or compressive strength. Some tests show good agreement for interlaminar and edgewise shear, but in other tests the interlaminar shear strength is higher. Depending on the test method, values for interlaminar or edgewise shear will range from below 4,000 psi to above 15,000 psi. In cross shear, where fibers are broken, values are higher and may be as high as 30,000 psi for unidirectional materials. A detailed discussion of shear properties and test methods is given in FPL-033, "Methods for Evaluating Shear Strength of Plastic Laminates Reinforced with Unwoven Glass Fibers," by K. Romstad.

FATIGUE PROPERTIES

Potentially, the fatigue resistance of fiberglas composites can be a significant factor in their favor. However, at the present time structural fatigue data is limited. There are no known material combinations which have been optimized for fatigue resistance, and development programs are restricted to applications in deep submersible pressure hulls.

Fatigue failures in reinforced plastics are not catastrophic as is characteristic of most metals. The first indications of failure appear as resin crazing, which continues to the point where compressive loads can no longer be sustained. Repeated tensile loadings produce delaminations long before the panel will fail. In sandwich structures failures usually occur when the fiberglas skins buckle, which indicates a skin-to-core bond failure.

The Bureau of Ships investigations on fatigue have only limited applications to aircraft laminates. These studies are based on 10,000 cycles, low rate of cyclic load application, and the materials are oriented for external pressure loadings only. One significant feature of this work is that ultrasonic test techniques were successfully used to detect resin debonding or crazing after samples had been through 10,000 cycles.

Fatigue is affected by the type and percent of resin. The epoxy resins appear to have better fatigue resistance than either polyesters or phenolics. The effect of resin content is not apparent at a low number of cycles, but is significant at 10 million cycles. For the nonwoven laminates, the optimum range is from 25% - 35% resin.

The nonwoven glass fiber laminates have shown superior performance over the woven fabrics. As unidirectional laminates, fatigue resistance is not as high as might be anticipated, due to a tendency to split in the weak cross-direction. This defect is offset by crossplies at ±5°. With alternate plies at right angles to each other, resistance still remains higher than for style 181 fabric. S-glass has resulted in higher values than E-glass, but much of the increase is due to initially higher strengths.

For most reinforced materials, the fatigue strength at 10 million cycles, based on coupon tests, appears to be in the order of from 25% - 30% of initial ultimate strength. The range may be slightly higher for S-glass. Table 6-2 shows a comparison of fatigue strength for E-glass and S-glass unwoven glass laminates at several ply orientations. Results are given for zero mean stress and for 25,000 psi mean stress in axial loading. Figure 6-1 shows fatigue properties as determined for fiberglas sandwich panels.

EFFECT OF MOISTURE

Moisture penetration is considered as having a deleterious effect on the mechanical properties of the reinforced materials. Improved finishes have been developed to increase "wet strength" retention. Typical results are shown in Section 5 and in Table 6-5. Recent studies have shown that although a small percentage of moisture is actually adsorbed, the effect is analogous to stress-corrosion in metals. Otto at Narmco has indicated that the maximum strength

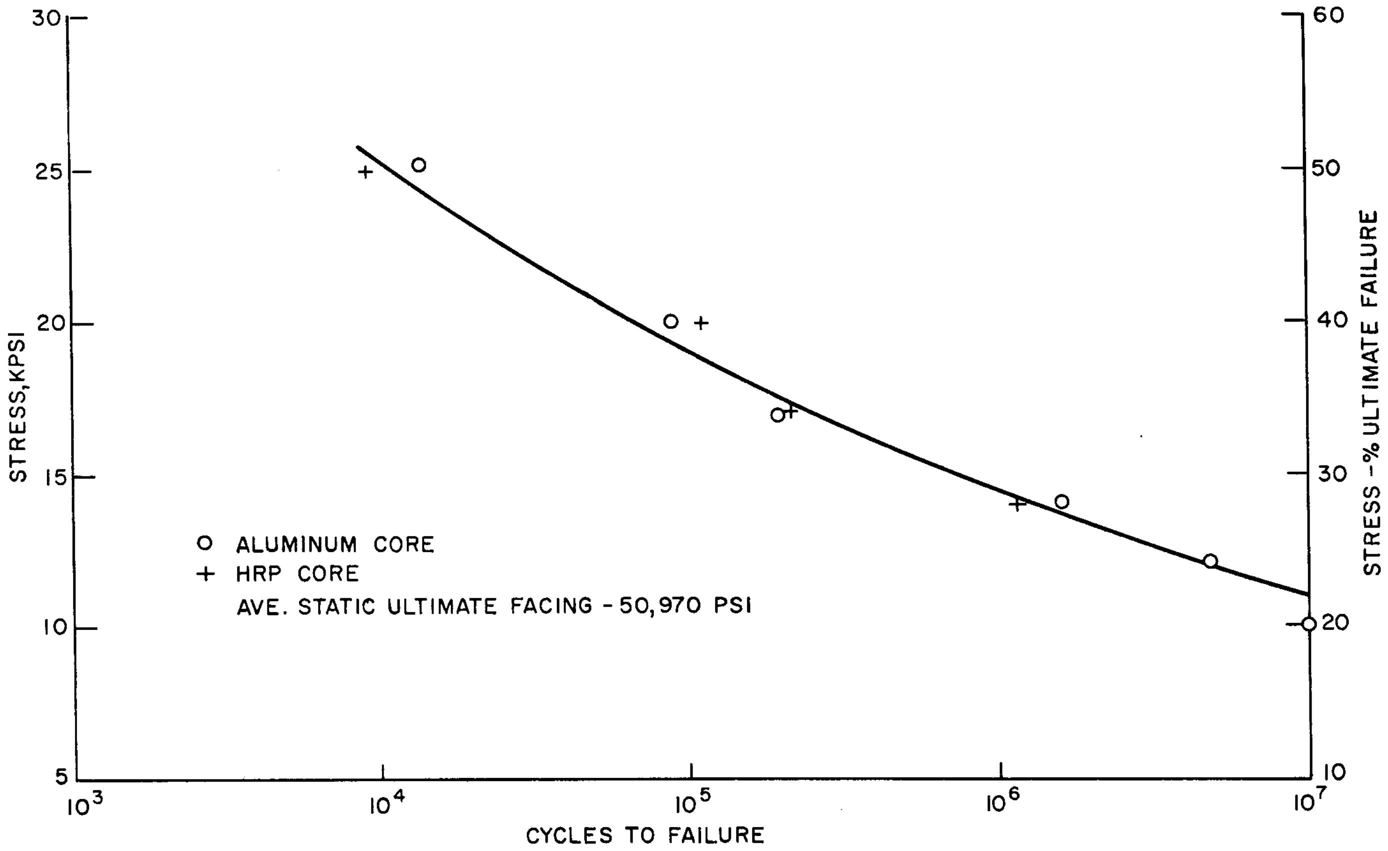


Figure 6-1. SN Curve of Sandwich with FRP Facing

loss in either coated or uncoated fibers is in the order of from 10% - 15% for E-glass after prolonged exposure. This figure, then, would represent the maximum anticipated fall-off in strength. Work at IITRI shows moisture pickup for S-glass cloth and unwoven laminates to be under 0.15%. Exposure to pressurized water for 500 hours had little effect on strength or fatigue properties.

RESISTANCE TO WEATHERING

Weather resistance is primarily a function of the resin system. To a lesser extent it depends on how well the resin and finish protect the glass from mositure penetration and finally on the chemical composition of the glass. Data available today is for materials at least 4 to 5 years old. A 3 year exposure is the normal test period while on rare occasions long range predictions are based on a 10 year exposure. Results tend to be pessimistic. Specimens are tested under the worst exposure conditions (inclined at 45° facing south) and the test stations which give the worst results are cited for expected service behavior. Extrapolation of data beyond the 3 year exposure can be deceiving, since the rate of degradation can fall off so that the lowest levels may be reached in the 3 year period. Performance is measured as a change in tensile, compressive and flexural strength or in the optical properties such as light transmission, yellowing and surface gloss. Each property is found to be affected differently after exposure. In this country only E-glass has been studied. Test results for S-glass have not yet been reported.

Some factors will affect weathering regardless of the resin system. These are the tightness of weave, fineness of surface cloths, surfacing techniques, finish on the glass, and the degree of cure. Any one resin type will be affected by the chemical structure of the polymer, the cross linking agents, the catalysts and additivies, diluents or impurities in the resin. The use of ultraviolet inhibitors generally improves only the optical properties.

The early general purpose polyesters were styrene cross linked and peroxide catalyzed. Surfacing materials were not used and cures were shortened to reduce costs. With these materials, designs were based on an expected 25% decrease in tensile strength as determined from exposures at Florida weather stations. Compressive strengths were found to be unaffected or in fact increased by as much as 25%. This increase was marked by increased rigidity or embrittlement of the resin, making the material weaker in impact strength. It was also noted that after 3 years the compressive strength began to decrease due to resin erosion, crazing or other uncertain reactions. Flexural strength increased for the first 1 or 2 years, but then decreased, sometimes as much as 25%, although a 10% - 15% increase was more common. Long term exposures indicated a loss of about one-third in 10 years. Available data shows that prestressing prior to exposure was not harmful, provided the samples were reconditioned at 50% RH or less. The heat resistant and flame retardant grade polyesters are less durable than the styrene general purpose type. Tensile losses of 30% and compressive losses of 25% can be anticipated. Acrylates such as methyl acrylate or methyl methacrylate give improved optical properties with somewhat improved strength retentions.

There is less data for the epoxy systems and what data is available is mostly for the bisphenol A type. It is noted in these specimens that original strengths were about 15% higher than for the polyesters. Generally the epoxies have good strength retention. Improved durability is claimed for the cyclo-diepoxide types.

Anhydride hardeners yield better properties than the aromatic amines which in turn are superior to the aliphatic amine hardeners. Diluents result in lowered strength retentions. As a generalization, it can be said that tensile strength decreases by about 10% in 3 years. Compression strength is unchanged or increases for periods up to 10 years. Flexural strength tends to increase up to 2 years and then decrease. While a 10% loss is normal for 3 years exposure, some samples have shown only a 3% decrease in 10 years.

Phenolics, used mostly for heat resistance, have been tested less frequently. Data shows them to be slightly inferior to the polyesters and epoxies.

Data on combined loading and weathering for the three resin systems mentioned is contradictory and appears to depend on the load conditions. As an estimate, it can be stated that loadings below 25% of ultimate strength have no effect. Above that figure, the effects of weathering will be greater than without loading.

EFFECT OF VARIATION IN GLASS

The effect on mechanical properties of variability in glass filaments, strands, yarns and fabrics has not been fully established. The program at Owens Corning (noted in Section 5) investigated only certain aspects of glass manufacture. An earlier investigation at DeBell and Richardson (see Section 9) studied variations in cloth weight and thickness, but was inconclusive as to their effects on mechanical properties.

Current glass-drawing practice has been to use a 204 hole bushing, but 408 hole bushings are now in production. Bushing hole size varies from 0.040 inches to 0.080 inches. Fiber diameter is governed by the speed of tube-off as well as hole diameter. It is the opinion at Owens Corning that in a G-fiber, 83% of the filaments will be in the nominal range of from 0.35 to 0.40 mils while the remainder will vary from 0.18 to 0.58 mils. The controlling factor in diameter size has been the yardage produced from one pound of glass. Fiber diameters are not measured; they are simply sorted on a weight-yardage basis. A particular yarn can then be composed of an undeterminate number of strands at a fixed yardage per pound. Weight variation of yarns is estimated to be ± 5%. Further variations are imposed by the amount of sizing and finished applied. The weight of the finish can range from 0.4% to 3.5% of the yarn weight. The number of strands or ends in the rovings can also vary. Specifications for 60-end roving allow a range of from 56 to 60; for 20-end it is from 19 to 20 and for 12-end it is from 11 to 12. Tabulated strand weight variations for E- and S-glass are shown in Table 6-3. The effect of heat cleaning of fabrics after weaving is considered to be detrimental, but to what extend is not known.

INFORMATION SOURCES ON MECHANICAL PROPERTIES

The Forest Products Laboratory has been the major source for coupon-type data. The basic material properties, test methods and methods of analysis have been developed at this laboratory for both laminates and sandwiches. The test program, which has been continued for about 20 years, has produced the data published in Military Handbooks 17 and 23. The Laboratory has established the mechanical properties under static loads, the directional properties, environmental effects and time-dependent effects for various types of resins and reinforcement combinations. A brief resume of this work is included in Appendix D. Data so far reported by Forest Products Laboratory has been evolved from work with E-glass reinforcements. The basic work on S-glass laminates has been completed and is now awaiting publication. Their most recent work on the fatigue properties of directional S-glass laminates is available in preprint form. The laboratory is now preparing revisions of Military Handbooks 17 and 23.

A second source of information has been the data purchased by the Air Force from a number of aerospace companies including General Dynamics Corporation, North American Aviation, McDonnell Aircraft Corporation, Boeing Company, Hughes Aircraft Company, and Douglas Aircraft Company. Essentially, this data represents qualification and evaluation tests on a wide variety of materials. There has also been an emphasis on properties at elevated temperatures. Typical General Dynamics/Convair data is shown in Table 6-5. The general topics covered in these contracts include: evaluation of unidirectional E-glass cloths, bond strengths of adhesives, resistance of materials to aircraft fuels, and evaluation of airfoil erosion protection materials (127, 128, 184 to 190). This data also tends to be outdated.

Government sources for such data include the Naval Air Engineering Center, the Engineering Research and Development Laboratory, and the Plastics & Packaging Laboratory of Picatinny Arsenal. Elevated temperature properties have been developed by the Air Force primarily at Southern Research Institute. A major foreign source is the Royal Aircraft Establishment at Farnborough. The information on properties furnished by material suppliers is generally disregarded, and is useful for identification purposes only.

Data taken from production runs represents a potential information source. Unfortunately, such data is rarely made available. Table 6-6 shows results of tests performed by Grumman Aircraft Engineering Corporation on samples taken from each radome fabricated for the A-6A. These tests were part of the quality control program for the radomes, and they relate performance to the physical and mechanical properties of the laminates. It is of interest to note that structural performance was improved after replacing the original polyester resin with an epoxy.

CONCLUSIONS

There are no worthwhile tests for the mechanical properties of reinforced plastics which will allow the analytical prediction of the life of a structure. Cumulative damage effects have not been investigated nor have any of the methods for fatigue life prediction been applied.

The reinforced composites require some sort of classification and standardization. It is suggested that five basic material types, listed in Table 6-4, are most appropriate for aircraft structures and can serve as a nucleus for initial investigations. Accumulation of data on these materials in both laminates and sandwiches would be helpful in establishing material standards.

No backlog of data on the testing of primary or highly stressed structures exists. There have simply not been enough such structures to test. Results from other fiberglas structures are not applicable to aircraft. Development of structural data, concurrent with analytical procedures for determining material behavior in terms of reinforcement and resin properties, is a prerequisite to judicious use of the composite materials.

Table 6-1. Tensile Strength - Unidirectional Epoxy Laminates*

Gage length, ins.	9 3/8	11 3/8	13 3/8	15 3/8
1/4 inch - net section width				
Av KPSI	89.5	98.7	107.4	70.1
Av KPSI, ends reinforced	118.4	--	126.6	--
1/8 inch - net section width				
Av KPSI	103.9	--	120.7	116.9
Av KPSI, ends reinforced	116.7	--	--	--

*FPL - 052, August 1964
1/8 inch thick specimens

Table 6-2. Comparison of S-glass and E-glass Unwoven Glass Laminates in Fatigue*

	S-glass - KPSI	E-glass - KPSI
Zero Mean Stress		
Unidirectional	40	30
Alternate plies ± 5°	40	32
Alternate plies - 0° and 90°	26	20
25,000 psi, Mean Stress		
Unidirectional	58	40
Alternate plies ± 5°	53	--
Alternate plies - 0° and 90°	42	--

*Preprint, AFML-TR-64-403

Table 6-3. Strand Weight Variation

Type	E-HTS		S-HTS*	
	Bare	Finished	Bare	Finished
G-140 fiber, % ±	2.5	3.0	4.0	4.6
G-67 fiber, % ±	6.0	6.5	--	--
K-37 fiber, % ±	4.1	4.6	--	--

*Supplied as G-fibers only

Table 6-4. Property Range - Epoxy Laminates with Various Reinforcements

Type	Tensile		Compression		Glass Content
	Strength KPSI	Modulus psi x 10^6	Strength KPSI	Modulus psi x 10^6	% Wt.
S-glass - Unwoven					
0°	160-180	6-7	100-130	5.5-6.5	60-80
± 5°	140-160	6-7	85-115	5.0-6.0	60-80
Crossply - 0° and 90°	100-120	3.5-4.0	70-85	3.5-4.5	60-75
S-glass - 181/HTS	85-100	3.0-4.0	60-65	3.5-4.5	65-72
E-glass - Unwoven					
0°	100-125	5.0-5.5	70-90	4.0-5.0	60-75
±5°	85-120	5.0-5.5	65-85	4.0-5.0	60-75
Crossply - 0° and 90°	60-80	2.8-3.5	55-75	2.8-3.5	60-70
E-glass - 7581/S550	60-75	2.8-3.5	55-60	3.0-4.0	60-70
E-glass - 181/Volan	50-65	2.8-3.5	45-60	3.0-4.0	60-70

Table 6-4A. Uniaxial Compressive Properties*

Material	Resin	Glass Type	Orientation		Glass Content % Wt.	Ult. Plate Compressive Strength - Av KPSI		Modulus of Elasticity 10^6 psi	Poisson's Ratio	
			Axial	Transverse		Constant Neck	Large Radius		Face	Edge
Scotchply 2004	--	E-HTS	2	1	52.6	40	68	3.1	--	--
Scotchply 2004	--	E-HTS	1	0	52.0	61	65	--	--	--
Scotchply 1002	--	E-HTS	2	1	66.9	75	--	--	--	--
Scotchply 1009	--	E-HTS	2	1	80.6	97	103	5.3	0.138	0.329
Scotchply 1009	--	E-HTS	1	0	79.9	110	123	7.0	0.227	0.287
E787	E787	E-HTS	2	1	78.7	108	106	5.7	0.185	0.321
E787	E787	E-HTS	1	0	80.8	135	153	7.6	0.252	0.273
Scotchply XP2345	E787	S-HTS	2	1	77.0	102	107	6.2	0.178	0.345
E787 - S cloth	E787	S/143 cloth	2	1	66.4	71	76	4.2	0.147	0.397
E787 - S cloth	E787	S/143 cloth	1	0	66.2	74	84	5.0	0.218	0.362
E787 - S roving	E787	S-HTS	2	1	82.7	106	118	6.6	0.145	0.307
E787 - S roving	E787	S-HTS	1	0	83.3	132	142	8.7	0.243	0.250
E787 - PPHG	E787	PPG Hollow Glass	2	1	73.5	107	110	4.0	0.183	0.350
E787 - PPHG	E787	PPG Hollow Glass	1	0	71.4	140	155	5.4	0.258	0.258

*IITRI data

Table 6-5A Laminate Properties of 181 Cloth

Volan A / Epoxy, Epon 828-MPD						Volan A / Phenolic, Conolon 506						Garan / Polyester, Laminac 4232, ATC					
Variable	Property	Value	Variable	Property	Value	Variable	Property	Value	Variable	Property	Value	Variable	Property	Value	Variable	Property	Value
0^o/Dry/RT	TS	43.2	0^o/Wet/RT	TS	43.4	0^o/Dry/RT	TS	44.7	0^o/Wet/RT	TS	42.3	0^o/Dry/RT	TS	41.3	0^o/Wet/RT	TS	40.8
	TM	2.97		TM	3.05		TM	3.12		TM	3.2		TM	2.71		TM	2.78
	CS	52.4		CS	47.4		CS	48.0		CS	40.6		CS	41.2		CS	29.7
	CM	3.48		CM	2.84		CM	3.98		CM	3.29		CM	3.03		CM	2.82
	PR	0.137		- -	- --		PR	0.145		- -	- --		PR	0.086		- -	- --
	BS	52.6		BS	46.3		BS	48.6		BS	40.5		BS	43.3		BS	33.8
45^o/Dry/RT	TS	26.1	45^o/Wet/RT	TS	25.4	45^o/Dry/RT	TS	24.22	45^o/Wet/RT	TS	23.4	45^o/Dry/RT	TS	19.8	90^o/Wet/RT	TS	40.3
	TM	1.75		TM	1.65		TM	2.17		TM	2.16		TM	1.17		TM	2.58
	CS	28.0		CS	26.4		CS	31.3		CS	28.6		CS	23.8		CS	27.7
	CM	1.45		CM	1.31		CM	2.57		CM	2.12		CM	1.44		CM	2.72
	PR	0.513		--	- --		PR	0.38		--	- --		PR	0.338		- -	- --
	BS	51.2		BS	47.2		BS	48.1		BS	44.1		BS	47.5		BS	34.6
90^o/Dry/RT	TS	44.1	90^o/Wet/RT	TS	42.0	90^o/Dry/RT	TS	46.95	90^o/Wet/RT	TS	44.37	90^o/Dry/RT	TS	42.1	45^o/Wet/RT	TS	17.0
	TM	3.01		TM	3.22		TM	3.15		TM	3.41		TM	2.71		TM	0.94
	CS	56.5		CS	48.9		CS	45.9		CS	39.9		CS	43.5		CS	16.7
	CM	3.14		CM	2.93		CM	3.6		CM	2.99		CM	3.76		CM	0.79
	PR	0.165		- -	- --		PR	0.14		- -	- --		PR	0.079		- -	- --
	BS	48.7		BS	46.7		BS	50.0		BS	39.5		BS	46.4		BS	36.0
0^o/100hr/300^oF	TS	32.2	0^o/1/2hr/500^oF	TS	23.3	0^o/100hr/300^oF	TS	36.07	0^o/1/2hr/500^oF	TS	38.9	0^o/1/2hr/500^oF	TS	35.2	0^o/1/2hr/500^oF	TS	33.4
	TM	2.65		TM	1.76		TM	3.22		TM	3.06		TM	2.89		TM	2.43
	CS	25.2		CS	4.3		CS	30.8		CS	33.3		CS	29.8		CS	18.9
	CM	2.84		CM	2.05		CM	3.25		CM	3.43		CM	2.52		CM	2.43
	BS	32.0		BS	6.4		BS	43.4		BS	40.4		PR	0.068		- -	- --
45^o/100hr/300^oF	TS	13.7	45^o/1/2hr/500^oF	TS	3.4	45^o/100hr/300^oF	TS	20.75	45^o/1/2hr/500^oF	TS	18.0		BS	35.7		BS	26.8
	TM	0.87		TM	0.19		TM	2.11		TM	1.56	45^o/100hr/300^oF	TS	16.0	45^o/1/2hr/500^oF	TS	10.1
	CS	11.7		CS	3.8		CS	20.7		CS	18.3		TM	1.09		TM	0.60
	BS	30.2		BS	7.2		CM	1.64		CM	2.28		CS	16.5		CS	11.5
							BS	44.8		BS	35.5		BS	37.9		BS	27.3
Details: Reinforcement, % - 58 - 69 Thickness: 0.110" - 0.138" Fabric construction: 12-14 plies, vaccum bag. Cure cycle: 150^oF/ 1/2hr; 200^oF/1hr; 250^oF/ 1/2 hr. Post cure: 200^o-400^oF, 5 hr schedule.						Details: Reinforcement, % - 71.8 Thickness: 0.100" to 0.152" Fabric construction: 13-15 plies, press mold Cure cycle: 250^oF to 350^oF/1/2 to 2 hrs. Post cure: 300^oF -350^oF, 3 hr schedule.						Details: Reinforcement, % - 62.4 Av. Thickness: 0.104" to 0.140" Fabric construction: 12 plies, vacuum bag Cure cycle; 200/2hr; 250/1hr; 300^oF/ 1/2hr. Post cure: 200^o -500^oF, 6 hr schedule.					

Table 6-5B Laminate Properties of 143 Cloth

Volan A / Epoxy, Epon 828						Volan A / Phenolic, Conlon 506						Garan / Polyester, Laminac 4232					
Variable	Property	Value	Variable	Property	Value	Variable	Property	Value	Variable	Property	Value	Variable	Property	Value	Variable	Property	Value
0°/Dry/RT	TS	63.45	- - -	- -	- - -	0°/Dry/RT	TS	74.12	- - -	- -	- - -	0°/Dry/RT	TS	59.8	- - -	- -	- - -
	TM	3.88	- - -	- -	- - -		TM	5.64		- -	- - -		TM	4.52		- -	- - -
	CS	63.1		- -	- - -		CS	75.2		- -	- - -		CS	47.6		- -	- - -
	CM	4.08		- -	- - -		CM	5.43		- -	- - -		CM	4.43		- -	- - -
	PR	0.215		- -	- - -		PR	0.278		- -	- - -		PR	0.21		- -	- - -
	BS	29.7					BS	29.3	- - -				BS	30.2		- -	- - -
45°/Dry/RT	TS	13.9	- - -	- -	- - -	45°/Dry/RT	TS	10.17	- - -	--	- - -	45°/Dry/RT	TS	12.15	- - -	- -	- - -
	TM	1.26					TM	2.11		- -	- - -		TM	1.11		- -	- - -
	CS	24.7					CS	26.2		- -	- - -		CS	2.27		- -	- - -
	CM	1.39					CM	2.19		- -	- - -		CM	1.09		- -	- - -
	PR	0.313					PR	0.258		- -	- - -		PR	0.16		- -	- - -
	BS	36.1					BS	26.4		- -	- - -		BS	27.3		- -	- - -
90°/Dry/RT	TS	10.72	- - -	- -	- - -	90°/Dry/RT	TS	6.97	- - -	- -	- - -	90°/Dry/RT	TS	9.6	- - -	- -	- - -
	TM	1.49					TM	2.14		- -	- - -		TM	.76		- -	- - -
	CS	25.5					CS	22.9		- -	- - -		CS	21.0		- -	- - -
	CM	1.46					CM	2.03		- -	- - -		CM	.78		- -	- - -
	PR	0.093					PR	.061		- -	- - -		PR	.051		- -	- - -
	BS	31.0					BS	20.4		- -	- - -		BS	23.3		- -	- - -
0°/1/2hr/300°F	TS	47.7	- - -	- -	- - -	0°/100hr/300°F	TS	65.67	0°/1/2hr/500°F	TS	53.72	0°/100hr/300°F	TS	48.23	0°/1/2hr/500°F	TS	52.52
	TM	3.81					TM	5.85		TM	5.47		TM	4.44		TM	4.0
	CS	17.3					CS	61.4		CS	35.6		CS	34.0		CS	22.6
	CM	2.8					CM	4.61		CM	4.90		CM	3.57		CM	3.94
	BS	18.2					BS	27.9		BS	22.1		BS	26.7		BS	22.0
45°/1/2hr/300°F	TS	7.27	- - -	- -	- - -	45°/100hr/300°F	TS	8.87	45°/1/2hr/500°F	TS	7.62	45°/100hr/300°F	TS	9.92	45°/1/2hr/500°F	TS	6.8
	TM	.37					TM	1.43		TM	1.35		TM	.80		TM	.46
	CS	4.0					CS	23.0		CS	15.4		CS	16.0		CS	9.5
	BS	19.4					BS	22.9		BS	19.4		BS	22.5		BS	18.1
Details: Reinforcement, % - 63.0 Av. Thickness: - - - Fabric construction: Same Cure cycle: Same						Details: Reinforcement, % - 62.5 Av. Thickness: 0.087" to 0.137" Fabric construction: Same Cure cycle: Same						Details: Reinforcement, % - 64.5 Thickness: 0.102" to 0.128" Fabric construction: Same Cure cycle: Same					

Legend:
TS - Tensile Strength, KPSI
TM - Tensile Modulus, PSI x 106
CS - Compressive Strength, KPSI
CM - Compressive Modulus, PSI x 106
PR - Poisson's Ratio
BS - Bearing Stress, KPSI

Test temperature: RT, 300°F, 500°F
Test direction: 0°, 45°, 90°

Table 6-6 Condensed Grumman Data - A6A Filament Wound Radome

RADOME NUMBER	COMPRESSION, KPSI						TENSILE, KPSI						FLEXURE, KPSI		
	Circumferential			Longitudinal			Circumferential			Longitudinal			Circumferential		
	Individual Value Av	Values, Spread of 10 Max. Av	Min. Av	Individual Value Av	Values, Spread of 10 Max. Av	Min. Av	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av
6	47	-	-	41	-	-	95	-	-	30	-	-	79	-	-
14	38	47	26	46	50	32	94	97	77	84	89	44	98	100	71
24	29	50	34	44	54	40	54	113	70	66	83	57	81	100	59
34	47	53	29	50	51	43	90	95	72	76	79	61	52	93	52
44	50	56	37	52	65	50	68	77	60	80	95	80	68	87	39
54	55	55	41	46	58	46	77	77	63	73	87	73	73	87	61
64	51	55	43	60	61	45	87	88	68	102	107	75	97	100	58
74	59	59	53	60	67	57	88	92	76	101	106	91	99	105	89
84	51	54	51	61	65	53	78	93	77	107	108	88	98	101	91
94	50	58	43	56	62	53	84	91	78	100	112	88	94	102	77
104	51	59	50	60	67	50	74	94	74	95	106	87	93	93	80
120	62	62	52	61	66	54	88	93	70	102	103	93	94	102	86
131	60	60	53	62	67	61	81	90	60	100	108	99	115	116	80
-															
132	65	-	-	69	-	-	74	-	-	106	-	-	100	-	-
133	57	-	-	64	-	-	81	-	-	101	-	-	115	-	-

RADOME NUMBER	FLEXURE (Cont)			PIN SHEAR, KPSI						RESIN CONTENTS, %[d]			VOID CONTENT, %			RADOME BEHAVIOR		
	Longitudinal			Circumferential			Longitudinal											
	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av	Individual Value (Av)	Values, Spread of 10 Max. Av	Min. Av	Sat	Re-ject	Unsat
6	34	-	-	5.9	-	-	7.0	-	-	17.7	-	-	-	-	-	-	1	-
14	68	71	33	7.2	7.7	5.7	7.7	8.1	5.9	15.5	18.6	14.6	-	-	-	3	2	5(1)[a]
24	63	67	54	7.3	7.8	6.4	7.2	8.4	7.2	13.8	16.6	13.8	7.5	-	-	5	2	3
34	60	67	56	6.6	8.0	5.9	7.1	8.3	6.9	16.6	16.8	15.9	6.8	7.9	5.8	7	1	2(2)[a]
44	64	73	46	7.2	7.3	5.3	7.4	8.1	5.4	16.3	16.7	15.6	8.0	8.5	7.6	6	4	0
54	48	71	43	5.9	8.4	5.9	7.2	8.4	6.3	16.2	17.3	15.9	7.8	8.2	6.3	6	3	1(1)[b]
64	91	94	38	8.6	9.0	5.5	8.5	9.2	6.4	17.2	18.5	16.3	7.2	8.5	6.7	6(4)[c]	2	2(2)[b]
74	89	94	83	8.5	8.9	7.9	8.6	9.2	7.8	16.9	18.4	16.9	7.9	8.1	6.7	10(8)[c]	0	0
84	92	96	77	9.3	9.3	7.9	9.4	9.5	7.2	17.6	18.3	17.2	6.4	7.6	6.4	10[c]	0	0
94	89	91	68	7.5	8.8	6.9	8.2	9.1	7.7	15.6	18.1	15.6	7.8	7.8	6.7	10[c]	0	0
104	76	94	66	7.7	8.6	7.4	8.3	8.5	7.0	17.8	18.3	15.7	7.4	7.9	7.3	10[c]	0	0
120	82	90	79	8.3	8.7	7.7	8.7	10.0	8.3	16.6	17.8	16.6	8.5	8.7	7.6	10[c]	0	0
131	104	104	90	9.0	9.1	8.4	9.1	9.6	8.3	17.4	18.3	16.3	7.4	6.6	8.4	10[c]	0	0
-	-																	
132	96	-	-	8.2	-	-	8.8	-	-	17.6	-	-	7.8	-	-	1[c]	-	-
133	92	-	-	8.9	-	-	9.8	-	-	17.1	-	-	7.4	-	-	1[c]	-	-

a - failed in flight

b - delaminated - scrapped

c - resin change to epoxy/MNA

d - Sp Gr. Range - 1.98-2.05

SECTION 7. APPLICATION OF THEORETICAL ANALYSIS TO AIRCRAFT STRUCTURES

Initial analytical treatments for fiberglass composites were developed about 20 year ago. These included methods for design of laminates and sandwich constructions. Coincident with the growth of filament winding, netting analysis concepts were devised to aid in design of the wound structures. Limitations of the netting analysis have led to modifications and to a greater interest in orthotropic analyses, currently receiving widespread attention. Of particular concern to aircraft designers at present are the simpler extensions of the netting analysis, the more comprehensive orthotropic analyses, and some recent work on sandwich structures.

The analytical methods for laminates have a common purpose: to relate external load reactions to the directional properties of the laminates and to predict behavior in terms of those directional properties. Their end results are structurally balanced designs in which material is efficiently placed to meet directional load conditions and in which weight penalties can be minimized. The newer approaches for sandwiches are seeking more accurate solutions and a lessening of the discrepancies found between theory and actual practice.

MODIFIED NETTING ANALYSIS

The netting analysis received its greatest impetus in design of rocket motor cases and it is now a well-established design tool for internally pressurized filament windings. It is characterized by a complete disregard for the elastic constants. Stress calculations depend entirely on the strength capability of the reinforcing fibers and their orientation in the structure. The netting analysis concept considers the reinforcements as forming a net membrane. It presumes that the continuous reinforcing fibers lie in stable geodesic paths and do not fold or crimp. Only the fibers resist the applied tensile loads; all the fibers are uniformly stressed. The resin-matrix functions to protect the fibers from external effects and to fill in the interstices between adjacent fibers. The matrix is assumed to carry no loads. A limitation of the netting analysis is that it can be applied only to tensile loading. Bending discontinuities, shear or compressive-buckling loads can not be calculated. Effects of a thickness parameter and interactions between laminate layers are neglected.

The modified analysis, as extended to aircraft structures, makes some compensation for these inadequacies, while still retaining the concept of a fiber net-forming structure. Procedures have been modified so that the entensional stiffness of the fibers can be used in the analysis. For example, in designing a typical sandwich and determining the effects of loads upon it, the assumption is made that the facings provide no flexural rigidity, but develop extensional stiffness in tension and rigidity in compression. The honeycomb or foam cores are assumed to have no extensional stiffness. They serve to separate the facings by the necessary distance and to furnish shear resistance to expected deflections. The compressive and tensile faces are designed on the basis of glass fiber strength alone. Fibers are placed in directions to resist the loads and the amount of fiber can be varied as needed. When shear in the plane of the facing is to be resisted, the principal shear directions are determined. Sufficient fibers are then placed at 45° to the shear directions, so that shear distortions are resisted by fibers in tension. The modified analysis can also be applied in cases where woven cloths are used in place of unidirectional reinforcements. Load defomations are resolved in terms of unit loads per end in both the warp and fill directions.

These methods have been found to be effective in design of spars, beams, helicopter rotor blades, tail fins, and wing surfaces. They have been successfully used at Boeing-Vertol for a number of parts and in the design of a complete aircraft. At other companies (like Aerojet-General and Hercules, which are extensively engaged in filament winding) modified netting methods are employed for design of aircraft parts.

The motivation behind Boeing-Vertol's use of netting analysis extensions is of interest since it sheds light on the need for such methods and their ultimate goals. Most design criteria are governed by the amount of deflection or vibration an aircraft has to withstand. Generally, fixed wing aircraft are subjected to critical deflections while rotary wing types are critical in vibration. There are exceptions where strengths, particularly compressive, dominate the design. In any event, design allowances must be determined to meet the critical loading conditions. These loadings, which originate at the specific mission requirement level, will vary for each design. Design data as gleaned from standard ASTM tests have proven to be inadequate. More dependable information is obtained by relying on fiber strengths as in the netting analysis extensions, together with a judicious use of bench tests. These bench tests can be closely correlated with structural performances. They are more in accordance with finished designs, since they are made under the respective plant fabrication conditions.

A formal mathematical treatment of either the netting analysis or its extension is not contemplated here. Complete presentations are given or are listed (62, 246, 248).

ORTHOTROPIC ANALYSES

Recent attempts to establish orthotropic analytical methods have been conducted at a number of organizations. A partial listing of companies which have published reports includes:

Aerojet-general (230)	-	Company Funded
Boeing (29)	-	Company Funded
Douglas (62)	-	Company Funded
Forest Products (284)	-	Government Sponsored
General Electric	-	NASA Contract
Hercules (151, 230)	-	Company Funded
Lockheed (173)	-	Company Funded
Philco (242)	-	NASA Contract
Rohm and Haas	-	Army Contract
Whittaker (292)	-	Company Funded and Army Contract

As a characteristic common to all of these methods, they use distinct elastic constants for the principal material directions. Stress tranformation equations, strain tranformation equations, and the generalized Hooke's Law enter into the analysis. In contradistinction to netting analysis, which can solve only simple two-dimensional membrane problems, orthotropic analyses can be applied to almost any type of problem.

In a typical plied fiberglas laminate, properties in the fiber direction are greater than in a direction transverse to the fibers. Properties in the direction of the laminate thickness will be less than those in the longitudinal or transverse directions. These principal material directions are considered to form mutually perpendicular axes. Orthotropic analysis sets out to predict material reaction to extended loading in terms of properties in these principal materials directions. In special cases where laminates are cross plied to give uniform properties in planar directions, orthotropic equation forms will degenerate to the common isotropic solutions for metals. In bending, however, solutions remain orthotropic, since properties in the thickness direction are still different.

The significance of orthotropic analysis is that it relates the directional properties of the laminate to the structure so that the classical equations of Timoshenko and others can be applied.

A simple summation of orthotropic methods is given in the following paragraphs. Generally they will fall into either of three groups as typified by the three methods listed below:

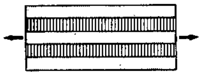
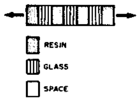
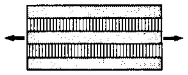
- The method developed by Greszczuk at Douglas Aircraft and containing procedures for filament wound structures based on fiberglas and resin properties. A summary of other similar methods is given by Nourse and Amick (230).
- The earlier method, typified by analyses recommended by Forest Products Laboratory. Separate contributions of the fiber and resin are not considered. Mathematical procedures are given for determining effects of external loading applied in any direction in terms of experimentally established laminate properties in principal material directions.
- A method proposed by Shaffer of NYU, which is a combination of the two above. The gross material directional properties are calculated on the basis of glass and resin properties. These properties are then used in procedures similar to those recommended by Forest Products Laboratory.

Table 7-1 summarizes the distinguishing characteristics of these methods.

Elastic Analysis - Greszczuk

Of the more recent publications on orthotropic analysis, an updated and comprehensive treatment is given by Greszczuk of Douglas Aircraft (62). It contains the development of the analysis from the known tensil properties of the resin and the glass to those of cylinders, domes, and conical structures of multilayer laminates subjected to buckling, torsional and bending loads.

Table 7-1. Comparison of Three Currently Used Orthotropic Analyses

Source	Douglas Aircraft Company (Greszczuk)	Forest Products Laboratory	Ekvall Shaffer
Data Required	. Elastic constants for resin . Elastic constants for glass . Volumetric Constant of glass or resin . Allowable strain in resin	Mechanical properties of gross composites.	. Elastic constants for resin . Elastic constants for glass . Volumetric constant of glass or resin . Allowable strain in resin (Note: Same as Douglas)
Type of Reinforcement	Unidirectional fibers	. Woven fabrics . Unidirectional fibers . Oriented whiskers	Unidirectional fibers
Schematic of Mathematical Models	Longitudinal (Calculated)  Transverse (Calculated) Slab 	Single layer  Tested in both directions	Longitudinal (Calculated)  Transverse (Calculated)  or  (Resin distribution generally determined from a close-packed model.)
Alignment of Layers in Multilayer Construction	. Axes of symmetry at right angles, and must coincide . Vertical orientation . Horizontal orientation . Pairs of equally reinforced layers oriented at ± angles with axes	Any direction	Any direction
Limitations	Laminate must be balanced to resist applied load	Unbalanced loading can be considered	Unbalanced loading can be considered
Assumptions	Materials are elastic	Materials are elastic	Materials are elastic
Distinctions	Distinction is made between the free elastic constants of laminates composed of all fibers oriented in a common direction, and the restrained constants where alternate layers are oriented in + α direction and - α direction.	Mutual influence of differently oriented laminates are ignored.	Mutual influence of alternately oriented laminates ignored.

Fundamentals of netting analysis are outlined and expressions for determining design properties of multilayer filament wound structures are developed. The analogy between the procedures for analyzing filament wound structures and laminates made from parallel fiber reinforcements is apparent.

The progression of calculations is such that the elastic constants are developed in an orderly sequence. They start with the established stress-strain relations of the resin and glass which are obtained from conventional tests. These are used to determine the following constants needed for the analysis:

E_r - Young's Modulus for the resin

M_r - Poisson's Ratio for the resin

E_f - Young's Modulus for the fiber

U_f - Poisson's Ratio for the fiber

It is generally presumed that the resin and fiber are isotropic supercooled liquids of infinitely high viscosities. A volumetric fiber/laminate ratio, K, is established. It is required to determine the contributions of the reinforcement and the resin binder to the mechanical properties of the laminate.

Theoretically, K is a maximum when the parallel fibers form a hexagonal or close packed laminate. If the resin fills only the voids between the fibers, K will be 0.92. In practice, however, K for laminates composed of unidirectional filaments can be most reliably controlled over the range from 0.65 to about 0.78. The volumetric ratio of 0.78 and lower allows the use of a square packed model. With this model equations yielding closer approximations are derived. Greszczuk first considers a laminate composed of resin and parallel fibers which is loaded in the fiber direction. Equations for the unrestrained elastic constants are derived. Secondly, an oriented laminate is considered in which the loading is at an angle to the fiber direction. Again the unrestrained elastic constants are calculated. A third laminate is then considered. This is composed of alternate layers. Fibers in each layer are unidirectional. One layer is oriented at an angle $+\alpha$ to the load direction; the second layer is oriented at an angle $-\alpha$. Additional layers as added must retain this symmetry; that is, they are parallel to the preceding alternate layer. Added in pairs, they maintain a balance as well as symmetry. This arrangement simulates the layer pattern in a helical type filament wound structure. The elastic constants are then calculated. In this case they are restrained.

The sequence of equations to determine the elastic constants for single layer balanced laminates are summarized in the following steps:

1. The "elastic" constants for the resin and the glass are determined by mechanical test.
2. The volumetric ratio of glass to laminate and resin to laminate are established is line with expected practical fabrication conditions.
3. The data from steps 1 and 2 are used to calculate the constants for a unidirectionally reinforced laminate in the longitudinal and transverse directions.
4. The constants determined in step 3 are used to determine the contants for a unidirectional laminate oriented as an angle α to the principal directions of loading.

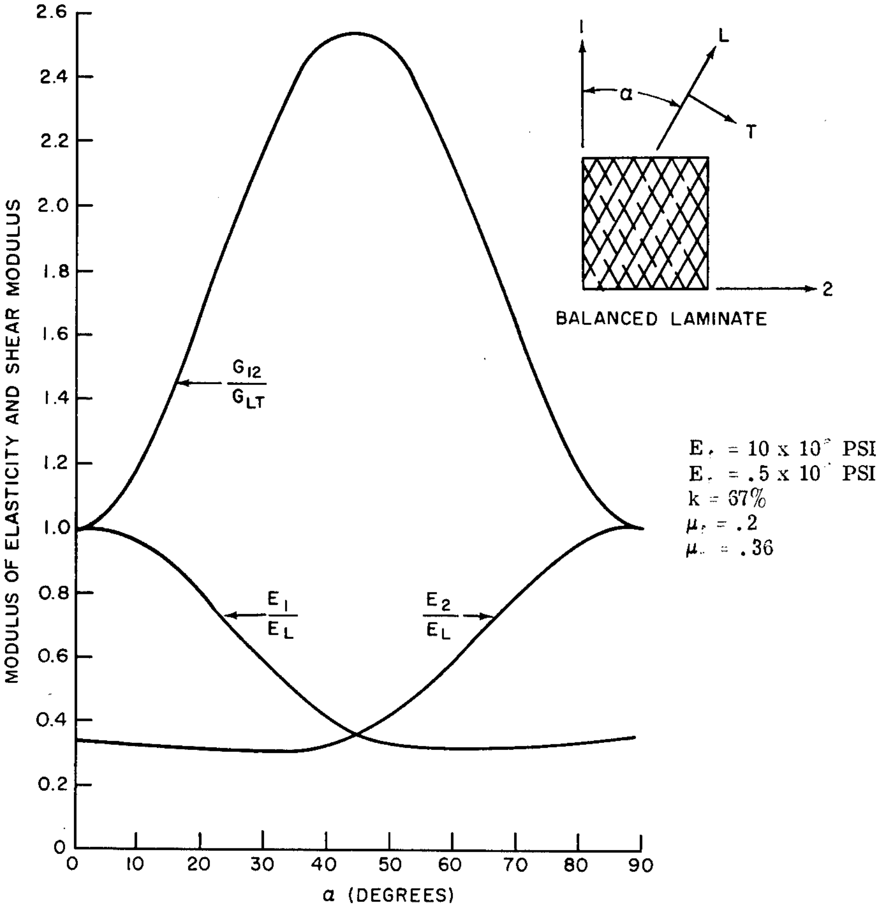


Figure 7-1. Variation of Modulus of Elasticity and Shear Modulus with Orientation Angle (Restrained Elastic Constants)

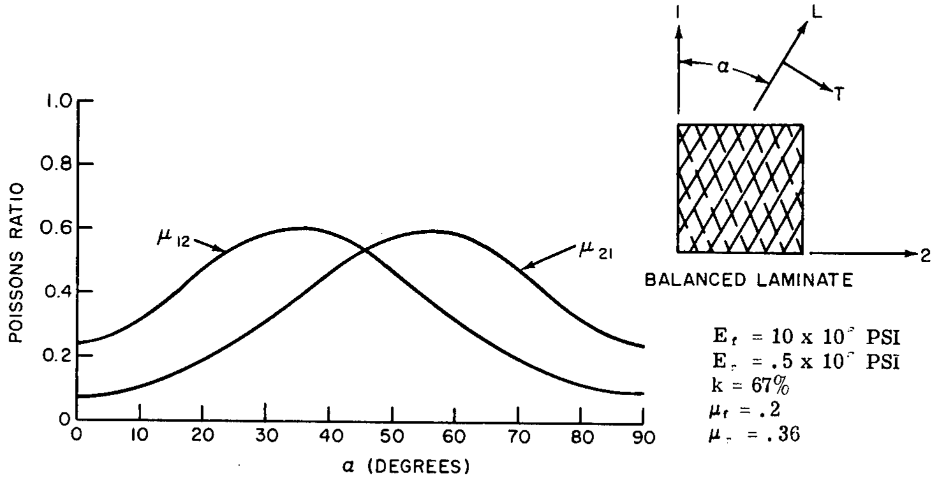


Figure 7-2. Variation of Poisson's Ratio with Orientation Angle (Restrained Elastic Constants)

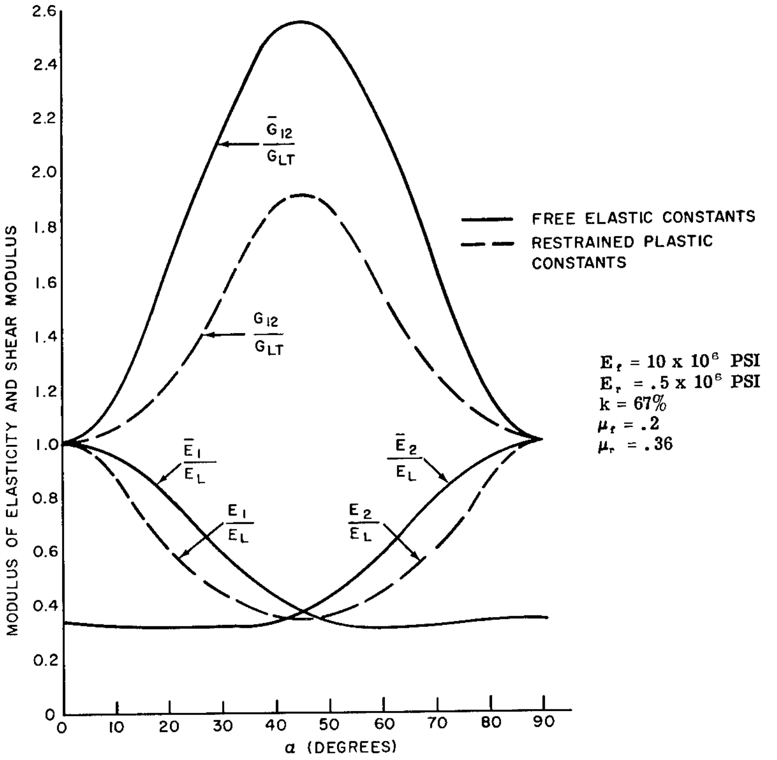


Figure 7-3. Comparison of Free and Restrained Elastic Constants

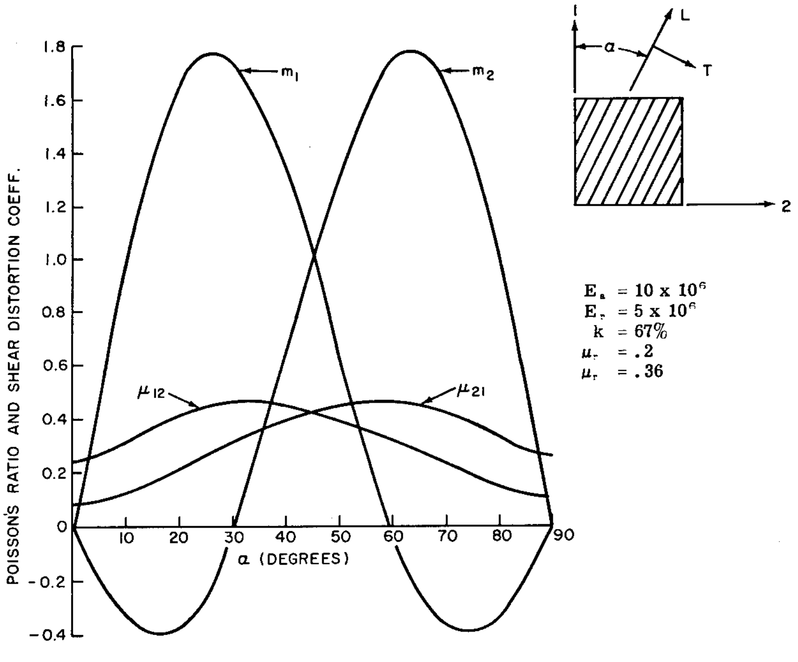


Figure 7-4. Variation of Poisson's Ratio and Shear Distortion Coefficients with Orientation Angle (Free Elastic Constants)

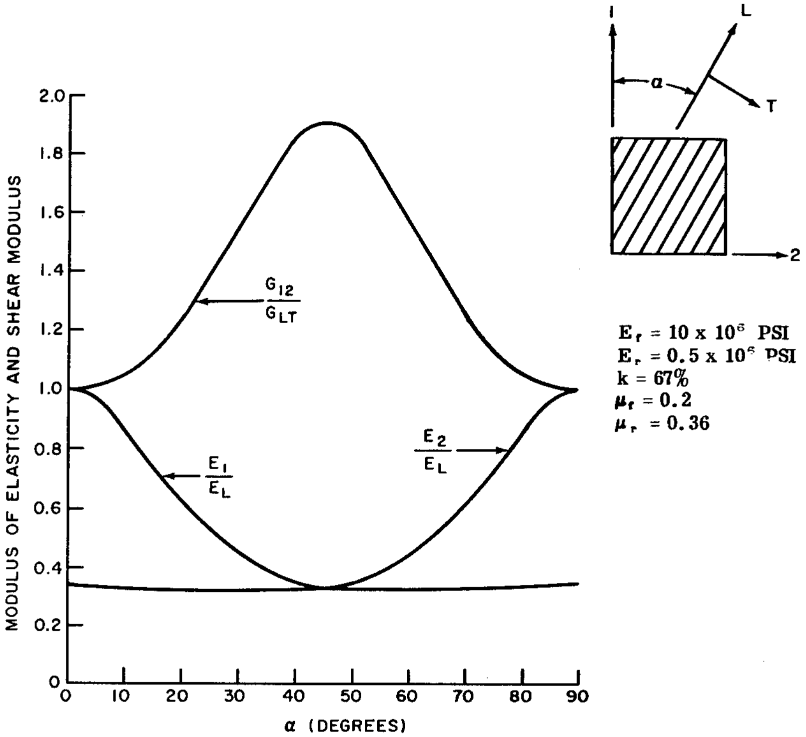


Figure 7-5. Variation of Modulus of Elasticity and Shear Modulus with Orientation Angle (Free Elastic Constants)

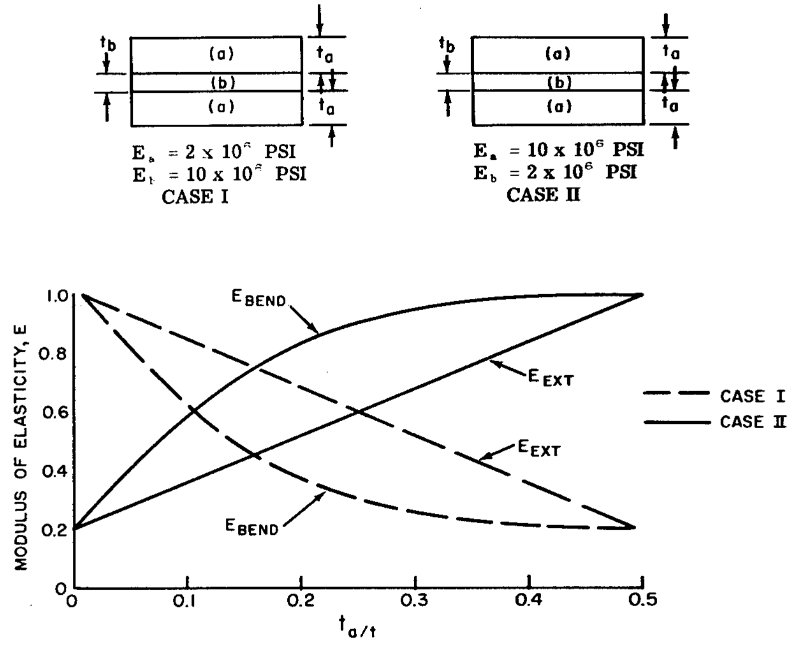


Figure 7-6. Comparison of Extensional Bending Moduli of Elasticity

5. A balanced laminate is assumed to be composed of two unidirectional laminates bonded securely at an angle 2 α to each other. The constants from procedures contained in step 4 are used to calculate the unrestrained constants for each laminate oriented at angle α to the load direction. These constants are then used to calculate the restrained constants for the balanced laminate in the 1 and 2 directions. (See Figures 7-1 to 7-3 for directional notation.)

It can be shown that a balanced laminate can be designed to maximum efficiency to resist any system of coplanar loads such as loads in directions 1 and 2 and shear with respect to directions 1 and 2. Conversely, it can also be shown that to develop the design properties of the laminate it is necessary to impose the intended loading system.

Subsequent equations consider the constants for multilayer laminate and the effects of laminate thickness. Composites which are made up of different layers, each of which is oriented at a different angle, can be handled provided that the axis of symmetry is common to all layers.

Figures 7-4 and 7-5 show the elastic constants including the shear distortion coefficients as they vary with α. Figure 7-6 shows the effect of thickness on extensional and bending moduli for a low modulus facing containing a high modulus core, and a high modulus facing containing a low modulus core.

Elastic Analysis - Forest Products Laboratory

This analytical method has been well-known for years and is only briefly touched on here. For details see Military Handbooks 17 and 23, or Reference 284. It provides a mathematical system by which effects of external loading on a composite structure can be related to the mechanical properties as experimentally determined. It is required that these properties be known in the principal material directions or along the natural axis of the material.

If α, β and ζ are taken for natural axes of the laminates, the α - axis could be made to coincide with the fiber direction for unidirectional reinforcements and with the warp direction for woven cloths. The β -axis is transverse to the α axis in the laminate plane. The ζ -axis is in the thickness direction.

In cross-plied laminates, where the natural axis for different layers do not coincide, it is necessary to establish an arbitrary common set of axes. The contribution of each layer as determined for its natural axis is related to the new set of axes. Such mathematical relations provide means for calculating the magnitude and direction of the principal design stresses. A variety of orthotropic laminates can be handled this way and it can be extended to include development of aeolotropic properties as well.

Elastic Analysis - Shaffer

Those approaches to orthotropic analysis which were devised to combine the advantages of both preceding systems are typified by the procedures derived by Shaffer (New York University). Table 7-1 shows the essential differences between the methods. Knowledge of the procedures used in the Greszczuk method and in the Forest Products method will enable the designer to understand and use this method.

In Shaffer's method, once the material properties for a single layer laminate are determined for the principal material directions, the value can be substituted into the equations for the procedures contained in the Forest Products method.

Mechanics of Resin-Glass Systems - Narmco

This method is currently under development on a TRECOM contract. Because of time limitations, it was not possible to review this work. As a consequence, a summary is directly quoted (293).

"Equations for stress and strain in a composite have been derived which include one set for the fibrous reinforcement, and another set for the matrix. General differential equations and special solutions for stress and strain distribution also have been formulated which describe the complete stress and strain fields in both components and account for their interactions."

"Equations indicate an undulating distribution of stress not only along one axis, but also along three orthogonal axes. Additionally, these equations show that uniformly distributed external loading produces undulatory distributions of internal stress and strain. This is contrary to expectation that only the residual stresses and strains would be sinusoidal. Another unforeseen fact of importance is that the wavelength of the stress and strain patterns is zero at the fiber axis and increases radially therefrom."

"The stress and strain distribution appears constant around one particular circle concentric with a single fiber of infinite length whether the matrix containing the circle is finite or extends infinitely along the fiber. However, the stress and strain distribution is periodic lengthwise along a cylindrical surface concentric with a single fiber. Also, when many parallel fibers are placed in a matrix, the stress-strain distribution in the matrix undulates around any circle concentric with any particular fiber."

"Some of the solutions that have been obtained are constrained by the boundary conditions of one fiber. However, almost all of the general equations that have been derived are applicable to multifibered composites. The extension of the analyses from single-fiber to multifiber systems is now proceeding."

Tensor analysis has been used extensively in this work, not only because of its efficiency of mathematical condensation, but also because its rules intrinsically contain many physical laws.

Mechanics of Resin-Glass Systems - Kies

The analytical methods of Narmco, Greszczuk and others indicate the need to optimize resin content of a laminate and to hold it to close controls during processing. An additional effect to consider is the strain induced in the resin under load conditions. As newer reinforcements with higher moduli are developed, the effect of resin strains will be of even greater importance.

Kies (212) of Naval Research Laboratory has investigated the maximum strains of fiberglass composites and has shown that a magnification factor of strains in the resin due to transverse strains in the laminate can be as high as the ratio of fiber modulus/matrix modulus. He has related the strain magnification factor to the volumetric fraction of flass fiber (K) in a unidirectional laminate. Tables 7-2, and 7-4 summarize the relationship of strain magnification to K, the volumetric fraction of glass. The ratio of glass modulus to resin modulus is taken to be 20. Figures 7-7 and 7-8 show the square array and close packed array of fibers used in deriving the relations between K and strain magnification.

SANDWICH CONSTRUCTIONS

Many of the basic equations for sandwich designs and reactions of sandwichs to specific loaded conditions have been published by Forest Products Laboratory. These and others developed for the aircraft industry are summarized in Military Handbook 23 and other reports. A Russian treatment for sandwich constructions is given in Reference 13. More work has taken place at Stanford University (271), the University of Oklahoma (232) and Dyna/Structures (66).

In general, theories for sandwich behavior have not shown close agreement with test results. A major reason for these differences lies in the fact that many variables are introduced in the fabrication of sandwiches. The more practical aspects of sandwich constructions are treated in Section 10.

Table 7-2. Effect of Glass Content on Tensile Strain Concentration in the Resin, Square Array (212)

Δ/R	Glass Volume Fraction	$\epsilon_{xr}/\bar{\epsilon}_x$
1.0	0.349	2.73
0.5	0.503	4.17
0.2	0.650	7.33
0.1	0.713	10.5
0.05	0.748	13.6
0.02	0.770	16.8
0.01	0.778	18.3
0	0.786	20*

*$\epsilon_{xr}/\bar{\epsilon}_x = E_g/E_r$

ϵ x R - strain in resin

ϵ_x - total strain

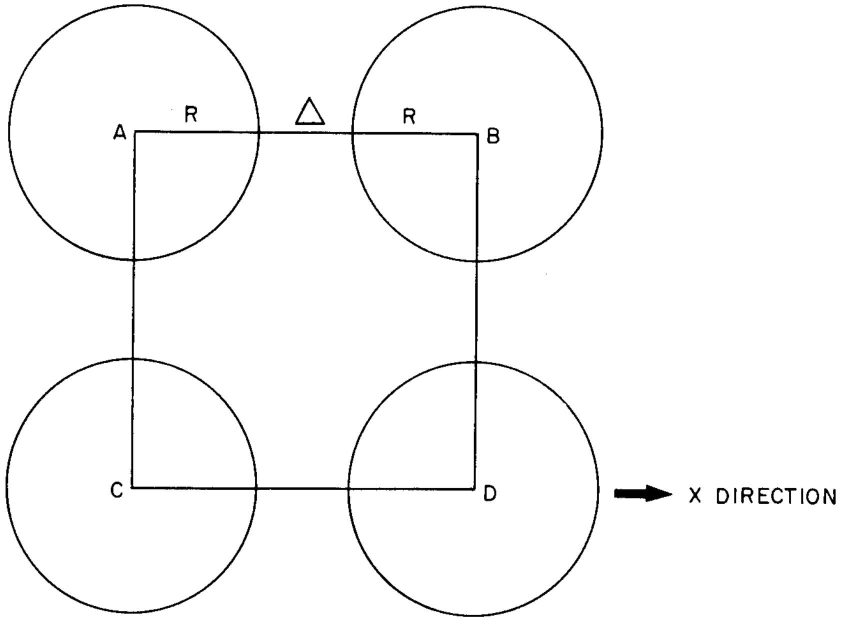


Figure 7-7. The Square Array of Rods in Resin

The unit cell extends one unit of length in the direction of the rods normal to the section shown. Tensile strains are considered in the x direction transverse to the rods. The resin spacing between glass rods in an ideal square array is Δ.

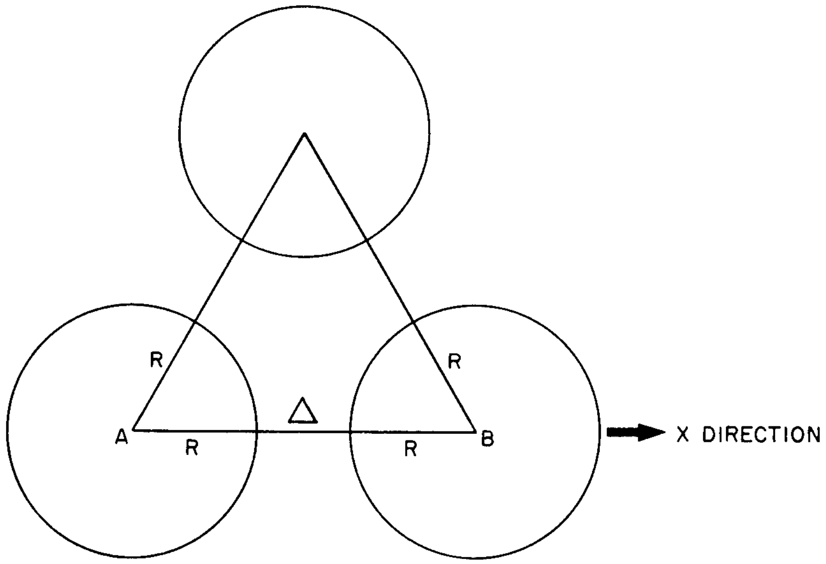


Figure 7-8. The Hexagonal Close-Packed Array

The tensile strains are considered along the line of centers AB.

Table 7-3. Shear-Strain Magnification in the Resin for a Square Array; Strain Direction Across Fibers (212)

Δ/R	Volume Fraction Glass	${}_r/\bar{}_r$
0.5	0.502	3.9
0.3	0.594	5.3
0.2	0.741	6.6
0.1	0.765	9.4
0.05	0.776	12.1
0.0	0.786	17.0*

* ${}_r/\bar{}_r = 0.71\ (G_g/G_r)$

${}_r$ - shear strain resin

$\bar{}_r$ - total shear strain

Table 7-4. Strain Magnification in Resin for Close-Packed Array, Tensile Strain Across the Fibers (212)

Δ/R	Volume Fraction Glass	$\epsilon_{xr}/\bar{\epsilon}_x$
1.0	0.403	2.72
0.50	0.58	4.16
0.36	0.65	5.12
0.20	0.75	7.32
0.10	0.82	10.5
0.05	0.86	13.7
0.02	0.888	16.8
0.01	0.896	18.2
0.00	0.905	20

SECTION 8. DESIGN CONCEPTS AND PROCEDURES FOR FIBERGLAS STRUCTURES

Designs reviewed in this section have been selected to illustrate basic differences between fiberglas and metal techniques and to indicate the potential and versatility of the composite materials. Design changes are traced for those cases where metals were gradually replaced by plastics. The general trends in fiberglas constructions are noted.

Limited material is available from which a judgment can be formed. In the first place, only a few primary structures have been built of fiberglas. Early wing and fuselage designs can be discarded since either the raw material, fabrication processes or design concepts are outdated. Present plastic designs for small commercial aircraft do not appear suitable for military types. What remains, then, are designs of helicopter rotor blades, programs for construction of fiberglas box-beams, and the proposed designs for an "all-plastic" aircraft. Certain joint designs which have been evaluated are also of value, since they relate to total structures.

It is emphasized that optimum solutions to design problems for FRP constructions differ radically from those of metals. A fundamental difference arises from the way in which finished properties are achieved in these types of construction. The properties of metals are controlled primarily by alloying, forging, heat treating, rolling, cold forming, and stretching operations performed at the mills. With the glass reinforced composites, the properties are determined by the choice of raw materials, the lay-up of the reinforcements, the curing, the post-cure, and the control of fabrication variables. The objective of manufacturing is not only to form the materials to shape, but to build in the required properties.

CLASSICAL DESIGN CONCEPTS

Two types of design concepts can be drawn from the widely varied proposals for the use of glass reinforced plastics in aircraft. The classical concepts are those already used in the design of metal aircraft; and the newer design concepts are those based upon plastics technology. In either case adaptations are possible which will incorporate metal and fiberglas into the construction or in which only fiberglas composites will be used.

In the classical approach to the design of metal aircraft, it is frequently necessary to derive simplified analogues for the stress analysis of complex aircraft structures. Two favored simplifications are the cylinder and the box beam. The cylindrical configurations generally provide a basis for analyzing the fuselage, tail booms, fuel tanks and similar components. The box beam configurations provide the analogues for the wings, horizontal stabilizers, vertical fins and other similarly shaped airfoil structures.

Application of the box beam and cylinder concepts to metal aircraft has been highly successful. Sophisticated analyses have been developed and translated into efficient computer programs. Lockheed, North American, Bell, Douglas and General Dynamics/Convair are known to have such programs in operation.

It can be assumed that most other aircraft companies have adopted similar programs. These programmed analyses for metal structures are used to determine the configurations necessary to satisfy maximum load conditions. Solutions are in the form of the number and type of beams, spars, ribs, skin panels, joints or other structural members that are required.

BOX BEAM PROGRAMS

Certain companies have started to adapt these methods to plastic aircraft structures. North American, for example, is applying their computer program to the design of a box beam for Naval Air Engineering Center (143, 145, 225) and presumably it will be used for design of their "all plastic" aircraft. After the configuration has been optimized, the method then develops the stress patterns and directional load distribution of the airframe structure. The program provides for the stress analysis of a structure subjected to any of 15 different combinations of loading conditions. In order to handle glass reinforced plastics, it was necessary to modify the analytical system. As modified, the procedure is adaptable to the stress analysis of orthotropic plates and is similar to the Forest Products Laboratory method. Any two-dimensional component in the structure can be analyzed. It provides for a choice from among 20 possible glass reinforced materials, including the established woven glass cloths or unidirectional tapes. The choice is optimized to give the highest strength to weight. Photostress techniques have been tried at North American for experimental verification of the analysis. As reported, the photostress test data provides feedback corrections which are then used to attain the optimum directional properties in the structure.

Based on a lifting surface design optimization, North American selected five plastic configurations for preliminary study: corrugated sandwich (Raypan) multispar, honeycomb sandwich multispar, corrugated sandwich multirib, stiffened skin multirib, and solid skin multispar. The first two turned out to be the lightest structures and consequently the following variations were singled out for a more detailed study:

- Multispar Raypan sandwich reinforced with extra laminations of 181 - S-glass.
- Multispar Raypan sandwich reinforced with extra laminations of unidirectional S-glass.
- Multispar honeycomb sandwich reinforced with extra laminations of 181 - S-glass.
- Multispar honeycomb sandwich reinforced with extra laminations of unidirectional S-glass.

Panels to be fabricated on the contract will be honeycomb sandwich with facings of 181 - S-glass reinforced with unidirectional S-glass plies. The spars will be constructed from Raypan. The first such panel built passed a pure bending test at 150% of design load, but failed in torsion at 130% of design ultimate.

Table 8-1 lists a number of box beams which were proposed to the Naval Air Engineering Center, including North American's (115, 134, 225). Figure 8-1 shows how the fiberglas constructions are adapted to these box beams. A similar fiberglas structure is now being built for NASA by Whittaker-Narmco (293). In this case the panel will be tested under thermal as well as mechanical loading.

Table 8-1. Proposed Box Beams

Company	Type	No. of Cells	Construction	Glass/Resin
General Dynamics/ Convair	Conventional	1	Sewn fluted core, 2 plies in flute, 6 plies outer face, 4 plies inner face	181 - S-glass/ epoxy
North American	Spars and ribs	2	Sandwich skins, aluminum honeycomb core. Raypan spars and ribs. Facing 181 plus undirectional	181 - S-glass and Unidirectional S-glass/ epoxy
Goodyear	Two ribs	3	Sandwich skins, aluminum honeycomb core. Facings 181 plus 143. Ribs solid laminate	181 - S-glass and 143 - S-glass/ epoxy
Goodyear	Two spars	3	Solid laminate. 3 Boxes wrapped together. Additional layers on top facing	181 - S-glass and 143 - S-glass/ epoxy
Whittaker - Narmco	Four ribs	1	Sandwich skins. Fiberglas PE core. Facings 1581, 1543 and unidirectional. Ribs are sandwich, same as skin	1581 - S-glass 1543 - S-glass 1009 - S-glass/ epoxy

The beams are usually assembled by mechanical fastening or bonding the skin panels to the ribs and spars. Goodyear has suggested an integral structure in which three boxes are wrapped together to form a box beam of three cells. The spars are thus fabricated as solid laminates. The compression facing, however, is a sandwich. Such a technique readily lends itself to the filament winding process. Convair has proposed a single cell box in which the facings are made from sewn fluted core stabilized by foam inserts.

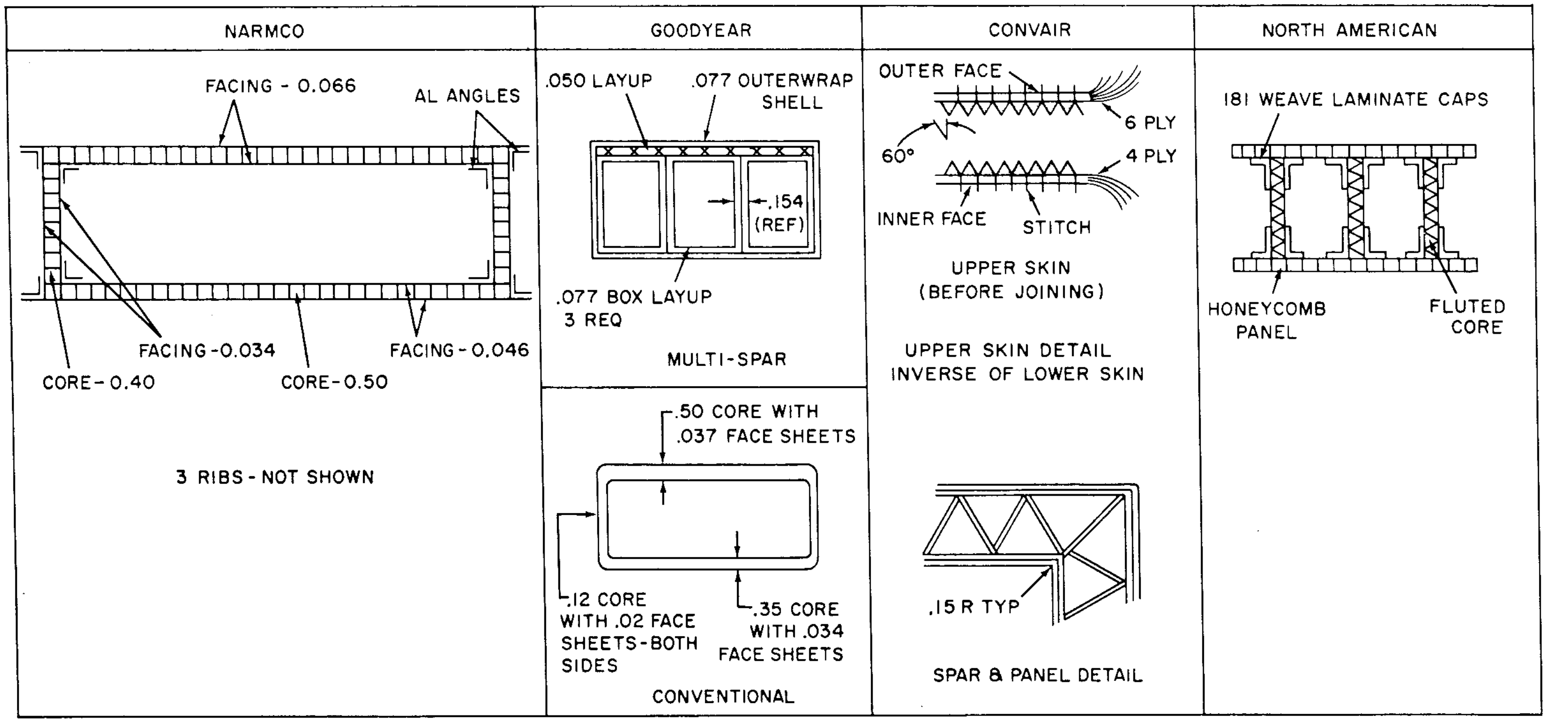


Figure 8-1. Proposed Box Beam Construction

The experimental box beams are subjected to bending, compression, shear, torsion, combined bending and torsion, and cyclic loading tests. Joints are usually tested separately. In more complex situations, cantilevered box beams can be constructed to provide for the mounting of outboard engines, wing tanks or landing gears. However, fiberglas programs have not reached this design stage.

Conversion of box beams to wings, stabilizers or rudders is a relatively simple procedure. Figure 8-2 indicates schematically how a box beam is adapted to a wing by addition of a leading and trailing edge.

The cylindrical analogues for the fuselage have received little attention in

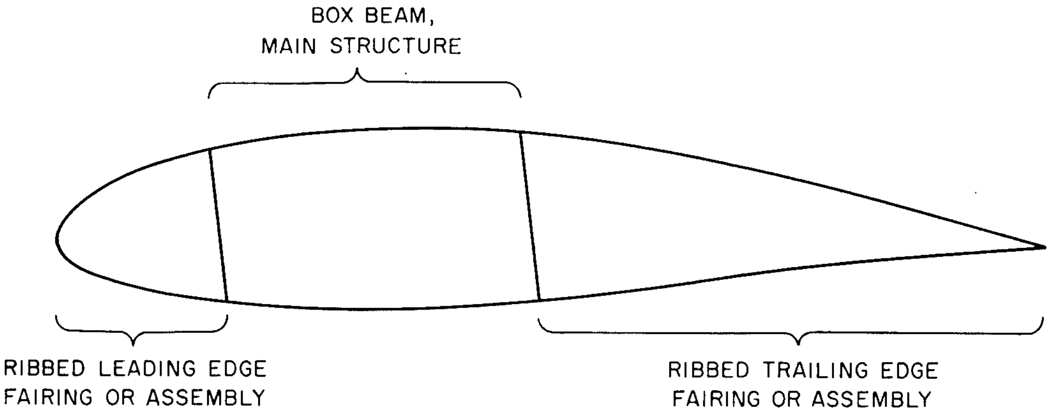


Figure 8-2. Adaptation of Box Beam to Wing

fiberglas. Presumably they can be constructed as rib, spar or ring stiffened cylinders with sandwich facings or as simple sandwich structures. Hercules Powder Company has run some tests on filament wound sandwiches with a variety of core materials as a preliminary step in designing the aft fuselage for the proposed North American YAT-28E plastic airplane.

NEWER DESIGN CONCEPTS

As the design departs from the classical concept, it becomes possible to make greater use of the advantages offered by plastics technology. Considerations which have led to the newer approaches are based on the fact that the airfoil surfaces of a "plastic" aircraft can provide structural integrity. This is in contrast to older designs in which it was expedient to devise simplified load carrying members to be contained within the structure. By making more effective use of the surface panels, weight penalties can be minimized.

The versatility of plastic constructions is perhaps best demonstrated in the development of the "all plastic" helicopter rotor blade. Discussion here is based primarily on work done at Boeing-Vertol (31, 34). Similar developments have taken place at Kaman Aircraft and others (168, 55).

Rotor blade performance requirements are briefly summarized:

- The blade must resist static droop.
- In flight it must maintain rigidity while supporting maximum vehicle loads and while subject to aerodynamic loads. Excessive upward deflection results in diminished lift. Longitudinal stiffness is imparted by structural members or as in Boeing designs by balanced weights at the blade tip. Stiffness in the chord direction is developed to maintain the airfoil.
- It must resist the diagonal warp that can result from the use of variable pitch mechanisms used to adjust the pitch to varying speed conditions. Rotational tip speeds vary from near sonic in forward motions to near stalling in backward motion.

In metal blades, the main structural spar (with Boeing-Vertol a D-spar) was formed from tubular stock. Steel and then aluminum were used. The spar as formed contained a connecting shank and a transition section before the D-section. A twist was imparted to the D-section to give a variable pitch to the airfoil. Ribs transmitted panel loads to the D-spar, the panels functioning simply as fairings. To prevent wrinkling of the panels, ribs were spaced at frequent intervals or a combination of ribs and stringers was used. Initially, joints were riveted. Later, adhesives were substituted at secondary connections.

In modified versions, the number of ribs and stringers was reduced by replacing aluminum fairings with fiberglas laminates. At the same time they were harder to dent and easier to repair. These glass fairings were best suited to adhesive joining. For the bonds to function as designed, it was necessary to maintain close tolerances on the distances between rib tabs and the corresponding attachment points on the D-spar. Since the twisting operation on the D-spar distorted these dimensions, the adhesive bonds became critical. Initial whirl tests of these blades have resulted in loss of fairings. It was presumed that similar failures would have occurred with aluminum fairings.

Analysis of such failures showed that the ribs were not securely bonded to the spar and the attachment of the rib to the spar was achieved only through the bonds of rib-to-fairing and fairing-to-spar. When there was no bond of rib-to-spar, the existing discontinuity gave rise to peel stresses. Lifting of the rib while it was in a warped condition produced a peeling action. When a fairing was lifted high enough to have its edge caught in the slip stream, it was torn off. The solution in this particular case was to hold closer tolerances on the D-spar, but a cost penalty was incurred in so doing. These blades are currently used on the Chinook.

The "all fiberglas" blade was developed under government contract to meet more severe service conditions expected. These Chinook blade replacements have been successfully whirl tested and flight tested but have not yet been installed on the helicopter as standard equipment (31).

The newer blade design uses the torsion resisting fiberglas skin as a structural member. Fibers are oriented to minimize plane shears and to provide bending stiffness. The spar, in this instance a C-configuration, is also of fiberglas. Aluminum honeycomb serves as blade filler, and it is stabilized in the chord direction by a series of sandwich ribs.

Details of the blade are shown in Figure 8-3. Essentially it is fabricated in one bonding assembly of three sub-assemblies.

Spar Assembly - Consisting of the spar proper, root end socket, and foam mandrel. The cross section of the spar proper changes from circular at the root to D in transition and to a final C-configuration. The spar is molded from "Scotchply-1002" unidirectional glass-epoxy and includes an hour glass shaped root end. The foam supports the transitional section of the spar during cure.

Skin Assembly - This is made from "Scotchply-XP 114", a biased crossply, which tapers from four layers at the root end to three and then two layers. The tip end is reinforced to pick up concentrated loads from the weighted tip. The trailing edge is reinforced with "Scotchply-1002".

Leading Edge Assembly - An abrasion strip is cold formed from 0.014 inch thick 18-8 type stainless steel. A flourocarbon extrusion, Fluorosint, is added to convey anti-icing fluids to slits in the leading edge. A permanent mass balance rod machined from 4130 steel is inserted through the Fluorosint and mechanically secured.

Bonding Assembly - This consists of the three sub-assemblies plus the stabilized aluminum honeycomb filler, root rib, and tip rib. These are bonded with PM 1000-191 adhesive.

The attachment of the blade to the root end is of interest. Following the bonding assembly, a split clamp is installed over the hourglass section. Although tests indicated that the clamp holds the blade securely, epoxy resin is used to bond the spar to the clamp. In this way the low interlaminar shear of the laminate, which would control in a bolted joint, is avoided. Other bonded joints, it is noted, are subjected to direct shear in the strongest direction of the adhesive bond.

Testing of materials and components was carried out with coupons made by the same processes and under the same conditions as planned for production. Samples were tested statically and in fatigue. The tests showed that the unidirectional materials were insensitive to propagation of cracks originating on the skin or the stainless steel.

Aside from such functional features as rotor trim tabs and variable pitch controls, the Kaman rotor blade differs from the Boeing blade in other respects. The Kaman blade depends on structural stiffness developed in the blade to prevent excessive static droop or deflection during flight. It also retains some of the features developed for the wooden spar stiffened blade.

This blade contains a D-spar built in two sections. The leading edge channel is formed of 12 layers of Scotchply oriented at $\pm$ 20° to the direction of the spar. The closing channel for the D-spar was .150 inch thick style 181 glass cloth laminate. The two sections were bonded together to form the D-spar. A fiberglas skin was wrapped from the fiberglas trailing edge over the honeycomb core and leading edge spar and back again to the trailing edge spar. Blade grips were used to connect the blade to the hub. Bolts through the root end of the spar clamped the blade between the blade grips. The endurance limits attained by these blades are 45, 000 inch-pounds in bending and 3, 000 inch-pounds in torsion. The Kaman blades have been flight tested for several years.

PROPOSALS FOR ALL-PLASTIC AIRCRAFT

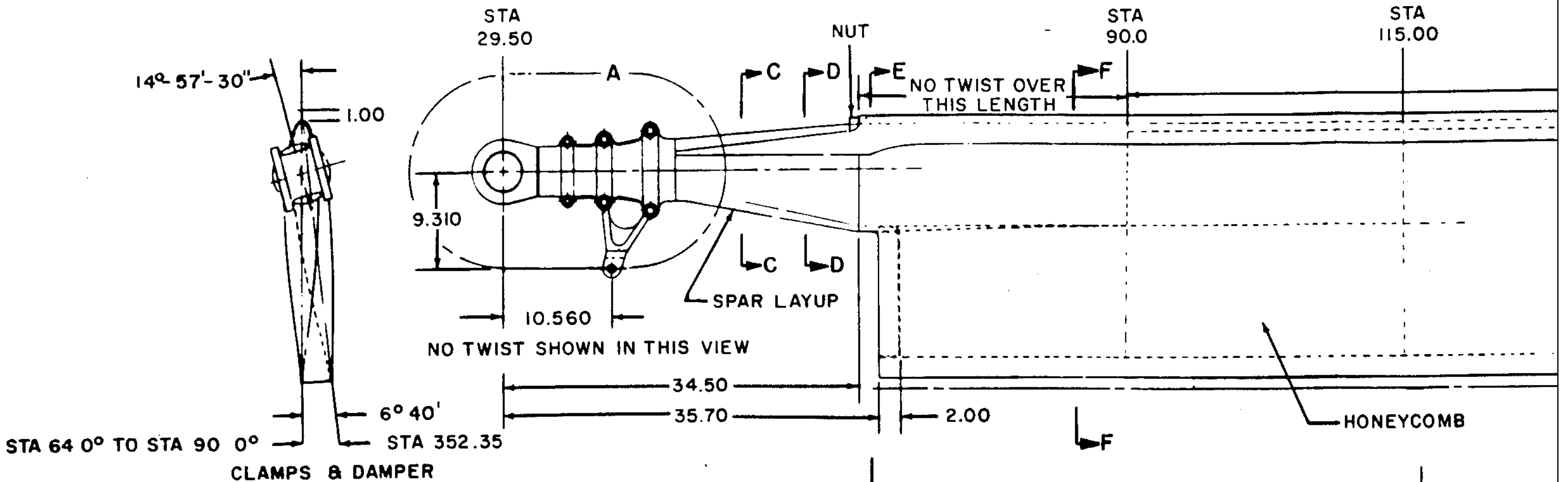
Four proposals to build glass reinforced plastic aircraft have been reviewed. These were submitted by the following companies:

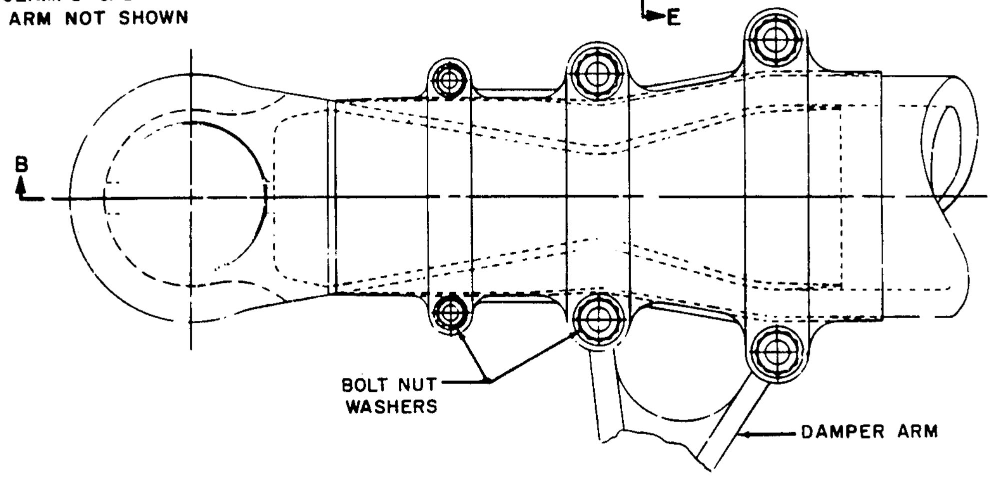
- General Dynamics / Convair, for a COIN type (117)
- Goodyear Aerospace, for a COIN type (136)
- Lockhead-California, for a COIN type (174)
- North American - Columbus, for a YAT-28E (226)

A detailed evaluation of these proposals is not attempted. Considerations are directed simply to design concepts as they represent effective use of the composite materials, particularly the wing, the methods of joining and proposed fabrication methods.

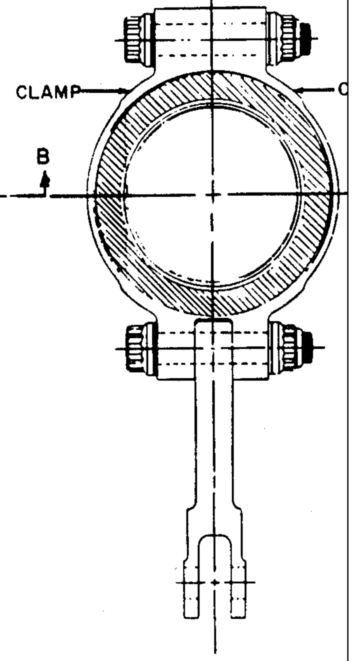
It is apparent that for the most part Convair, Lockhead and North American designs follow the classical concepts for primary aircraft structures. With the exception of the airfoil components, supporting elements such as spars and ribs closely resemble metal counterparts in their arrangements. They are designed, as with metals, to take loads in edgewise compression and bending. The skins which complete the airfoil are similar to rib stiffened fairings. Some stringers are eliminated by the use of rib stiffened sandwiches. The major improvement appears to be that the larger skin panels require fewer sub-assemblies in the overall structure.

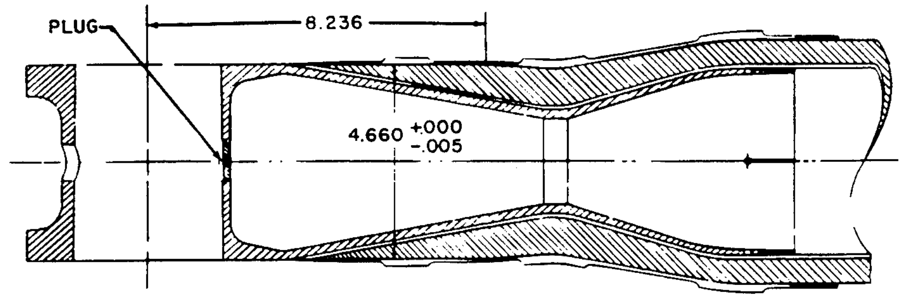
①





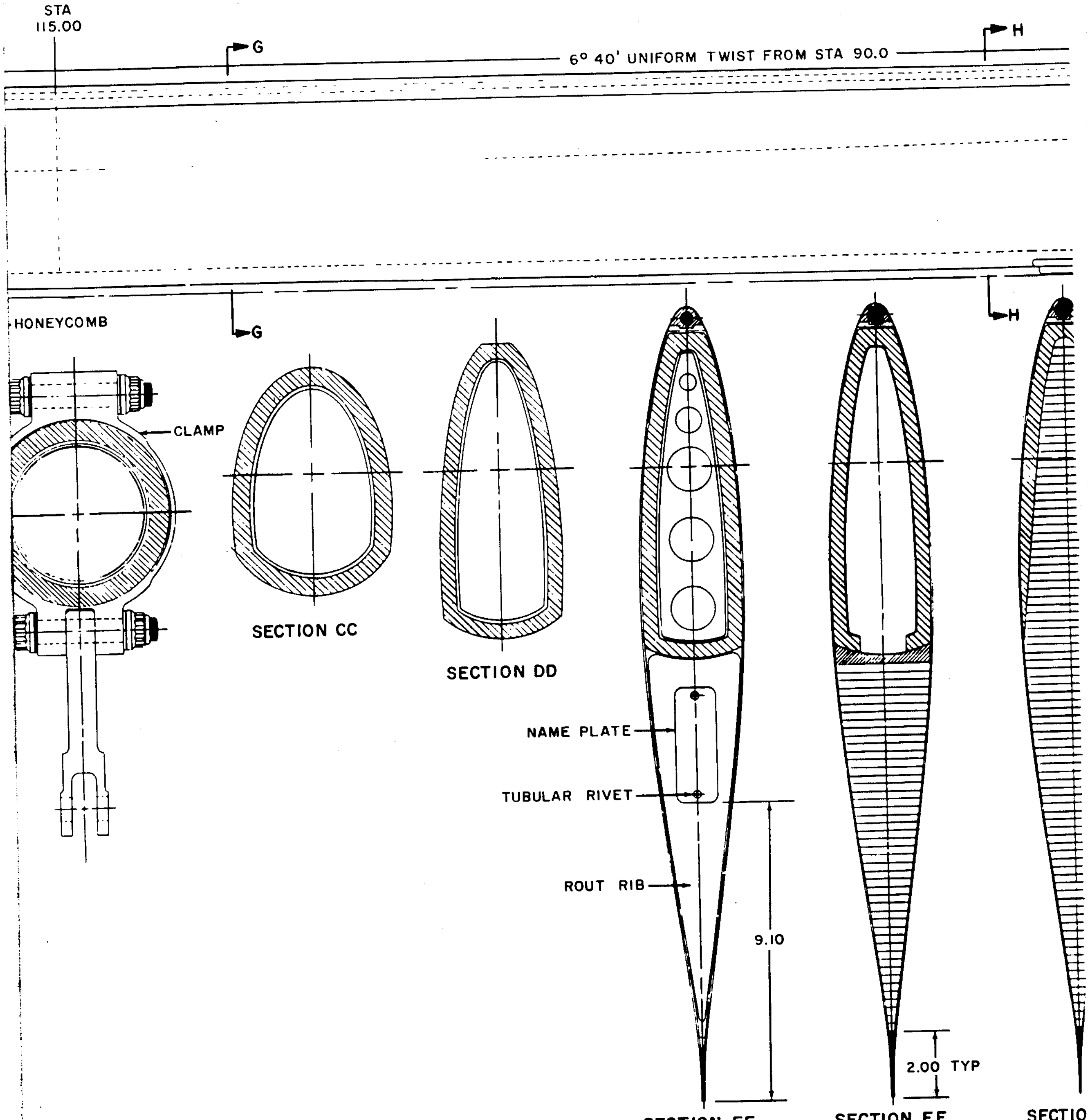
DETAIL A



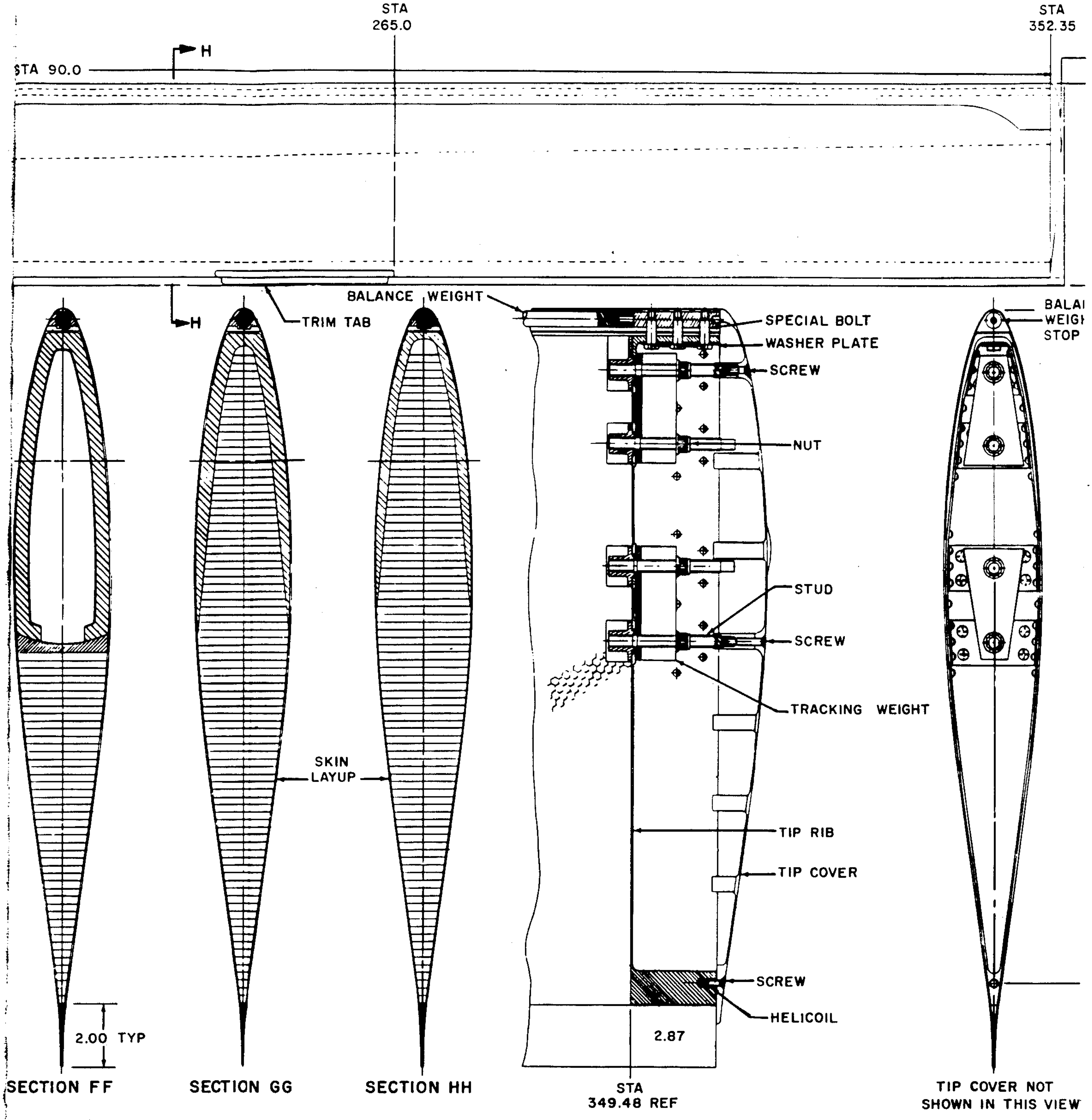


SECTION BB

②


STA
115.00
G
6° 40' UNIFORM TWIST FROM STA 90.0
H
HONEYCOMB
G
H
CLAMP
SECTION CC
SECTION DD
NAME PLATE
TUBULAR RIVET
ROUT RIB
9.10
2.00 TYP
SECTION EE
SECTION FF
SECTIO

3


STA 265.0
STA 352.35
STA 90.0
H
H
TRIM TAB
BALANCE WEIGHT
SPECIAL BOLT
WASHER PLATE
SCREW
NUT
STUD
SCREW
TRACKING WEIGHT
SKIN LAYUP
TIP RIB
TIP COVER
SCREW
HELICOIL
2.00 TYP
2.87
BALA
WEIG
STOP
SECTION FF
SECTION GG
SECTION HH
STA 349.48 REF
TIP COVER NOT SHOWN IN THIS VIEW

4

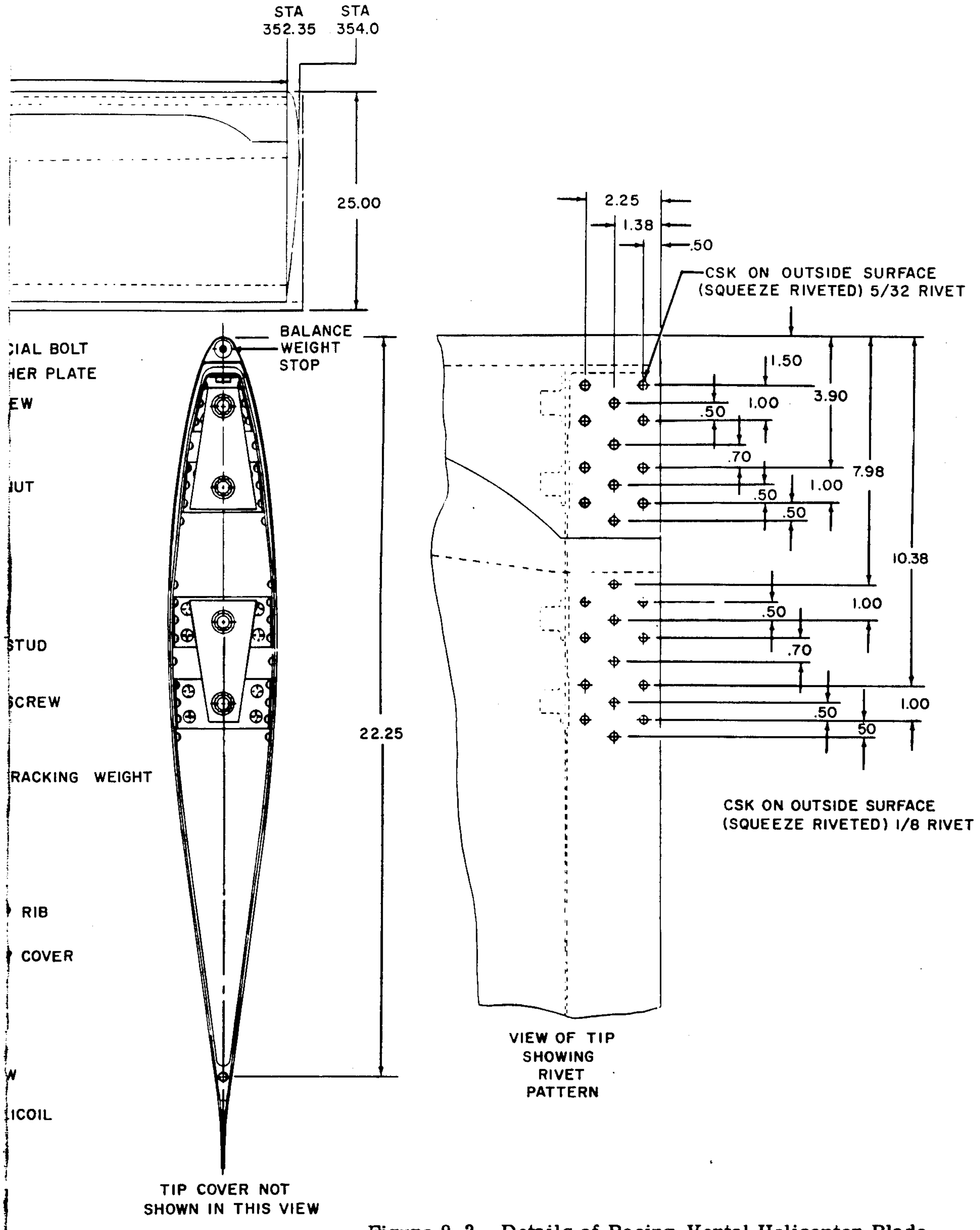


Figure 8-3. Details of Boeing-Vertol Helicopter Blade

Skin panels, ribs and spars are fiberglas sandwiches throughout, but it is questionable whether any significant improvement is gained from such rib and spar constructions. Some attempts have been made for more efficient structural use of the composite materials by having unidirectional laminates carry part of the torsional loads, by reinforcing fuselages with unidirectional foam stabilized longerons and by self-stabilized filament wound sandwich structures for the booms or aft fuselage.

Essentially, however, the proposals are close approximations of metal designs.

The Goodyear design for a wing or other airfoil shape is an exception, and it departs from the traditional box structures. The wing itself forms an integral bonded assembly composed of spar-like trusses and stressed sandwich skins. The entire wing box is designed to resist primary loads. The truss bracing is connected to the skin panels in a continuous fiberglas splice. At local stress points such as fuselage or engine pylon attachements, hard points are developed and are reinforced with metal inserts and added glass plies. The trusses distribute hard point loads away from the wing center sections. A construction method has been proposed which can lead to a single molded assembly, thus taking advantage of the formability of fiberglas. The molding procedure is also adaptable to forming of tail surfaces, stabilizers and flaps. After molding, only the addition of fittings, hinges or similar hardware is required to complete the assembly. The sequence of the lay-ups leading to the final molding is shown in Figure 8-4. Both vacuum and internal pressure can be applied during cure. Details of some joint designs in the wing and fuselage are illustrated in Figure 8-5.

Convair indicates that filament winding may be used for the two booms in their Model 48 aircraft. A filament wound sandwich, with modified sewn fluted cores, is expected to eliminate some stiffeners and provide a single-piece assembly. The rest of the airframe structure will consist of monocoque fluted core skins with a laminated fiberglas structure. Close-outs for the fluted core skins, as in a trailing edge, are made with caps of solid laminates. Convair intends to verify the design by replacing assemblies in their metal COIN with fiberglas parts beginning with the vertical stabilizer. Some of their proposed attachment methods are shown in Figure 8-6.

North American has proposed a filament-wound aft fuselage, again as a sandwich. Internal supports will be required only at the tail section, to distribute empennage loads. The wing is made in three major assemblies - a center section and left and right sections. The outboard sections are joined to the center in a continuous splice at a station where loads are relatively light. Threaded fasteners reinforced with metal inserts molded into the facings complete the splice. The top wing skin is a one-piece molding which also forms the leading edge. The forward fuselage is designed with longerons, frames and bulkheads to stabilize the skin panels. S-glass will be used, but only in the highly stressed areas. Most of the core material will be Raypan fluted core. Typical joints are shown in Figure 8-7.

The Lockhead proposed wing is a typical box structure with front and rear spanwise shear webs fabricated from sandwiches. There are 12 ribs, also sandwich, in each wing, which is made in three sections. The wing box is attached to the fuselage through four bathtub-type fittings. Fiberglas materials will be both

E-glass and unidirectional S-glass. Aluminum honeycomb is proposed for all sandwich constructions. Adhesives will be low temperature curing modified epoxies with synthetic fabric carriers. Whittaker-Narmco is listed as a subcontractor for fabrication of the plastic parts.

ADHESIVE BONDED JOINTS

Experimental data for adhesive bonded joints are generally derived under laboratory conditions and are not directly applicable to specific designs. The intended purpose of the data is more to provide designers with qualitative information and comparisons between various adhesive systems. Published design theories, based on this data, provide analytical definitions of the mechanics of bonded joints, or establish means for relating loading effects determined on laboratory specimens to loading effects found in an actual structure.

A recent survey conducted by Forest Products Laboratory for the Air Force gives a comprehensive summary of such developments in design of bonded joints (113). The study was restricted to lap type joints, since it was concluded that these are the most common, and nearly all bonded joints can be simplified to a lap joint for analytical purposes.

Further guidance in design of joints is contained in two translations of Soviet compilations. One considers the problem of joining (169) and the other is related to design problems of sandwich panels (13). Additional information is given in references 8, 10, 216, 243, 285.

This type of information can only be useful in preliminary designs. It is found that joint problems in reality cannot be divorced from overall design considerations for any particular structure. No satisfactory mechanical tests are available for experimental determination of shear or tensile strengths of adhesive joints as they occur in composite structures. Properly designed bench tests must be depended upon to provide design data more closely approximating actual loading conditions. Most companies, therefore, have conducted these bench tests as an integral part of the design for each component. A series of such tests, as run by Boeing-Vertol, are outlined in Section 9.

SUMMARY OF STRUCTURAL DESIGNS

Most of the designs reviewed in the survey are not sufficiently developed to take full advantage of the potentials inherent in the fiberglas composites. The formability and directional properties of these materials offer possibilities for future designs. It is only in those cases where developments have proceeded over a period of years that practical and efficient designs have evolved. Specific reference is made to the Boeing-Vertol and Kaman helicopter rotors.

While concepts derived in the rotor blades may not be directly applicable to wing structures, they do indicate directions for practical solutions. In essence, skins are designed to resist loads, the substructure is an integral construction with the skin panels, designs eliminate many troublesome joining problems and wherever possible fibers are made to resist shear loads in tension.

Similar trends are shown for the propeller blades (see Section 3), although

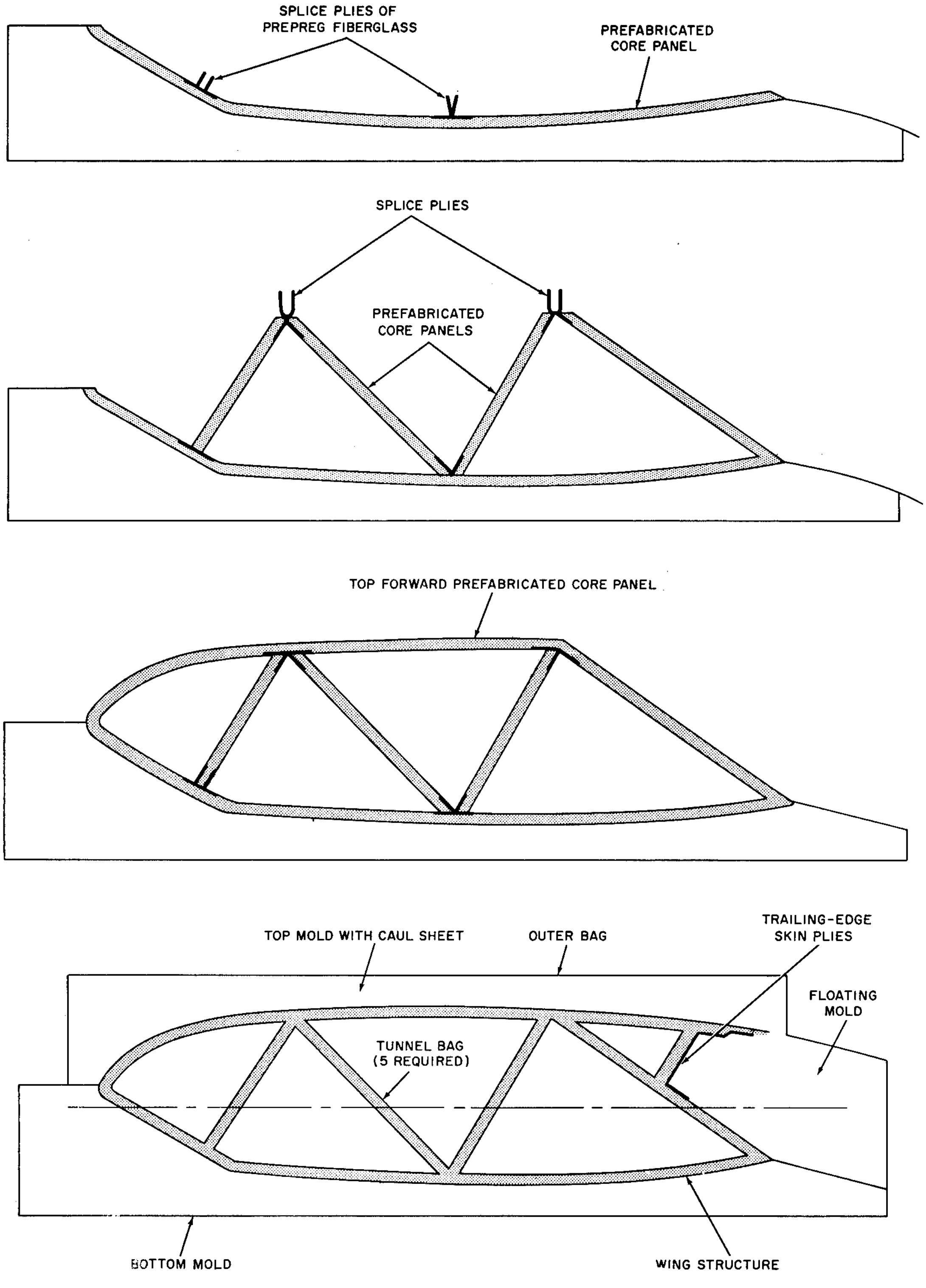


Figure 8-4. Goodyear - Sequence in Wing Layup

propellers are of relatively minor importance in the overall structure. The Goodyear designs for airfoils present an advancement, but are still not optimum constructions.

It is not surprising to find that the plastic aircraft in Convair and Lockhead designs weigh more than their metal counterparts. The Convair metal structures weigh 2, 430 pounds compared to 2, 617 pounds for the plastic. Lockhead estimates 1, 053 pounds for the metal parts as against 1, 109 pounds for the same parts in plastic. North American, in contrast, estimates the total plastic structure to weigh 3, 286 pounds compared to 3, 757 pounds for the metal version.

Certain cases exist where it is expedient to retain conventional box beam concepts. In such instances it may be more practical to design with metal substructures and use the reinforced plastics for skin panels only.

The conclusion is reached that additional developments are required to attain optimum plastic designs. Particular consideration should be given to the wing and to methods of fastening and joining. The means for such developments are presently available and are indicated throughout various sections of this report.

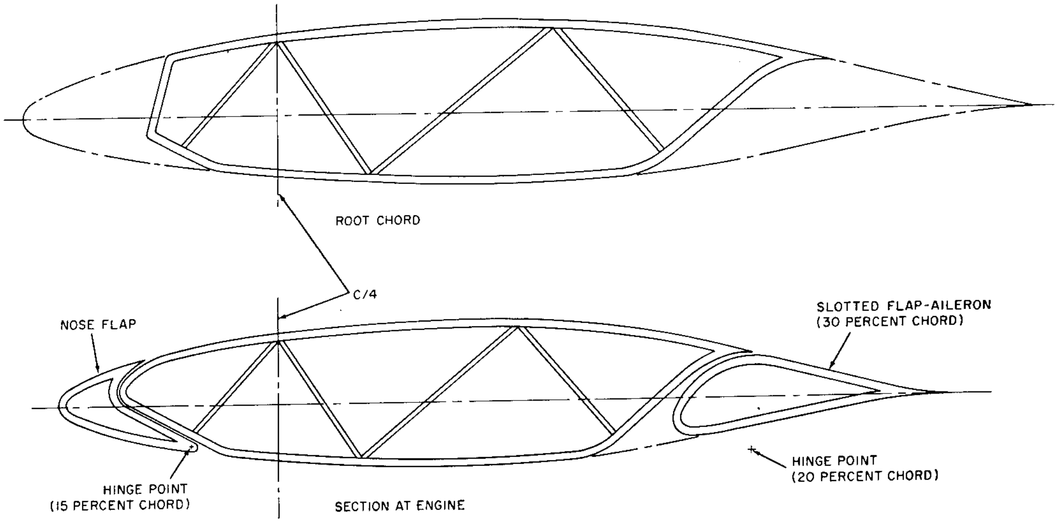
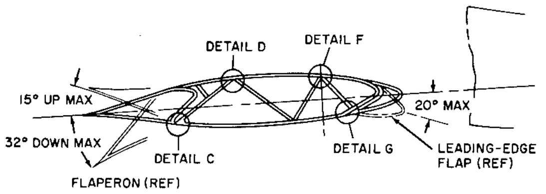
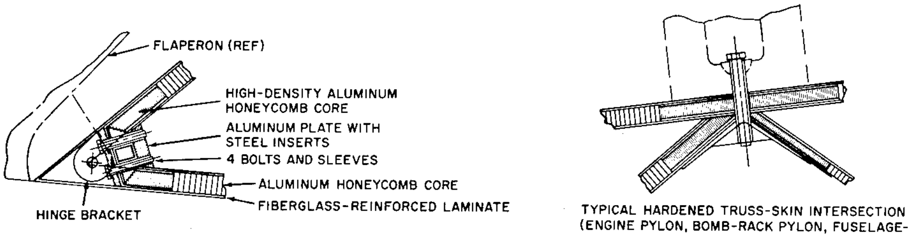


Figure 8-4A. Goodyear - Wing Sections



SECTION AT ENGINE PYLON

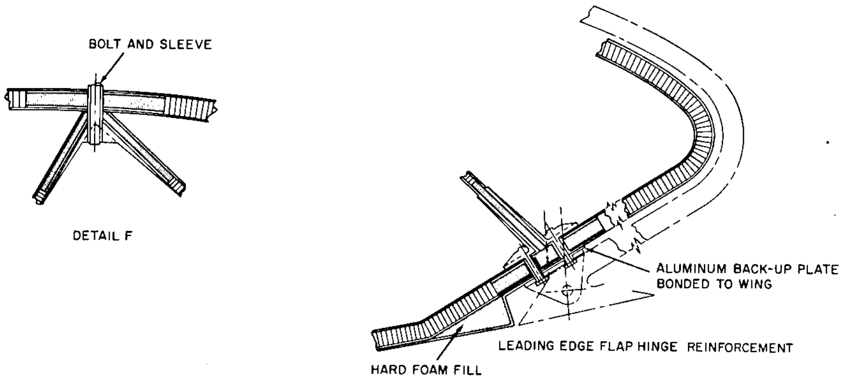


TYPICAL FLAPERON HINGE REINFORCEMENT

DETAIL C

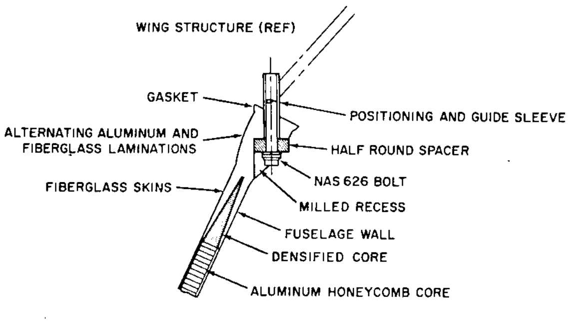
TYPICAL HARDENED TRUSS-SKIN INTERSECTION (ENGINE PYLON, BOMB-RACK PYLON, FUSELAGE-WING ATTACHMENT)

DETAIL D



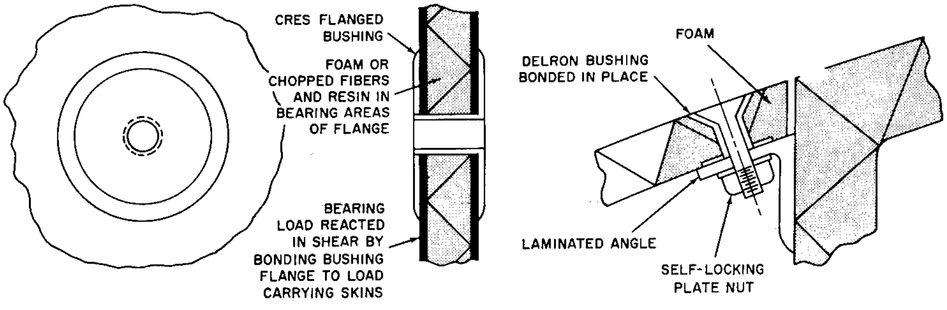
DETAIL F

DETAIL G



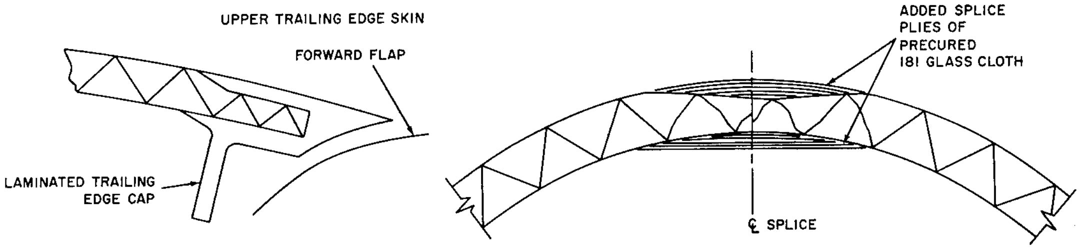
REPRESENTATIVE FUSELAGE ATTACHMENT FITTING

Figure 8-5. Goodyear-Wing Construction and Joints



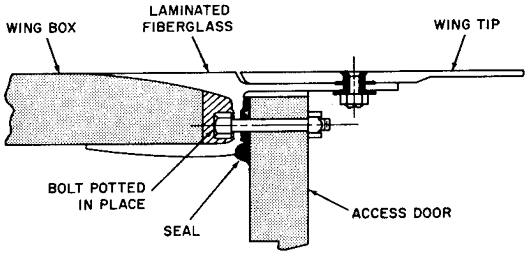
HARD POINT INSTALLATION

TYPICAL LEADING EDGE ATTACHMENT



TYPICAL SPLICE

FIBERGLASS LAMINATED CAP



WING TIP INSTALLATION TO WING BOX

Figure 8-6. Convair - Proposed Joints

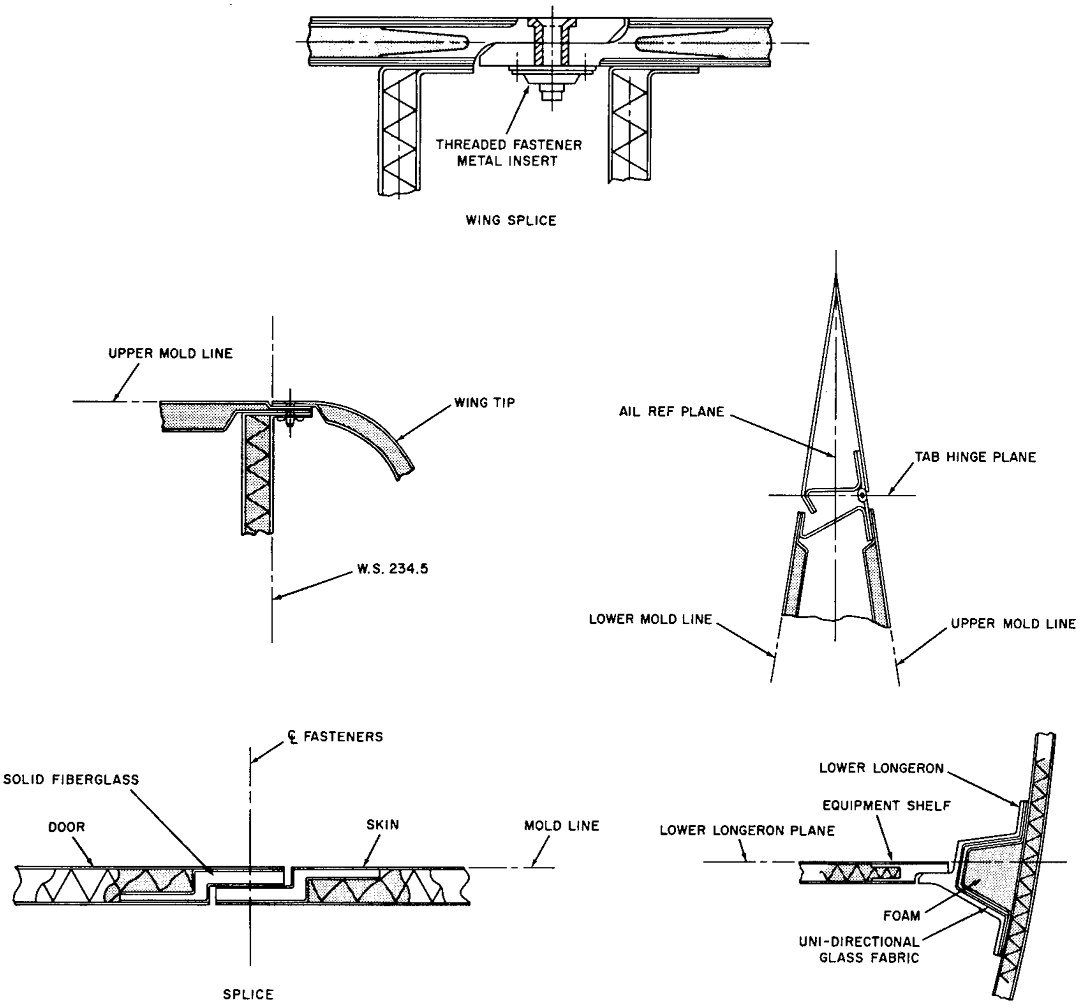


Figure 8-7. North American Aviation - Proposed Joints

SECTION 9.

MANUFACTURE OF FIBERGLAS REINFORCED COMPONENTS

CURRENT MOLDING METHODS

Nearly all reinforced plastics parts are in the form of sandwich, made with honeycomb cores. Continuation of this type of structure can be expected in the future, perhaps with some changes in the core materials. These sandwich constructions can be fabricated in what has been called a one-step process in which prepreg glass fabrics and the core are combined in one molding operation, or they can be formed by bonding premolded facings to the core. To a lesser extent, monolithic shaped laminates with or without stiffeners need to be fabricated.

Secondary operations include bonding of sub-assemblies, attachment of hardware, stabilization of joint areas, and incorporation of foams or potting compounds into the structure. Preparations prior to molding include the machining and shaping of the core and cutting of the glass cloth patterns. Finishing steps such as sanding or grinding are frequently required.

Table 9-1 lists the current molding techniques as developed in the reinforced plastics industry. Of these, vacuum bagging, autoclaving and pressure bagging are used extensively for aircraft constructions. Occasionally straight compression molding is used to form shaped parts or relatively flat laminates. Filament winding has been applied only to radome housings or ducting.

The autoclave, vacuum or pressure bag methods are hand operations. As such they can be expected to produce non-uniform parts and are not readily adapted to automation. They have, however, been successfully applied in the production of aircraft components for a number of reasons. Usually the parts being made such as leading and trailing edges, ailerons, canopies and rotor blades would be difficult to manufacture as metals without expensive tooling or an increase in the number of sub-assemblies. At the current low production rates, simple shaped skin panels are more economical because of lower tooling costs, easier closeout of the cores and less riveting. Reliability has been achieved to a certain extent by close integration of the structural design, materials selection, tooling design and quality control with the actual manufacturing process. Strict attention to details and in-process controls, particularly in relation to pressure and temperature cycles during cure, tend to reduce operation error.

The hand lay-up molding methods and related operations have been subjected to considerable criticism and in many instances, justifiably so. Criticism is usually directed to what can be described as general shop practice, including such factors as environmental control, cleanliness, materials storage and handling, supervision, operator training and other intangibles. Shop conditions vary greatly from one establishment to another. It has been found, however, that those companies which make many reinforced parts and have exhibited design and manufacturing capability with these materials have also set up well organized molding shops which employ assembly line methods as closely as possible. It is also noted that where production may be only two or three aircraft a month, each having as many as 400 to 500 parts in addition to replacement parts for other aircraft, it is difficult to see how these operations could be automated whether the construction is in metal or plastic.

Table 9-1. Fabrication Methods for Reinforced Plastics, Compared

Method	Fiberglas form	Glass content % by wt.	Relative Comparison		
			Strength	Tool cost	Labor
Spray-up	Chopped roving	25-30	Low	Low	High
Hand lay-up:					
Contact pressure	Cloth or mat	30-45	Low	Low	High
Vacuum bag	Cloth or mat	45-60	Medium	Low	High
Pressure bag	Cloth or mat	50-65	High	Low	High
Autoclave	Cloth or mat	50-65	High	Medium	High
Hydroclave	Cloth or mat	50-65	High	High	High
Vacuum impregnation	Cloth or mat	30-45	Low	Low	High
Compression molding	Cloth or mat	50-75	High	High	Low
Matched die molding	Premix + or chopped roving	25-40	Low	High	Low
Centrifugal casting	Mat or chopped roving	25-35	Low	High	Low
Filament winding	Roving	65-85	High	High	Low
Tape winding	Woven tapes, oriented tapes	50-65	High	High	Low
Cable clave	Woven tapes or roving	50-70	High	Medium	Low
Continuous sheet lamination	Cloth or mat	20-30	Low	High	Low
Pultrusion	Roving	55-70	High	High	Low

A number of steps have been taken to improve the hand lay-up methods. In the first place, widespread use of prepreg materials represents a definite advance. Prepregs allow closer control of resin content and resin flow, and they prevent excessive resin build-up within the core cells.

The one-step process for manufacture of sandwich structures has been developed at Boeing Company and Goodyear Aircraft Corporation. Essentially it is aimed at lowering costs by reducing the number of steps in the molding operation. In this process the sandwich assembly is pressurized by autoclaving or bagging within a polished female mold. A smooth aerodynamic finish is obtained on the surface of the part next to the mold. In most cases only one such surface is required. There are a few cases, as in a rotor blade, where smooth exterior surfaces are needed on both faces of the part. The one-step process tends to give a dimpled surface adjacent to the flexible bag. This condition has been alleviated by addition of filler compounds and reworking of the surface. A dimpled sandwich structure is normally considered as crippled. Boeing has claimed, however, that compression tests show the one-step process to yield higher strengths. The explanation for such behavior is that in the two-step process a complete bonding of core to facing is not effected. The cores cannot be machined accurately enough to contact both faces evenly. Even though the high spots on the core are crushed during the molding, the low regions are not contacted. As a result there are unbonded areas, thick glue lines, and areas with no filleting of core cells. In the one-step molding, a more complete bonding takes place with good filleting and improved cell stabilization. These latter effects more than compensate for whatever dimpling occurs. Comparative Boeing data is shown in Table 9-2. Goodyear, on the other hand, has not been too successful with the one step process when metal inserts had to be incorporated into the structure. Comparisons made at the University of Oklahoma Research Institute indicated that the two-step process gave greater strengths. These tests, it is noted, were made on flat press cured samples where good core bonds can be achieved. At any rate, the one-step process appears to have sufficient merit and warrants further development.

The University of Oklahoma Research Institute, under contract with TRECOM, is also working on improved processes for sandwich structures. One phase of the program is the development of a method for impregnating three layers of glass fabric simultaneously. The three layers are formed into a prepreg which is then handled as a single ply. It is hoped in this way to obtain more uniform resin distribution and better control of resin content. The prepreg is precured in press platens prior to bonding to the honeycomb core. Facings molded from these prepregs are claimed to have low void contents. So far the impregnating has been performed with style 181 cloth. Future plans call for prepreging of some high modulus weaves.

Table 9-2. Core Shear Stress in Short Beam Flexure (27)

Process	Shear Strength, Psi	
	Initial	21 days at 125°F/100% RH
Two-step	377	146
One-step	710	540

Whittaker-Narmco has indicated a preference for a pressure bag technique which is similar to compression molding. A flexible pressure bag is attached to a press platen and is operated as a male plunger. It is claimed that this procedure can give accurate control of the temperature and pressure during curing cycle and does away with the need for a large autoclave. This method, and compression molding in general, would seem to be applicable to the molding of smaller parts or panels. It could result in closer tolerances, and it is more amenable to automation. For larger parts, multi-ram presses might be required and it is not likely to be an economical process unless improved short curing cycles were adapted.

NEW OR PROPOSED FABRICATION METHODS

Filament winding has been suggested as a process for fabricating the aft fuselage and the wing. With the present winding machines, this process appears to offer possibilities for the aft fuselage, but would require machine modifications for the winding of optimized wings. Filament winding, of all processes for FRP, is best suited for automation. In-process and quality control measures have also been more closely defined. Winding of sandwich constructions, however, is a relatively rare operation. It has been done for radomes and cylindrical missile containers. Development programs are now in progress for winding thick walled, ring stiffened sandwiches for deep submersibles. The winding of sandwich-type missile cases has been the subject of an earlier feasibility study. In winding radomes it has been found necessary to apply a supported film adhesive between the sandwich and facings to insure adequate bonding and prevent excess resin migration to the core cells. The as-wound exterior surface will not be aerodynamically clean and may require either bagging or machining. The problem in the filament winding of a wing is the control of machine motions to obtain directional properties while preventing fiber buildups in certain regions. Box-like structures should be readily wound, if the machine is provided with tension controls for the flat surfaces.

Present commercial processes for making continuous panels are not applicable to aircraft structures. A variation is now under development at Narmco for making continuous sheet from rovings. So far it is confined to unidirectional flat laminates. To be useful it would be necessary to adapt the method to cross-plied curved panels.

Aerojet-General is investigating a continuous pultrusion technique for manufacture of I-beam, T-beam or similar structural reinforcing shapes. Fiber orientation is directed to meet specific stress patterns. Beams can be designed to resist interlaminar shear loads by putting in cross fibers, diagonal filaments or short fibers in the web. In this process glass strands are passed through an impregnation bath, through a forming die, and then to an oven for continuous curing. A braiding machine permits orientations other than unidirectional. The process is limited to constant cross sections, but can be enlarged to fabricate box beams or channels.

EFFECTS OF PROCESS VARIABLES

A few developments have been undertaken to ascertain the effects of processing on finished properties. These studies, which are based on coupon type testing, have not been related closely enough to present processes and materials to be of value. Such factors have been evaluated as the effects of voids and unbonded areas on sandwich strength, effects of moisture penetration, and strengths versus thickness, resin content and cure conditions.

This data is more of an exploratory nature. It shows general trends, as in Figure 9-1, which gives variation in strength versus resin content for polyester/ glass cloth laminates. Studies of filament winding variables have been more fruitful. Effects of winding tension, variation in winding patterns, resin content, type of roving or prepreg, and repairs have been more closely delineated.

TOOLING FOR MOLDING OR BONDING SANDWICH ASSEMBLIES

Typical molds for autoclaving, vacuum bagging, and pressure bagging of sandwich constructions are shown in references 55, 119, 136, 174, 226, 234 and 247. These are usually female molds in which the exterior surface is placed next to the mold surface. Clam shell or positive pressure type molds are used less frequently. Here the core and facings to be bonded or molded are confined to a fixed space. When parts are so confined, tight dimensional tolerances can be held, but at the expense of the crushing of the core or of getting poorer bonds from non-uniform pressure distribution. The molds used for bagging and autoclaving appear to be more advantageous for the following reasons:

- More uniform adhesive bonds are obtained because of more uniform pressure.
- Trapped air or other volatiles are more easily removed.
- Tooling costs are usually lower.
- Damage to the core can be more easily avoided
- Structural properties are more consistent
- In most instances the overall sandwich thickness is not critical, so that sandwiches can be molded with maximum allowable tolerances and still maintain an aerodynamic surface.

The major disadvantage is that heat must be transferred from one side only, so that longer times are required to obtain optimum cures.

In long pieces, such as a wing or rotor, differences in thermal expansion present problems. Where there is a variable pitch or varying cross section of the part, the problem becomes more critical. Two solutions for such situations have been proposed. The first, proposed by Goodyear and others, is to use epoxy rib stiffened tooling so that there is no mismatch in thermal expansion. In the second instance, Boeing proposes a heated liner which will be designed to expand at the same rate as the part. It will be backed by a cooled base which will not expand away from the liner.

The molding of sewn or fluted cores also seems to present difficulties. The questions are: how to support the flutes so that more than 15 psi can be applied, and how to transfer heat to the center of the cores. Present solutions, where either permanent foam mandrels or removable metal mandrels are used, do not appear adequate for optimum curing conditions.

Molds have been fabricated from a variety of materials. Glass cloth/epoxy molds have been found satisfactory for short duration runs. When low pressures are applied, their service life can be extended. Grumman has had successful experiences with electro-deposited nickel molds. These are readily formed from wooden or plastic masters. Aluminum tooling has been used extensively where excessive wear is not anticipated. For long run permanent production tooling, normal tool steels are still preferred.

There have been no government-sponsored programs to investigate tooling applicable to aircraft parts. The one related program, sponsored by the Air Force and conducted by Rocketdyne, studied designs and materials for filament winding mandrels. It appears that additional work in this direction would be profitable in evaluating the various materials, effects of mold surface condition, methods for improving heat transfer, maintaining proper thermal expansions, and for molding unidirectional materials without disturbing fiber alignment.

RELIABILITY AND QUALITY CONTROL

Quality control procedures, as established by aircraft manufacturers, include:

- Qualification and batch acceptance tests for all raw materials: resins, catalysts, hardeners, glass fabrics, prepregs, adhesives, fillers, honeycomb cores, foams, potting compounds, etc.
- Qualification and acceptance tests for all non-productive materials: parting compounds, bagging materials, cleaning solutions, solvents, peel plies, etc.
- In-process controls: machining of cores, lay-up sequences, application of release agents, methods of bagging and maintaining pressure and vacuum, temperature and pressure cycles during cure, post-cure and cooling cycles, etc.
- Inspection procedures and nondestructive test methods.

These procedures have been found to be adequate for parts now being fabricated, particularly in cases where strengths and weights are not critical. Examples are listed in references 30, 36 and 38.

When it comes to optimized primary structures, as the all-plastic wing or fuselage, the situation is different and the establishment of adequate reliability controls will be a major problem. Here, restrictions are imposed not only by higher design allowables but by the fact that overall weight must be controlled to within ± 1 percent. Present radar applications, it is noted, are held to approximately ± 2 percent.

Reliability, it can be expected, will begin with the drawing of glass filaments. (See Section 5.) Aircraft manufacturers have had little concern, up to the present, with the production of glass fibers. Their main interest has been with the woven cloth. High grade glass filaments in low end counts, such as S-HTS or E-801 strands, show a fiber diameter variation of ± 5 percent and a weight per yard variation of about ± 2 percent. What effect this will have on finished molded products is not known, but it may be anticipated that tighter controls on fiberglas strands and filaments will be necessary.

Variability in woven fabrics as to weight, thickness and amount and distribution of finishes is relatively high. It is somewhat lower in the non-woven unidirectional cloths. An investigation was conducted a few years ago by DeBell and Richardson, under Navy contract, of the effects of fabric variations on mechanical properties. The study was also related to the adequacy of military specifications and general practices within the weaving mills. Two yarns were selected for weaving - one relatively even in weight having a coefficient of variation of 2.06 percent, and the other a coefficient of 5.01 percent. Cloths were woven at three mills with these yarns and comparisons were made with run-of-the-mill fabrics. Results as given in Table 9-3, showed weights to be within allowable government specifications of from 8 to 10 ounces per yard. Thickness tended to exceed the upper specification limits of from 8 to 12 mils. The controlled low variation yarn gave some improvement. Results on material properties were less conclusive. It was also concluded that humidity, temperature and heat cleaning introduced variations. Military specifications related to fabrics for laminates are listed in Appendix A. Allowable variations are considered too wide for effective quality control.

Problems of reliability with resin systems are not as critical, and the present control tests seem to be adequate. The prepregs, however, will require controls, based on methods as developed for the glass filaments and woven cloths, combined with resin controls and additional controls related to storage and handling. At present no government specification exists for either prepreg cloth or roving, although individual companies have set their own standards. Standards for the aluminum honeycomb cores are well established from practices with metals. The fiberglas cores, however, show large variations as to weight, glass content, and shear strengths.

ASTM coupon type tests are widely used for material qualification, batch acceptance, and quality control. To a lesser extent they are useful for in-process controls. Here, bench tests which closely simulate designs and actual manufacturing procedures are of greater value. These are particularly helpful in evaluating joints, sandwich bonds and strengths of panels. Typical bench tests of structures for hanger joints, as designed by Boeing-Vertol, are illustrated and commented upon in Appendix E. These tests, for example, show a range of values from a high of 3160 pounds ultimate load to a low of 1940 pounds ultimate for variations in methods of attachment. The regions and modes of failure are indicated. The tests provide a means for obtaining preliminary design allowables and for working out process details. The method for loading these panels is illustrated in Figure 9-2.

NONDESTRUCTIVE TESTING

Nondestructive test methods for plastics structures have advanced considerably in the past few years. This has been due to the development of quality control systems in the missile industry. There still is a need for standardization of nondestructive test methods and interpretation of test results. At the present time, ASTM is starting to classify defects in glass reinforced plastics, and is studying

Table 9-3. Variations in Weight, Style 181 Cloth; Comparison of Output from Three Mills*

Mill A Weight	Mean Average $\overline{X}$ oz/sq yd	σ oz/sq yd	C_V %
C-P Lot A Yarns	8.706	0.0768	0.88
C-P Lot B Yarns	8.62	0.091	1.05
ROM	8.608	0.1075	1.25
Thickness	mils	mils	%
C-P Lot A Yarns	13.157	0.307	2.33
C-P Lot B Yarns	12.99	0.390	3.00
ROM	12.104	0.463	3.83

Mill B Weight	Mean Average $\overline{X}$ oz/sq yd	σ oz/sq yd	C_V %
C-P Lot A Yarns	8.878	0.050	0.57
C-P Lot B Yarns	8.796	0.0785	0.90
ROM*	8.483	0.135	1.59
Thickness	mils	mils	%
C-P Lot A Yarns	12.242	0.400	3.27
C-P Lot B Yarns	12.359	0.462	3.73
ROM*	12.083	0.731	6.0

Mill C Weight	Mean Average $\overline{X}$ oz/sq yd	σ oz/sq yd	C_V %
C-P Lot A Yarns	8.879	0.073	0.83
C-P Lot B Yarns	8.85	0.088	1.00
ROM	8.562	0.268	3.12
Thickness	mils	mils	%
C-P Lot A Yarns	12.095	0.376	3.11
C-P Lot B Yarns	11.850	0.416	3.55
ROM	11.69	0.390	3.33

---

*Includes 181 from 150 1/2 yarn

Notes:

Source: Eakins, Fourteenth RPD, SPI.

CP - Lot A, Controlled Process, $C_V \pm 2.06\%$

CP - Lot B, Controlled Process, $C_V \pm 5.01\%$

ROM - Run of Mill

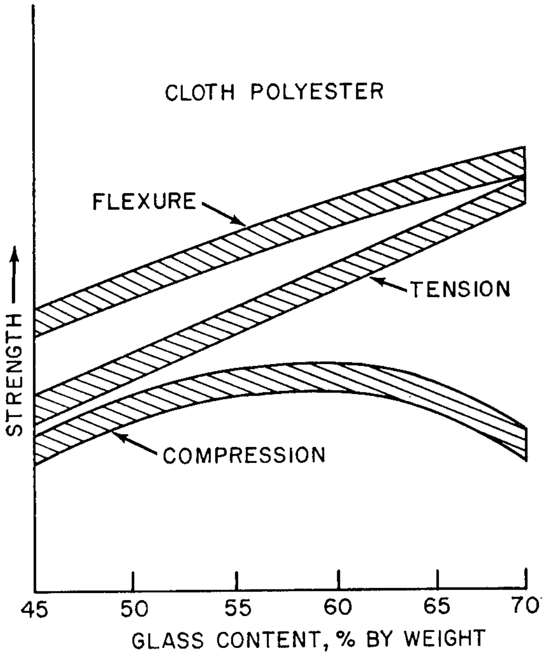


Figure 9-1. Strength vs. Glass Content, Polyester Laminates

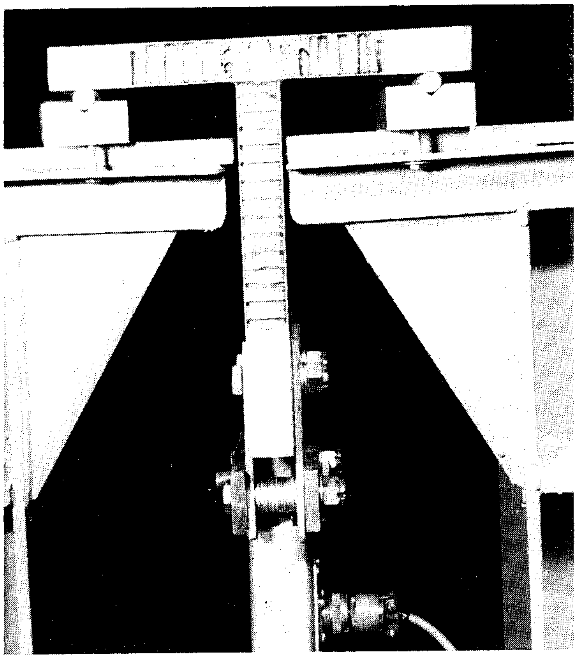


Figure 9-2. Bench Test for Sandwich Panel (Boeing-Vertol)

test methods suitable for various types of defects and structures. Up to the present, test standards have been developed by companies for their own use. These are based on service tests to determine the performance when defects of a certain magnitude are present, and to determine allowable defects. Significant defects that may be present in reinforced plastics structures are listed in Table 9-4.

For aircraft structures, the ultrasonic methods appear to be the most suitable. Radiographic methods have limitations, since both sides of the item tested must be accessible, and these tests are more expensive. In contrast, ultrasonic techniques are generally considered favorable for in-plant and on-site testing of plastics structures. Ultrasonic instrumentation is generally less expensive, is portable, and can locate and evaluate various types of defects. Quantitative prediction of mechanical properties, such as elastic modulus, tensile, shear, compressive strength, and density can be related to ultrasonic responses of plastic materials. The ultrasonic methods are:

- Pulse Echo Method - Transducer only on one side; - a favorable method for plastics.
- Through Transmission Method - Usually a transducer on each side of test part.
- Resonance Method - Usually used to measure thickness; one transducer on the outside is required.
- Frequency Modulation Method - Similar to pulse-echo method except continuous waves with periodically altered frequencies are used. This is a relatively new and experimental method presently being evaluated by certain missile contractors.
- Fokker Bond Tester, Stub Meter, and Coinda-Scope Methods - Usually used to determine bond strength. These instruments generally indicate only no-bond or some-bond, but not the exact degree of bond. The Fokker tester is frequently used in the aircraft industry.

The Porta-Shear tester provides a means for making direct shear strength measurements. In this test a quarter-inch sample is cut in the facing of a sandwich, and tested in place. Although it is not nondestructive testing, the hole in the skin is easily patched. The tester is presently being investigated by General Dynamics (122).

Absorption of moisture can be a serious problem, as attested by failures in the B-52 chin radome (39). Test methods to determine moisture absorption by dielectric means have been developed. These methods show promise, but would require additional study to be applied to aircraft structures.

As a general conclusion regarding NDT in aircraft structures, it can be stated that present methods for reinforced plastics are adequate for production controls. Means for further improvements are indicated. The application of NDT and other test methods for periodic field tests needs to be developed.

Table 9-4. Common Defects in Reinforced Plastics, and NDT Methods for Detection

Defect	Definition	Applicable NDT Methods
Delamination	Separation of layers in a laminate	Ultrasonics, Sonics, Radiography, Microwave, Corona discharge, Infrared, Manual tapping, Visual inspection (for transparent plastics only), Dielectrics.
Unbond	Poor or no adhesion between two adjacent surfaces.	Similar to above.
Foreign objects	Metallic or nonmetallic inclusions.	Ultrasonics, Radiography, Microwave, Infrared, Candling.
Fracture	Rupture of surface without complete separation of laminate.	Visual inspection.
Air bubble	Air entrapment within and between plies, noninterconnected.	Ultrasonics, Radiography, Microwave, Corona discharge, Infrared, Candling.
Blister	Rounded elevation of the plastic surface.	Visual inspection.
Burnt area	Thermal decomposition and discoloration of surface area.	Visual inspection.
Orange peel	Surface roughness	Visual inspection.
Pitting	Small crater on the surface of the plastic.	Visual inspection.
Porosity	Presence of numerous pits.	Ultrasonics, Radiography, Microwaves, Corona discharge, Infrared, Candling.
Shrink mark or sink	Dimple-like depression on the surface.	Visual inspection.
Disorientation of fibers	Area where the reinforcement has moved.	Radiography, including tracer methods.
Wrinkles	Surface imperfections.	Visual inspection.
Short	Incompletely filled out.	Visual inspection.
Thickness variations	Change in thickness.	Ultrasonics, Radiography, Sonic, Eddy current, Radiation detection devices, Microwaves.
Moisture	Absorption of moisture.	Dielectric
Degree of cure	Cure stage of resin and catalyst.	Ultrasonics, Sonics, Microwave, Electrical resistivity.
Surface crazing	Fine surface cracks.	Penetrants, Filtered particle, Electrified particle, Visual inspection.
Poor surface finish	Lack of smoothness.	Visual inspection, or Profilemeter.
Resin variations	Resin-rich or resin-poor areas.	X-Ray or Beta-Ray back scatter.
Glass to resin ratio	Ratio of resin to glass reinforcement.	Beta-Ray back scatter.
Internal stresses	Stresses due to shrinkage or applied forces.	Strain gages, Nuclear spin Resonance, Polarized light.
Surface stresses	Stresses on surface layer only.	Brittle coating, Photoelastic coatings, Strain gages.
Surface hardness	Resistance to abrasion or penetration.	Penetration or Barcol

SECTION 10. COST EFFECTIVENESS

A major advantage claimed for fiber reinforced plastics is that lower costs can be expected than for similar metal parts. Specific figures to substantiate this claim were not obtained. Estimates of the cost effectiveness of FRP were based on the possible reduction in number of parts for subassemblies (by fabrication of multiple parts in one operation), and by lower tooling costs (through fabrication of panels which require minimum machining). Such cost effectiveness depends on the number of parts being fabricated. It appears that, for a relatively small number of parts or single prototype parts, the tooling for FRP is considerably cheaper and reproduction tools can be built in a very short time. As the number of parts increase, a point will be reached at which tooling costs for metal will be cheaper - that is, where high rate processes such as punching and stamping can be used. However, as the number of parts is further increased, more elaborate tooling for FRP will again be competitive with metals.

To effect a tie-in with the subject of this report, it would be necessary to report a detailed cost analysis for a FRP aircraft. This will not be attempted here. However, two reports on cost estimating of aircraft manufacture have been studied, and are briefly commented on.

In their cost estimating techniques, the Rand Corporation uses a series of steps based on statistical evaluations of previously determined aircraft costs (252). They use data from bombers, fighters, cargo ships and trainers. These are not specified, but are in the sonic and subsonic range. Costs are broken down into airframe, propulsion, and electronics, for which the former only would be of concern in the context of this study. Estimating is further broken down into: direct labor, materials, overhead, subcontracting, engineering, tooling, and general and administrative expenses.

The Rand work gives three statistical regression equations: for direct labor costs, for engineering costs, and for tooling costs. These are given as reported:

Statistical regression equation for direct labor cost is:

$$\log X_1 = -0.93496 + 0.64350 \log X_2 + 0.77811 \log X_3$$

where:

X_1 = unit direct man-hour cost (thousands of man-hours for 100th unit)

X_2 = aircraft maximum speed (in knots)

X_3 = airframe weight (in thousands of pounds)

Statistical regression equation for engineering cost is:

$$\log X_4 = -4.35530 + 1.74831 \log X_2 + 0.83263 \log X_3$$

where:

X_4 = total engineering cost including testing and flight testing (in millions of 1961 dollars) for approximately first 100 units

X_2 = aircraft maximum speed (in knots)

X_3 = airframe weight (in thousands of pounds)

Statistical regression equation for tooling cost is:

$$\log X_5 = -2.78057 + 1.09854 \log X_2 + 0.99700 \log X_3$$

where:

X_5 = total tooling cost (in millions of 1961 dollars) for approximately first 100 units

X_2 = aircraft maximum speed (in knots)

X_3 = airframe weight (in thousands of pounds)

Rand cautions, however, that the cost estimating relationships presented are based on the cost data available at that time. As more data become available, the estimating relationships should be reviewed for their validity and usefulness. No matter how valid the data might be, however, Rand suggests that at most the estimates made with their techniques should be no more than points of departure for a more detailed cost analysis of the airframe. The following are particularly pointed out:

- Cost estimating relationships describe the interrelation between two or more variables within the limited range of the data. Extrapolation must be done with considerable care since the interrelation may not be valid outside the observed range.
- It was necessary to assume a higher degree of homogeneity than is actually the case, in order to use the data available for this study.
- One must make sure the construction and materials of the airframe to be estimated are similar to those of the airframes on which the cost estimating relationship is based. The weight and general purpose of the two items, for example, might be exactly the same, yet the performance of one might be considerably better than that of the other. Thus, the improved item might cost considerably more.
- It is preferable to have the predicted costs on the high side, because costs were predicted with the actual characteristics of the first lot of airframes. Most of the variables tend to increase during the development of the aircraft.
- Since overhead is almost always regarded as being a linear homogeneous function of direct labor costs, a considerable oversimplification is involved.
- Tooling and engineering costs have gone up over time as a per cent of manufacturing costs.
- A final point to keep in mind is that materials are estimated in dollars rather than in some unit such as man-hours, which is constant over the years.

According to Rand, weight appears to explain more variations in cost than any other physical characteristic. If a component's weight is doubled, its cost is increased but by a somewhat lower amount. Also, the effect of quantity is considered without exception in the cost estimating of all aircraft subsystems. The airframe generally achieves the greatest cost reduction on increase in quantity; on the average of about a 20 per cent reduction of cost with every doubling of quantity. Involved here is the "learning curve" effect, which is a decrease in recurring production costs as more units are produced.

It is customary in airframe estimating to make a distinction between those costs that recur and those that do not. Tooling costs for the first 100 units should be considered two-thirds nonrecurring and one-third recurring. Engineering change costs are nonrecurring, but are a percentage of the recurring production costs. Development engineering is estimated as a percentage of the total engineering cost for the first 100 aircraft.

Raw materials, hardware, and purchased parts are the three main segments of the materials cost of an airframe. For most subsonic airframes of the type produced in the past, purchased parts amounted to around two-thirds of the cost; raw materials and hardware accounted for about one-sixth each. Initial engineering is a one-time cost that includes basic engineering design and development and the shop expenditures incurred in support of the basic engineering, the various engineering tests and flight testing. Initial tooling is nonrecurring and contains the cost required to design and fabricate the original set of tools as well as duplicate tooling necessary to meet peak delivery rates. Sustaining tooling is a recurring cost, which includes improvements, rework and replacements. General and administrative expenses ordinarily amount to 5 to 6 per cent of total airframe cost, excluding fee or profit.

Convair also has made an extensive study of cost estimating (121). They state that the number of fabricated parts in an airframe, the number of parts to be machined, and the quantities and types of materials required are factors which strongly influence the cost of new airplanes. Their report contains empirical curves, formulas, and statistical tabulations for estimating these strategic airframe cost factors. The curves and formulas relate cost factors to preliminary design weights, performance, and dimensional parameters. They caution, however, that their curves were developed from data on aluminum-type airplanes, and should therefore not be applied to trisonic and hypersonic aircraft. The use of their curves in many instances requires an intimate knowledge of reference-point airplanes and of the specific airframes evaluated.

The estimating procedures are based on the experiences in the production of Convair aircraft models F-102A, F-106A, B-58A, 880 and 990. Separate treatments are accorded the complete airframe and each of the following groups: wing, tail, body, nacelle, and air induction system. In each segment, primary concern is for the number of dissimilar parts in the group, the number of pieces, and the per cent of total weight of the parts. For each airplane studied, detailed tabulations are given for material distribution, breakdown by types of manufactured parts, and distribution of machine shop parts into size categories. From the total data, the formulas for cost estimating are derived.

As pertinent to this study of FRP construction in aircraft, the Convair tables were examined for indication of the relative proportion of reinforced construction or components in the various groups described. The results are presented in Table 10 - 1.

TABLE 10 - 1

Percentage Analysis of Reinforced Parts in Some Aircraft (121)

Aircraft	Assembly	Type of Reinforced Material	Per cent of Total Weight	Per cent of Total Dissimilar Parts	Per cent of Total Pieces
Convair					
F-102A	Wing	Aluminum honeycomb core	0.5	0.4	0.4
	Tail	Aluminum honeycomb core	1.2	0.1	0.1
	Tail	Plastic honeycomb core	-	0.5	0.3
F-106A	Tail	Aluminum honeycomb core	4.9	8.5	7.4
	Tail	Plastic honeycomb core	1.3	0.8	1.0
B-58A	Wing	Aluminum honeycomb core	0.8	5.7	4.7
	Wing	Plastic honeycomb core	3.5	13.5	8.8
	Wing	Steel honeycomb core	1.0	1.0	1.4
	Tail	Aluminum honeycomb core	6.9	13.2	12.3
	Body	Aluminum honeycomb core	1.9	8.2	26.5
	Body	Plastic honeycomb core	0.1	1.9	1.2
	Nacelle	Aluminum honeycomb core	0.4	0.7	0.8
	Nacelle	Steel honeycomb core	2.7	2.4	1.5
	Nacelle	Fiberglas sheet	0.1	0.7	0.5
880	Wing	Aluminum honeycomb core	1.4	3.1	1.6
	Tail	Aluminum honeycomb core	0.9	2.2	1.7
	Tail	Plastic honeycomb core	0.1	0.1	-
	Body	Aluminum honeycomb core	1.4	0.6	0.4
990	Air induction system	Aluminum honeycomb core	0.8	0.7	0.5

SECTION 11. GOVERNMENT PROGRAMS RELATED TO FIBERGLAS AIRCRAFT STRUCTURES

AIR FORCE PROGRAMS

The Air Force has been active in the development of glass reinforced composites since the beginnings of the industry. Early programs were related to specific structures such as radomes, prototype wings and aft fuselage. Later trends were to material developments in reinforcements, resins and filament winding. Emphasis has been on high temperature materials for aerospace applications.

The Air Force has been the major support for work at Forest Products Laboratory and is presently engaged in revising MIL Handbooks 17 and 23. A number of contracts have been negotiated with aerospace companies for the purchase and compilation of information on a variety of aircraft materials.

Programs have been sponsored by segments which are now included in the Air Force Materials Laboratory. These are the Nonmetallic Materials Division, Materials Application Division, and the Manufacturing Technology Division. Present interest in reinforced materials for aircraft has extended to the Flight Dynamics Laboratory and the Studies and Analyses Division of the Air Force Systems Command. The programs listed in Table 11-1 have been sponsored by the Air Force Materials Laboratory except for one now being negotiated by the Flight Dynamics Laboratory.

NAVY PROGRAMS

Early Navy programs at the Bureau of Aeronautics also included engineering studies of prototype structures. Later work at the Bureau of Weapons has been a combination of material developments and the development of specific aircraft structures. Typical programs in reinforcements have been with hollow glass fibers, aluminum coated fibers, and silica core sheath fibers. During the Polaris program, the Special Projects Office sponsored many programs in all phases of filament winding. Many of these, though not specific to aircraft, are of particular interest and are included in Table 11-1. Similarly, several Bureau of Ships investigations in connection with deep submergence are added to the Navy list.

Present aircraft programs are centered around work being done at the Naval Air Engineering Center.

ARMY PROGRAMS

The Army programs on aircraft have all been conducted through the Transportation Research Command. Studies related to the use of reinforced plastics were initiated in 1961 under contract to Hayes International Corporation. The objective of this contract was to determine the feasibility of the fiberglas composites in primary structures of Army aircraft, including fixed and rotary wing types.

Present programs are concerned with the fabrication, mechanical properties and analytical procedures for fiberglas sandwich constructions. These will lead to fabrication and testing of full scale structures to validate and optimize the analytical procedures.

The Army has relied on the other services to develop the newer reinforcing and matrix materials, but is emphasizing the need for improved reliability in present materials for aircraft use. Army programs are listed in Table 11-1.

Table 11-1. Current or Recent Programs Related to Aircraft

A. Air Force Programs

Company	Programs
Aerojet-General	Development of improved processes for filament-wound reinforced plastics structures, includes studies of YM31A glass, S-glass and some newer resins.
Aerojet-General	Research to obtain high-strength continuous filaments (type 29A glass).
Brunswick	An evaluation of new and potentially heat resistant glass reinforced plastics. Continuing.
Douglas Aircraft	Ultrasonic techniques and standards for testing filament wound structures.
Lightning & Transients R.I.	Review of LTRI programs on lightning protection. Complete.
Owens-Corning Fiberglas	Glass reinforcements for filament wound composites, a study of manufacturing variables in glass drawing.
Owens-Corning Fiberglas	Sizing system for S-glass compatible with PBI and polyimide resin systems.
Owens-Corning Fiberglas	New high strength, high modulus glass fiber. Target $1 x 10^6$ psi tensile, $18 x 10^6$ psi modulus.
Owens-Corning Fiberglas	Research on high modulus, high temperature fibers. Surface treatment to improve strength, heat resistance and chemical resistance of fibers.
Shell Chemical	Development of epoxy resins to be used with long time high temperature laminates.
Texaco Experiment	Methods for fabricating boron reinforcements for composite materials.
Whittaker-Narmco	Development of polybenzimidazoles high temperature resins and adhesives.
- - - - - - - - - - - -	Exploratory investigation on design and analytical procedures for fibrous composites in aircraft. Now being negotiated. It is believed contract will go to Southwest Research Institute.

Table 11-1 (cont'd)

B. Navy Programs

Company	Program
Aerojet-General	Development of high-strength preimpregnated rovings for filament winding.
Aerojet-General	Development of improved resin systems for filament wound structures.
Curtiss-Wright	Study of X-19 fiberglas propellers. Feasibility and first phase of a fiberglas shank for the X-19 propeller.
Dyna Structures	Study of the methods of structural optimization for flat sandwichpanels. Program is completed.
Grumman Aircraft	Fabrication of rotodome for E-2A. Development of vertical tail surfaces for the E-2A. Program is continuing.
Gyrodyne	QH-50 helicopter rotor blades under study.
Hamilton Standard	Study of P-2 fiberglas propeller blades.
IITRI	An investigation of material parameters influencing creep and fatigue life in filament wound laminates. (BuShips)
Kaman Aircraft	Development of FRP cowlings, ducts, fairings and panels for the UH-2AB. There have been several contracts on this which are completed.
Lockheed Aircraft	Design and development of P-2 and P-3 aft fuselage section.
Materials Research Lab	Investigation of factors controlling the strength of composites, interface fracturing and fracture toughness of adhesive joints.
Naval Air Engineering Center	Investigation of reinforced plastic laminates and sandwich structures, testing of box-beams. Work continuing in-house.
North American Aviation	Design and manufacture of a horizontal stabilizer for the T-2A. Program is completed.
North American Aviation	Design and manufacture of fiberglas box-beam for evaluation. Program is continuing.
North American Aviation	Study of fiberglas elevator and control surfaces for the OV-10A (COIN).
Pittsburgh Plate Glass	Basic study of hollow fiberglas fibers.
Whittaker-Narmco	Interlaminar shear of filament wound plastics.
Whittaker-Narmco	Development of high temperature resins and adhesives.

Table 11-1 (cont'd)

C. Army Programs

Company	Program
Boeing-Vertol	Protective materials for erosion of helicopter rotor blades.
Hayes International	A feasibility study of reinforced plastics for primary structures of Army aircraft.
Hayes International	Reinforced plastic landing gear for the UH-1 helicopter.
Kaman Aircraft	Compilation and analysis of test data on fiberglas reinforced plastics.
Oklahoma University	Previous contracts have been to determine strength properties and analytical procedures for sandwich structures with fiberglas facings. A continuing contract will study process methods for sandwiches, including methods for making prepregs.
Stanford University	Previous contracts have dealt with both fabrication and analytical procedures for sandwich structures. A continuing contract will study analytical methods for the sandwiches.
Whittaker-Narmco	A study of the mechanical relationship of reinforcements and the binder matrix. Study is now continuing.

APPENDIX A

MILITARY SPECIFICATIONS, STANDARDS, AND HANDBOOKS RELATING TO AIRCRAFT STRUCTURES

Number	Title
AFSCM 80-1 "HIAD"	Handbook of Instructions for Aircraft Design (Headquarters, Air Force Systems Command)
ANC-23	Sandwich Construction for Aircraft, Part II - Materials Properties and Design Criteria
MIL-HDBK-17	Plastics for Flight Vehicles, Part I - Reinforced Plastics
MIL-HDBK-23, Part I	Composite Construction for Flight Vehicles, Part I - Fabrication, Inspection, Durability, and Repair
MIL-HDBK-23, Part III	Composite Construction for Flight Vehicles; Part III - Design Procedures
MIL-HDBK-23	(Proposed Chapter, October 1964) Sandwich Cores
FED-STD-175	Adhesives, Methods of Testing
FED-STD-406	Plastics, Methods of Testing
MIL-STD-401A	Sandwich Construction and Core Materials, General Test Methods
MIL-A-927	Adhesive, Synthetic Resin (For Phenolic Laminates)
MIL-A-5090	Adhesive, Heat Resistant, Airframe Structural, Metal to Metal
MIL-S-5711	Structural Criteria, Piloted Airplane Structural Tests, Flight
MIL-P-7094	Plastic Parts, Aircraft Requirements and Tests for Rain-Erosion Protection of

MILITARY SPECIFICATIONS, STANDARDS, AND HANDBOOKS RELATING TO AIRCRAFT STRUCTURES (cont)

Number	Title
MIL-C-7438	Core Material, Aluminum, for Sandwich Construction
MIL-C-7439	Coating, Rain Erosion Resistant and Rain Erosion Resistant with Anti-Static Surface Treatment, for Plastics Laminates
MIL-R-7575B	Resin, Polyester, Low-Pressure Laminating
MIL-R-7705	Radomes, General Specification for
MIL-S-7998	Sandwich Construction Core Material, Balsa Wood
MIL-P-8013	Plastic Materials, Glass Fabric Base, Low Pressure Laminated
MIL-C-8073	Core Material, Plastic Honeycomb, Laminated Glass Fabric Base,, for Aircraft Structural Applications
MIL-STD-8073A	Core Material, Plastic Honeycomb, Laminated Glass Fabric Base, for Aircraft Structural Applications
MIL-C-8087	Core, Material, Foamed-in-Place, Polyester-Diisocyanate Type
MIL-G-8602A	Glass, Laminated, Flat, Aircraft
MIL-S-8698	Structural Design Requirements, Helicopters
MIL-F-8785(ASG), Notice No. 1	Flying Qualities of Piloted Airplanes
MIL-A-8860(ASG)	Airplane Strength and Rigidity, General Specification for
MIL-A-8861(ASG)	Airplane Strength and Rigidity, Flight Loads
MIL-A-8862(ASG)	Airplane Strength and Rigidity, Landplane Landing and Ground Handling Loads

MILITARY SPECIFICATIONS, STANDARDS, AND HANDBOOKS RELATING TO AIRCRAFT STRUCTURES (cont)

Number	Title
MIL-A-8863(ASG)	Airplane Strength and Rigidity, Additional Loads for Carrier-Based Landplanes
MIL-A-8864(ASG)	Airplane Strength and Rigidity, Water and Handling Loads for Seaplanes
MIL-A-8865(ASG)	Airplane Strength and Rigidity, Miscellaneous Loads
MIL-A-8866(ASG)	Airplane Strength and Rigidity, Reliability Requirements, Repeated Loads, and Fatigue
MIL-A-8867(ASG)	Airplane Strength and Rigidity, Ground Tests
MIL-A-8868(ASG)	Airplane Strength and Rigidity, Data and Reports
MIL-A-8869(ASG)	Airplane Strength and Rigidity, Special Weapons Effects
MIL-A-8870(ASG)	Airplane Strength and Rigidity, Vibration, Flutter, and Divergence
MIL-S-9041	Sandwich Construction; Plastic Resin, Glass Fabric Base, Laminated Facings and Honeycomb Core for Aircraft Structural Applications
MIL-A-9067	Adhesive Bonding, Process and Inspection Requirements for
MIL-C-9084	Cloth, Glass, Finished, for Polyester Resin Laminate
MIL-F-9118	Finish, Glass Fabric, for Reinforced Plastic Laminates
MIL-R-9299	Resin, Phenolic, Low Pressure Laminating
MIL-R-9300A	Resin, Epoxy, Low Pressure Laminating

MILITARY SPECIFICATIONS, STANDARDS, AND HANDBOOKS RELATING TO AIRCRAFT STRUCTURES (cont)

Number	Title
MIL-P-9400	Plastic Laminate Materials and Sandwich Construction, Glass Fiber Base, Low Pressure Aircraft Structural Process Specification Requirements
MIL-S-17917	Sandwich Construction, Aluminum Alloy Facings, Balsa Wood Core
MIL-C-21275A	Core Material, Metallic, Heat-Resisting, for Structural Sandwich Construction
MIL-G-21729A	Glass-Fiber Base Laminate, Epoxy Resin
MIL-R-21931	Resin, Epoxy
MIL-R-25042	Resin, Polyester, High Temperature Resistant, Low Pressure Laminating
MII-S-25392	Sandwich Construction, Plastic Resin, Glass Fabric Base, Laminated Facings, and Polyester Diisocyanate Foamed-in-Place Core for Aircraft Structural Applications
MIL-P-25395	Plastic Materials, Heat Resistant, Low Pressure Laminated Glass Fiber Base, Polyester Resin
MIL-P-25421	Plastic Materials, Glass Fiber Base - Epoxy Resin, Low Pressure Laminated
MIL-A-25463	Adhesive, Metallic Structural Sandwich Construction
MIL-R-25506A	Resin, Silicone, Low-Pressure Laminating
MIL-P-25515	Plastic Materials, Phenolic-Resin, Glass-Fiber Base, Low-Pressure Laminated
MIL-P-25518	Plastic Materials, Silicone Resin, Glass-Fiber Base, Low Pressure Laminated

APPENDIX B
MILITARY AIRCRAFT, CURRENT OR UNDER CONSTRUCTION (14, 17)

Fighter Aircraft

Designation	Type	Manufacturer	Speed	Gross Weight (1000 lbs)	Service	Note
F-4B F-4C "Phantom II"	Biservice fighter (2-place) all-weather carrier-based strike, interception and reconnaissance.	Lockheed-Georgia Co.	1650 mph. max.	40	USNavy Marine Corps USAF	F-4B F-4C, RF-4C
F-5A F-5B	Tactical fighter; Fighter and combat trainer	Northrop Corp.	925 mph, max. 885 mph, max.	12.92 13.5		-
F-8E "Crusader"	Carrier-based fighter aircraft; all-weather intercept and surface attack.	Ling-Temco-Vought, Inc,	Near Mach 2	28.7	USNavy	-
F-11A "Tiger"	Fighter, normally carrier-based, (single-place)	Grumman Aircraft Engr. Corp.	Mach 1.2 category	21.174	US Army	-
F-100A, C, D "Super Sabre"	Fighter bomber (single-place)	North American, Los Angeles	800 mph, max.	21 (empty)	-	-
F-100F "Super Sabre"	Fighter bomber and proficiency trainer (2-place)	North American, Los Angeles	800 mph, max.	22.336 (empty)	-	-
F-104G	Multi-purpose fighter-bomber	Lockheed-California Co.	1500 mph	-	USAF NATO (Japan) (Canada)	F-104A to 104D F-104N F-104J, F-104DJ CF-104, CF-104D
F-105D "Thunderchief" F-105F	Multi-purpose fighter-bomber, all-weather strike, reconnaissance and interception (single and 2-place)	Republic Aviation Corp.	Mach 2.25	48.4	USAF	-
F-106A,B "Delta Dart"	Jet interceptor (single or 2-place)	Convair Division, General Dynamics	Mach 2.0	35	USAF	-
F-111A F-111B	Tactical multi-purpose fighter (2-place) reconnaissance and interception variable sweep wing, all-weather attack	General Dynamics Corp.	Mach 2.5	70-80 65	USAF USNavy/Marines	-
Strike Aircraft						
A-3 "Sky Warrier" B-66	Attack-bomber and reconnaissance aircraft (3-place)	Douglas Aircraft Douglas Aircraft	600-700 mph	70	USNavy USAF	A-3 B-66
		NAVY - A-3: EA-3A; EA-3B; RA-3B; TA-3B. USAF - RB-66: RB-66A, B, C, B-66D; WB-66D; B-66E				
A-4A, B, C, D. "Skyhawk"	Attack and close-support aircraft, carrier-based (1-place)	Douglas Aircraft Company, Inc.	600-700 mph	24	USNavy	-
A-5A "Vigilante"	Carrier-based attack reconnaissance and tanker	North American Aviation	Mach 2.1	60	USNavy	-

Designation	Type	Manufacturer	Speed	Gross Weight (1000 lbs)	Service	Note
A6A "Intruder"	Low-level attack close support aircraft (2-place) carrier-based	Grumman Aircraft Engr. Corp.	(Classified)	-	USNavy	-
A-7A	Light attack aircraft (UAL) (carrier-based)	Ling-Temco-Vought	577 mph, max.	26	USNavy	-
RB-47L "Stratojet"	Multi-sensor reconnaissance with residual strike ability	The Boeing Company	Mach 0.86	202	USAF	-
B-52 "Stratofortress"	Long-range strategic missile carrier and bomber (6-place)	The Boeing Company	650 mph, max. (mach 0.98)	488	USAF	-
XB-52 YB-52 B-52A thru H						
B-58A "Hustler"	Long range strategic bomber and reconnaissance (external payload pods) (3-place)	General Dynamics, Fort Worth	Mach 2.0	160	USAF	-
Reconnaissance and Anti-Submarine Warfare (ASW) Aircraft						
E-1B "Tracer"	Airborne early-warning and fighter direction	Grumman Aircraft Engr. Corp.	-	27	USNavy	-
E2A "Hawkeye"	Electronic reconnaissance aircraft, 5-place	Grumman Aircraft Engr. Corp.	(Classified)	48.5	USNavy	-
OV-1 "Mohawk"	Surveillance aircraft; observation (visual, camera and radar) (2-place)	Grumman Aircraft Engineering Corp.	296-310 mph	12.675	US Army	-
P-2 "Neptune"	Anti-submarine warfare and patrol aircraft	Lockheed Aircraft Corp.	345 mph, max.	76.338	USNavy NATO	-
P-3 "Orion"	Anti-submarine warfare (ASW) aircraft; search and strike and ocean patrol	Lockheed-California Company	425 mph	127.5	USNavy	-
RA-5C "Vigilante"	Attack and tactical reconnaissance aircraft, carrier-based.	North American Aviation, Columbus Division	Mach 2	72	USNavy	-
U-2A	Day multi-sensor reconnaissance	Lockheed Aircraft Corp.	Mach 0.9	17.5	USAF	-
YF-12A (formerly A-11)	High-performance research intercepter and reconnaissance	Lockheed Aircraft Corp.	Mach 3.0	70	USAF	-

Transport, Training, and Utility Aircraft

Designation	Type	Manufacturer	Speed	Gross Weight (1000 lbs)	Service	Note
C-1A "Trader"	Carrier on-board delivery transport	Grumman Aircraft	220 KT	21	USNavy	-
C-124 C-124A C-124E	Multi-purpose troop and cargo transport aircraft	Douglas Aircraft	235 mph, max.	175	-	-
C-130E HC-130H "Hercules"	Multi-purpose cargo and personnel transport	Lockheed-Georgia Company	557 mph, max.	3.18	USAF	C-130 configuration used by Navy, Marine Corps.
C-133A, B "Cargo master"	Multi-purpose troop and cargo transport aircraft	Douglas Aircraft	323 mph, max.	286	USAF	-
C-135B "Stratolifter	Long-range jet transport (developed from KC-135A Strato tanker) (126 troops or 44 stretchers, 54 ambulatory)	The Boeing Company	638 mph, max.	275.5	USAF	(MATS)
VC-137	VIP transport version of C-135B (derived from Boeing 320-B commercial)	The Boeing Company	-	-	-	-
C-140A VC-140B "Jet Star"	Personnel transport	Lockheed-Georgia Company	550 mph, max.	41.5	USAF	-
C-141A "Star Lifter"	Long-range multi-purpose cargo and personnel transport	Lockheed-Georgia Company	550 mph, max.	318	USAF	-
VC-4A "Gulf Stream"	Executive transport	Grumman Aircraft	303 KT	35.1	US Coast Guard	-
CX-HLS	Long-range heavy logistics transport system	(either) Boeing Douglas Lockheed	Mach 1.0	700	-	-
T-1A	Jet trainer aircraft	Lockheed Aircraft Corp.	600 mph,	16.4	USNavy	-
T2A/T2B "Buckeye"	Basic jet trainer, 2-place	North American, Columbus	492 mph, max.	10	USNavy	-
T-33	Jet trainer (2-place)	Lockheed-California Company	580 mph, max.	15	USAF USNavy	TV-2
T-37B T-37C	Jet primary trainer (2-place)	Cessna Aircraft Co.	408 mph 351 mph	6.588 8.222	-	-
T-38A "Talon"	Jet trainer (2-place)	Northrop Corporation	844 mph	11.7 (at TO)	USAF	-
T-39A, B, C "Sabreliner"	Multi-purpose utility trainer; combat readiness	North American - Los Angeles	500 mph, max.	17.76	-	A lso VIP transport
U-3A U-3B	Utility transport, (4-5 place)	Cessna Aircraft Co.	217 mph 221 mph	4.83 4.99	USAF	-
U-8F "Seminole"	Twin-engine utility transport (6-place)	Beech Aircraft Corp.	237 mph	7.7	US Army	

Research Aircraft

Designation	Type	Manufacturer	Speed	Gross Weight (1000 lbs)	Service	Note
EC-131G	Electronics and missile research (other version for cargo, ambulance, navigation training and liaison)	General Dynamics/Convair	243 KT	47	USNavy	-
RB-57F	High altitude air-sampling and reconnaissance aircraft (modified Martin B-57)	General Dynamics Corp. Fort Worth Division	(classified)	-	-	-
X-15	High-speed rocket propelled, for high altitude, hypersonic research	North American Aviation, Los Angeles	(to) mach 8	(to) 49.64	USAF /NASA	-
X-21	Laminar Flow Control demonstration aircraft	Northrop Corporation	561 mph, max.	83	USAF	-
XB-70A "Valkyrie"	Research aircraft (2-place)	North American Aviation, Los Angeles Division	2000 mph, max.	-	-	-
Assault Helicopter						
CH-53A (Mod S-65)	Heavy assault helicopter, amphibious (41-place)	Sikorsky Aircraft	196 mph, max.	33.484	US Marine Corps.	-
207 "Sioux Scout"	Experimental attack-fighter "Flying Mockup" (2-place tandem)	Bell Helicopter Co.	125 mph, max.	3.05		-
"Tomahawk"	Proposed Armed helicopter (version of UH-2 with uprated engines)	Bell Helicopter Co.	173 mph, max.	7.667	US Army	-
Observation Helicopter						
OH-4A	Light observation helicopter (4-place)	Bell Helicopter Co.	127 mph, max.	2.5	US Army	-
OH-5A (Mod 1100)	Light observation helicopter (4-place)	Hiller Aircraft Co.	-	-	US Army	-
OH-6A (Mod 369)	Light observation helicopter, utility helicopter (4-place)	Hughes Tool Co. Aircraft Division	147 mph, max.	2.1	US Army	-
OH-13S	Military observation helicopter (3-place)	Bell Helicopter Co	105 mph, max.	2.85	US Army	-
Transport, Rescue and Antisubmarine Helicopter						
CH-3C (Mod S-61R)	Rear-loading transport helicopter (28-place)	Sikorsky Aircraft	165 mph, max.	19.5	USAF	-
CH-21C	Medium transport helicopter (Operational, no longer in production)	Bell Helicopter Co.	131 mph, max.	13.5	-	-
CH-37 (Mod S-56)	Heavy transport helicopter (36-place)	Sikorsky Aircraft	130 mph, max.	31.0	US Army	-

Designation	Type	Manufacturer	Speed	Gross Weight (1000 lbs)	Service	Note
CH-46A/ UH-46A "Sea Knight"	Transport helicopter, amphibious (27-place)	Boeing-Vertol	165 mph, max.	19.0	USNavy US Marine Corps	
CH-47A "Chinook" (Model 114)	Heavy transport, rear-loading ramp (46-place)	Boeing-Vertol	170 mph, max.	33.0	US ARMY	-
HH-43B/F Mod. K-600-3 K-600-5	Rescue and utility helicopter (12-place)	Sikorsky Aircraft	120 mph, max.	5.969	USAF	-
HH-52A (Mod S-62A)	Transport and rescue helicopter (13-place)	Sikorsky Aircraft	109 mph, max.	8.1	US Coast Guard	-
OH-43D UH-43C (Mod K-600)	Rescue helicopter (4-5 place)	Sikorsky Aircraft	105 mph, max. 100 mph, max.	5.367 6.203	USAF	-
QH-50C	Drone anti-submarine helicopter	Gyrodyne Co. of America Inc.	90 mph, max.	2.27	USNavy	-
5H-3A (Mod S-61B)	Anti-submarine and transport helicopter, amphibious (28-place)	Sikorsky Aircraft	172 mph, max.	18.044	USNavy	-
VH-34, SH-34, CH-34, UH-34, HH-34 (Mod S-58)	Medium transport and utility helicopter (12-18 place)	Sikorsky Aircraft	122 mph, max.	13.0	All Military services	-
YCH-54 (Mod S-64)	Flying crane prototype, heavy-lift helicopter	Sikorsky Aircraft	117 mph, max.	38.0	US Army	-
Tip-turbojet	Heavy-lift flying crane, 20-ton payload	Hiller Aircraft Co.	-	-	US Army	-
Utility, Support and Training Helicopter						
OH-23D	Utility trainer (3-place)	Hiller Aircraft Co.	95 mph, max.	2.7	US Army	-
OH-23G (Model 12E)	Utility helicopter (3-place)	Hiller Aircraft Co.	96 mph, max.	2.8	US Army	-
OH-23F (Model E4)	Utility helicopter (4-place)	Hiller Aircraft Co.	96 mph, max.	2.8	-	-
UH-1 B/E (204) "Iroquois"	Military utility and armed helicopter (8-10 place)	Bell Helicopter Co.	138 mph, max.	6.6	US Army (UH-1B) US Marine Corps (UH-1E)	-
UH-1D (205)	Squad-carrier utility helicopter (15-place)	Bell Helicopter Co.	138 mph max.	6.6	US Army	-
UH-1F (204) "Iroquois"	Missile site support helicopter (11-place)	Bell Helicopter Co.	1.38 mph, max.	6.6	USAF	-

Designation	Type	Manufacturer	Speed	Gross Weight (1000 lbs)	Service	Note
UH-2A/B "Seasprite"	General utility helicopter (6-13 place)	Kaman Aircraft Corp.	162 mph, max.	8.637	USNavy (Army)*	-
476-3B-1 (no military designation)	Utility helicopter; instrument helicopter training (12-place)	Bell Helicopter Co.	105 mph, max.	2.95	US Army	-
Research and Experimental Helicopter						
XH-5/A/N	Rigid-rotor development helicopter	Lockheed-California Co.	174 mph, max.	3.5	US Army/Navy (contract) NASA/TRECOM	-
XU-9A (Mod. 385)	Hot cycle research vehicle (Hot cycle tip propulsion)	Hughes Tool Co.	170 mph, max.	15.3	US Army (contract study)	-
Mod 16H-1 "Pathfinder"	High-speed compound helicopter (5-place, ducted prop.)	Piasecki Aircraft Corp.	178 mph, max.	5.7	-	-
Strike and Reconnaissance, V/STOL Aircraft						
P. 1127	Experimental vectored (VTOL) strike, reconnaissance fighter	Hawker Siddeley Aviation, Ltd.	Mach 8 max.	-	Tripartite (US, UK, Germany)	Operational evaluation
P. 1154	(Follow-on to the P. 1127)	Hawker Siddeley Aviation, Ltd.	Mach 2	-	-	In design stage
"Coin"	Counter-insurgence	North American, California	-	10	Tri-service	-
Transport, V/STOL Aircraft						
CV-7A "Buffalo"	STOL tactical transport	de Haviland (Canada)	212 KT	38	USArmy	-
X-19	Tilt-prop experimental transport (6-place) (V/STOL)	Curtiss-Wright Corp.	460 mph.	(VTOL)-13.66 (STOL)- 14.75	USAF (contract)	-
X-22A "Frontiersman"	Experimental transport, dual tandem ducted propellar (8-place)(V/STOL)	Bell Aerosystems Co.	350 mph, max.	(VTOL)-15 (STOL)-17	Tri-service research	-
XC-142A	Tilt-wing (deflected slip stream experimental medium transport (32-place (V/STOL)	L-T-V, Hiller, Ryan	400+ mph, max.	(VTOL)-42.5 (STOL)-44.5	Tri-service (development)	-
Research, V/STOL Aircraft						
XV-4A "Hummingbird"	Augmented jet ejector research aircraft (2-place) (VTOL)	Lockheed-Georgia Co.	660 mph, max.	7.2	US Army (contract)	-
XV-5A	Fan-in-wing research aircraft (V/STOL)	Ryan Aeronautical Co. with General Electric Co.	545 mph, max.	(VTOL-12.5 (STOL)-15.5	US Army (contract)	-

*A high-speed compound version with augmented engines is being flown in an Army research program.

APPENDIX C

DEFENSE DOCUMENTATION CENTER LITERATURE SEARCH

A retrospective search of the Defense Documentation Center files from 1960 to the present was conducted prior to the writing of this report. Descriptor terms were broken into two groups for a machine search. The first group gave a complete drop-out, while the second group was coordinated for a selection of terms to yield the highest number of relevant references. Cards received as an outcome of this search contained only the AD numbers. Abstracts were obtained from DDC-TAB bulletins for review of reports. The abstracts were then scanned for selection.

SUMMARY OF DESCRIPTORS

Total number of descriptors	71
Number of descriptors for complete search	12
Number of descriptors for coordinated search	59
Number of relevant unclassified abstracts	79
Number of unclassified abstracts received	430
Percent relevancy	18.4%

LIST OF DESCRIPTORS

Group 1. Complete Drop-Out

Aerospace planes - Design
Airframes - Design
Sandwich construction
Sandwich panels
Army aircraft - Structures
Fuselages - Aerospace planes
Aircraft - Loading (Mechanics)
Airframes - Structural properties
Airplanes - Military requirements
Aircraft finishes - Plastic coating
Fuel tanks - Plastics
Aircraft - Fatigue

Group 2. Combined and Programmed for Coordinate Searching

Laminated plastics - Airframes
Laminated plastics - Mechanical properties
Laminated plastics - Sheets
Laminated plastics - Tensile properties
Laminated plastics - Tests
Laminated plastics - Thickness
Laminated plastics - Exposure
Laminated plastics - Damping
Laminated plastics - Glass textiles

Group 2. (Cont.)

Laminates - Airframes
Laminates - Mechanical properties
Laminates - Sheets
Laminates - Tensile properties
Laminates - Tests
Laminates - Thickness
Laminates - Exposure
Laminates - Damping
Laminates - Glass textiles

Aircraft - Aerodynamic characteristics
Aircraft - Material

Airframes - Design
Airframes - Fairings
Airframes - Fatigue (Mechanics)
Airframes - Loading (Mechanics)
Airframes - Stresses

Structural parts - Airborne
Structural parts - Honeycomb cores
Structural parts - Plastic coatings
Structural parts - Shear stresses
Structural parts - Stresses
Structural parts - Vibration

Epoxy plastics - Laminates

Honeycomb cores - Bonding
Honeycomb cores - Plastics
Honeycomb cores - Sandwich construction

Composite materials - Airplanes
Composite materials - Glass
Composite materials - Glass textiles
Composite materials - Laminates
Composite materials - Plastics

Landing gear - Design
Landing gear - Loading (Mechanics)
Landing gear - Naval aircraft

Glass textiles - Adhesion
Glass textiles - Bonding
Glass textiles - Failure (Mechanics)
Glass textiles - Honeycomb cores
Glass textiles - Laminates
Glass textiles - Mechanical properties
Glass textiles - Stresses
Glass textiles - Tensile properties

Group 2. (Cont.)

Rotor blades (Rotary wings) - Erosion
Rotor blades (Rotary wings) - Fatigue (Mechanics)
Rotor blades (Rotary wings) - Vibration

Helicopter rotors - Fatigue (Mechanics)
Helicopter rotors - Materials

Helicopter - Design
Helicopter - Vibration

Stresses - Composite materials

APPENDIX D

SUMMARY OF FOREST PRODUCTS LABORATORY WORK ON FIBERGLAS REINFORCED PLASTICS AND SANDWICH CONSTRUCTIONS

FIBERGLAS REINFORCED PLASTICS

<table>
<tr><th rowspan="2">Type of work</th><th colspan="2">Materials involved</th><th rowspan="2">FPL reports involved</th><th rowspan="2">Typical data</th><th rowspan="2">Remarks</th></tr>
<tr><th>Reinforcement</th><th>Resin</th></tr>
<tr><td>Mechanical properties of laminates
Tension
Compression
Bending
Moduli</td><td>Glass cloths:
112/114 finish
116/114 finish
120/114 finish
128/114 finish
143/114 finish
162/114 finish
164/114 finish
181/114 finish
112/Volan A
120/Volan A
143/Volan A
181/Volan A
181/Heat cleaned</td><td>Polyester
Epoxy
Epon 828
Epon 1001
Silicone
DC 2106</td><td>1820 (1951) (1958)*
1820A (1953) (1960)
1820B (1955) (1960)
1820C (1956)
1820D (1958)
*Second date. "Reaffirmed Data"</td><td>Condition - dry
<table>
<tr><td>Parameter:</td><td colspan="6">Tension and Modulus</td><td colspan="4">Compression and Modulus</td></tr>
<tr><td>Load direction:</td><td colspan="2">0°</td><td colspan="2">45°</td><td colspan="2">90°</td><td colspan="2">0°</td><td colspan="2">90°</td></tr>
<tr><td>Material:</td><td>Kpsi</td><td>Ex10⁶</td><td>Kpsi</td><td>Ex 10⁶</td><td>Kpsi</td><td>Ex10⁶</td><td>Kpsi</td><td>Ex10⁶</td><td>Kpsi</td><td>Ex 10⁶</td></tr>
<tr><td>Polyester - 181</td><td>45.5</td><td>2.7</td><td>20.5</td><td>1.30</td><td>42.0</td><td>2.55</td><td>34.5</td><td>3.3</td><td>36.4</td><td>3.2</td></tr>
<tr><td>Polyester -143</td><td>89.8</td><td>5.7</td><td>14.0</td><td>1.66</td><td>10.7</td><td>1.69</td><td>52.0</td><td>5.2</td><td>22.0</td><td>1.6</td></tr>
<tr><td>Epoxy -181</td><td>56.7</td><td>3.6</td><td>33.5</td><td>2.4</td><td>53.4</td><td>3.5</td><td>45.9</td><td>3.75</td><td>45.9</td><td>3.25</td></tr>
<tr><td>Epoxy -143</td><td>104.8</td><td>5.3</td><td>15.7</td><td>1.9</td><td>12.6</td><td>2.3</td><td>73.9</td><td>5.6</td><td>32.4</td><td>2.28</td></tr>
</table></td><td>Conditions dry and wet
Sample thickness
Mostly 1/4 inch
Some 1/8 inch</td></tr>
<tr><td>Mechanical properties at elevated temperatures
Tensile
Compression
Flexure
Interlaminar shear
Bearing
Tensile-stress rupture</td><td>Glass cloth -
181-A1100
181-Volan A
Asbestos</td><td>Silicone -
DC 2106
Phenolic -
CTL-91LD
RM-41RPD
Phenyl Silane-
CTL 37-9X
Narmco 534
Epoxy
ERSB-0111
Epon 1031</td><td>WADC TR 59-216
WADC TR 59-229
WADC TR 59-569
WADD TR 60-177, Pt-I
ASD TDR 61-482
ASD TDR 62-629
RTD TDR 63-4091
RTD TDR 63-4154</td><td>Typical curves shown in Figures No. D-1 and No. D-2</td><td>Tested at varying intervals at temp. up to 100° F.</td></tr>
<tr><td>Directional properties of laminates
Tension & modulus
Compression & modulus
Flexure & modulus
Panel shear & modulus
Edgewise shear & modulus</td><td>Glass cloths:
112/114 finish
116/114 finish
120/114 finish
128/114 finish
143/114 finish
162/114 finish
164/114 finish
181/114 finish
182/114 finish
183/114 finish
184/114 finish
Also: (finish unspecified; Volan A?)
112/Volan A
120/Volan A
143/Volan A
181/Volan A</td><td>Polyester
Epoxy</td><td>1803 (1949) (1956)
1803A (1950) (1956)
1803B (1955)
1803C (1957)
1821 (1951) (1958)
1821A (1953) (1959)
1841 (1959)
1853 (1955) (1960)</td><td>Typical directional graphs shown in Figure No. D-3 No. D-4 No. D-5</td><td>Directional rosette developed from stated tests and theoretical calculations</td></tr>
</table>

FIBERGLAS REINFORCED PLASTICS (CONT)

Type of work	Materials involved		FPL reports involved	Typical data	Remarks
	Reinforcement	Resin			
Bolt bearing properties of laminates	Glass Cloth 112 120 143 162 181 184 112/Volan A 181/Volan A	Polyester Epoxy (Epon 828/CL)	1824 1824A 1824B 1824C	Epoxies appeared to be somewhat stronger than the polyesters. (A letter report of August 1961 gives results of testing of several different types of joints, loaded in tension, 1/8 inch bolts with 1/8 inch laminate. Material tested was Epon 1031 epoxy, with 181/Volan A	Laminates were 1/8 and 1/4 inch thick. D/t (bolt diameter to thickness) equals 1 and 4.
Dimensional stability	Glass cloths: 181/Volan A	Polyester (Selectron 5003)	1858 (1956)	Change (from 0 relative humidity to 'soaked') Warp — Change, % Parallel — 0.01 to 0.077 Perpendicular — 0.007 to 0.073 Note: slight increase with increased resin	
Interlaminar shear	Eleven fabrics Four fabrics (Unidirectional) One fabric One fabric	Polyester Epoxy (Epon 828/CL) Phenolic (CTL-91LD) Silicone (DC 2106)	1848 1848A 1892	Data for Epoxy - 828/CL and Scotchply 1002 Combination — 0° — 45° 181 Volan A/ Scotchply 1002 Epoxy — 6.3 — - Unidirectional/ Scotchply 1002 Epoxy — 6.7 — 5.5 Crossplied unwoven/ Scotchply 1002 Epoxy — 6.1 — 6.5 112/Volan A/ 828 CL — 5.4 — 6.3 120/Volan A/ 828 CL — 4.5 — 5.3 143/Volan A/ 828 CL — 6.3 — 6.2	Given at both 0° and 45° to the warp (some cases) Wet and dry (some cases) Modulus of rigidity Samples 1-1/2 approx
Poisson's Ratio	Glass cloths: 112/Volan A 120/114 128/114 143/Volan A 162 181/Volan A 182/114 184/Volan A	Polyester (Selectron 5003) Epoxy (Epon 828/CL)	1853 (1955) 1860 (1957)	Ratio at: — 0° — 90° Cloth — Resin 112 — Polyester — 0.13 — 0.13 120 — Polyester — 0.15 — 0.14 128 — Polyester — 0.14 — 0.11 143 — Polyester — 0.25 — 0.10 162 — Polyester — 0.17 — 0.13 181 — Polyester — 0.11 — 0.13 182 — Polyester — 0.19 — 0.14 184 — Polyester — 0.13 — 0.12 181 — Epoxy — 0.17 — 0.18	Samples - 1/8" thick

FIBERGLAS REINFORCED PLASTICS (CONT)

<table>
<tr><th rowspan="2">Type of work</th><th colspan="2">Materials involved</th><th rowspan="2">FPL reports involved</th><th rowspan="2">Typical data
Condition - dry</th><th rowspan="2">Remarks</th></tr>
<tr><th>Reinforcement</th><th>Resin</th></tr>
<tr><td>Thickness (laminate), properties variations with</td><td>Glass cloths:
181/114</td><td>Polyester</td><td>1807 (1955)</td><td>
<table>
<tr><th>Tests -</th><th>Static</th><th>Bending and</th><th>Modulus</th></tr>
<tr><td>Thickness, Inches</td><td>1/16</td><td>1/4</td><td>1/2</td></tr>
<tr><td>Tension</td><td>43.1</td><td>46.8</td><td>47.0</td></tr>
<tr><td>Modulus</td><td>2.76</td><td>2.87</td><td>2.60</td></tr>
<tr><td>Compression</td><td>34.9</td><td>39.6</td><td>33.7</td></tr>
<tr><td>Modulus</td><td>2.96</td><td>2.83</td><td>2.76</td></tr>
</table>
</td><td>(See also
Figure No. D-6)</td></tr>
<tr><td></td><td>112/114
181/114
183/114
181/Volan A
181/112</td><td>Polyester
(alkyd/styrene)
Epoxy
(Scotchply 1002)
Silicone
(PC 2106)</td><td>1831 (1954)
1876 (1960)</td><td>
Typical Data:*
<table>
<tr><th>Materials:</th><th colspan="3">181/Polyester</th><th>181/Epoxy</th></tr>
<tr><td>Test:</td><td>Tensile</td><td>Modulus</td><td>Comp</td><td>Compression</td></tr>
<tr><td>Av. Mils:</td><td></td><td></td><td></td><td></td></tr>
<tr><td>25</td><td>33.2</td><td>2.22</td><td>30.2</td><td>35.3</td></tr>
<tr><td>35</td><td>-</td><td>-</td><td>-</td><td>48.0</td></tr>
<tr><td>50</td><td>36.7</td><td>2.48</td><td>41.7</td><td>58.9</td></tr>
<tr><td>75</td><td>39.0</td><td>2.28</td><td>43.9</td><td>58.3</td></tr>
<tr><td>105</td><td>39.1</td><td>2.32</td><td>44.6</td><td>53.5</td></tr>
<tr><td>250</td><td>41.4</td><td>2.52</td><td>50.1</td><td>40.6</td></tr>
<tr><td>600</td><td>-</td><td>-</td><td>-</td><td>55.0</td></tr>
<tr><td>1000</td><td>-</td><td>-</td><td>-</td><td>46.6</td></tr>
<tr><td>1500</td><td>-</td><td>-</td><td>-</td><td>48.5</td></tr>
</table>
</td><td></td></tr>
<tr><td>Fatigue properties</td><td>Glass cloth:
112/Volan A
120/Volan A
143/Volan A
181/Volan A
181/114
181/A1100
181/heat cleaned
(unidirectional)
184/Volan A
Asbestos</td><td>Epoxy:
(Epon 828/CL)
(Shell X12100
(Scotchply 1002)
(Scotchply 1009)
(Scotchply
XPL06
Phenolic:
(CTL91LD)
(BU 17085)
(RM 984RPD)
Polyester:
(Plastron 911)
(Paraplex 43)
(PPL7-669)
(Selectron 5003)
Silicone:
(DC 2106)</td><td>1884 (1961)
WADC 59-27 (1959)
ASD-TDR-63-768 (1963)
(summary)
ML-TDR-64-86 (1964)
ML-TDR-64-168 (1964)</td><td>Figures No. D-7 and No. D-8 give SN curves, notched and unnotched, various amounts of precycling, moisture content, variation in mean tensile load.

*Data compiled from reported tabulations, Ed.</td><td>Curves tend to come together at high cycles</td></tr>
</table>

FIBERGLAS REINFORCED PLASTICS (CONT)

Type of Work	Materials involved		FPL reports involved	Typical data	Remarks
	Reinforcement	Resin			
Creep - Stress rupture	Glass cloth: 181/Volan A (+unidirectional)	Epoxy Polyester	1839 (1953) (1959) 1863 (1957) 2039 (1958) 2228 (1961)	Data: tensile, compression, flexure and shear type loading, at 0° and 45° to warp. Conditions: 73° F - 50% RH 73°F - in water. Plotted as stress rupture - time to failure at various percents of stress - strain vs. time. Show: vary small strain at 0°; strain about 6 times greater in 45° direction. Figures No. D-9, No. D-10 and No. D-11 give typical tensile-stress rupture data.	Also: materials typical for shipboard application.
Weathering, effect on laminates	Glass cloth: 181/Garan 181/Volan A Asbestos (Pyrotex)	Epoxy: (Shell X-131) Phenolic (CTL-91LD) (R/M 980RPD) Polyester: (Vibrin X-1068) (Paraplex P-43)	WADC-TR-55-319 (and supplements 1-4) (1956 - 1962) ASD-TR-61-145 (1961)	Table No. D-1 gives typical data on weathering, showing the effects of 3 years exposure at various test sites. Properties considered in study of weathering are: flexure, hardness, and elevated temperature/flexure; appearance, weight increase, flexure modulus, dry and wet and elevated temperature. General conclusions: factors affecting weathering include: type of reinforcement and resin, curing agent, resin content. Prestressing had no effect.	
Moisture, effect on laminates	Glass cloth: 181/Garan 181/Volan A 181/T31 181/114 181/A1100	Epoxy: (Epon 828/CL) Polyester: (Laminac 4232) Phenolic: (CTL91LD) Silicone: (DC 2106)	1819 (1950) (1956) WADC TR 58-486 (1959)	Type of data: tension, compression, flexure. Conditioning: Immersed in water - 73°F 100°F/100%RH/30 to 90 days. Boil 1/2, 2,4 hours. Conclusion: Compression and Modulus most affected. Believe 2 hour boil equals 30 day immersion, not generally accepted.	
Prestressing; effect on mechanical properties of laminates	Glass cloth: 112/114 143/114 181/114 181/Volan A 184/114	Polyester Epoxy: (Epon 828/CL)	1811A (1951) (1958) 1856 (1956) 1856A (1957) 1870 (1959)	Type of data: Effect on tensile, compressive preloading on tensile, compressive shear properties. Typical results: Tensile preloading affected compressive properties; lowering compression. Compressive preload had no effect on tensile or compression. Believe that crazing from tensile preload allows more moisture.	

Type of work	Materials involved	FPL reports involved	Typical data - or - Remarks
Mechanical Properties: Compression Shear Flexure Tensile	Aluminum honeycomb cores Aluminum multiwave cores Glass-fabric cores Commercial aluminum cores Experimental aluminum cores Plastic cores	1849 (1955, 1962) 1855 (1956) 1861 (1957) 1887 (1962)	Mechanical Properties of Core Materials (Selected) (See also Table 5-22)
Axial compression	Curved panels, sandwich: plywood facings aluminum facings fiberglas facings	1558 (1959)	Axial-Compression Tests - Curved Panels of Sandwich (Selected)
Edgewise compression	Aluminum facings, granulated-cork cores Steel facings, granulated-cork cores Steel facings, Cellular-cellulose-acetate cores Aluminum facings: Balsa (end-grain) cores Cellular-cellulose-acetate cores Cellular hard rubber cores Glass-cloth laminate facings, balsa wood cores.	1810 (1961)	Edgewise Compressive Strength, Sandwich, Glass-Cloth Laminate Facings and Balsa Wood Cores (Selected)
Shear Strength	Sandwich column, aluminum facings, cellular cellulose acetate cores	1815 (1955, 1960)	Comparison of Maximum Loads Due to Shear Failure in Sandwich Columns, Aluminum Facings on Cellulose Acetate Cores (Selected)

Type	Core Density(PCF)	Compressive Strength(PSI)	Shear	(PSI)
Aluminum (.005" foil corrugated to 3/8" cells)	3.05	234	152 (LT)	79 (LR)
	3.96	361	238 "	118 "
	5.18	571	332 "	172 "
	6.32	767	393 "	226 "
Aluminum, multiwave (1/4" cells)	4.03	212	136 (TL)	75 (TW)
	4.61	436	274 "	154 "
	4.59	465	308 "	162 "
Glass fabric (112, 114, phenolic; 1/4" cells)	3.46	286 (dry)	165 (TL, dry)	82 (TW, dry)
	4.44	442 "	282 " "	137 " "
	5.65	457 "	247 " "	127 " "

Facing:	Thickness (in.)	Width (in.)	Length (in.)	Curvature (in.)	Failure	
					Stress (lb per sq in)	Type
Aluminum: (0.005-in.) Core: 1/8" end-grain balsa	0.147	11.7	30.0	9.4	27,000	Crimping
	0.133	11.6	26.5	10.4	48,400	Crimping
Core: 1/8" cellular cellulose acetate	0.130	16.8	30.0	15.0	33,900	Crimping
	0.138	19.5	30.0	28.0	11,800	Separation
Fiberglas: (0.009-in.) Core: 1/8" end-grain balsa	0.140	11.5	30.0	10.0	16,650	Compression
	0.138	25.2	29.5	25.9	6,600	Buckling
	0.137	70.5	70.5	72.3	1,650	Buckling
Core: 1/8" cellular cellulose acetate	0.143	16.6	29.8	15.2	8,350	Crimping
	0.144	31.6	28.9	38.5	5,200	Buckling

Facing thickness (in.)	Core thickness (in.)	Compressive strength of facing materials (PSI)
0.016 (4-ply)	1/4	19,840
0.032 (8-ply)	1/4	20,800
0.044 (16-ply)	1/4	22,250

Dimension of column		Slope of clamps	Lateral Deflection, Final	Column loads, Final
Length (in.)	Thickness (in.)	(in. per in.)	(in.)	(lb per in. of width)
10.26	0.546	0.0184	0.10	1,253
10.26	0.546	0.0373	0.15	980
20.26	0.545	0.0242	0.50	1,065
20.26	0.545	0.0565	0.60	785
30.01	0.539	0.0388	0.96	665
30.01	0.541	0.0847	1.65	490

SANDWICH CONSTRUCTION (CONT)

<table>
<tr>
<td>Fatigue
(Tested in Shear)</td>
<td>Aluminum face and end-grain balsa core sandwich (Curve A)
Fiberglas-honeycomb core with fiberglas-laminate facing (Curve B)
Fiberglas-honeycomb core with aluminum facing (CurveB)
Fiberglas-laminate face and end-grain balsa core Curve C)
Cellular-hard-rubber core with aluminum facing (Curve D)
Cellular-hard-rubber core with fiberglas-laminate facing (Curve D)
Aluminum facing and aluminum honeycomb core (Curve E)
Glass-fabric-laminate facing with alkyd isocyanate foamed-in-place core (Curve F)</td>
<td>1559-B (1956)
1559-C (1956, 1962)
1559-D (1956, 1962)
1559-E (1956, 1962)
1559-H (1956, 1962)
1599-J (1952, 1958)</td>
<td>See Figure D-12</td>
</tr>
<tr>
<td>Effect of Cell Size on Compressive Strength</td>
<td>Aluminum facings:
Sitka spruce core
Balsa core
Expanded hard rubber core
Honeycomb core</td>
<td>1817 (1956)</td>
<td>Critical Stresses at Buckling of Faces (Selected)
(Total thickness = 0.52 ; Facing thickness = 0.012)
<table>
<tr><th>Material</th><th>Cell radius (in.)</th><th>Critical stress</th></tr>
<tr><td>Balsa</td><td>0.250</td><td>31.1</td></tr>
<tr><td></td><td>0.375</td><td>14.9</td></tr>
<tr><td></td><td>0.500</td><td>11.3</td></tr>
<tr><td>Cellular hard rubber</td><td>0.250</td><td>22.5</td></tr>
<tr><td></td><td>0.375</td><td>17.6</td></tr>
<tr><td></td><td>0.500</td><td>8.0</td></tr>
<tr><td>Paper honeycomb</td><td>0.1875</td><td>37.0</td></tr>
</table>
</td>
</tr>
<tr>
<td>Effect of Core Thickness on Shear Properties</td>
<td>Aluminum honeycomb core</td>
<td>1886 (1962)</td>
<td>Shear Properties of Aluminum Honeycomb Core (Selected data)
(Compression Loading)
<table>
<tr><th>Thickness (in.)</th><th>Length (in.)</th><th>Width (in.)</th><th>Density (lb/cuin.)</th><th>Shear Modulus (LT) (PSI)</th><th>Density (lb/cuin.)</th><th>Shear Modulus (WT) (PSI)</th></tr>
<tr><td>1/2</td><td>6</td><td>4</td><td>2.86</td><td>38,100</td><td>2.89</td><td>18,200</td></tr>
<tr><td>1</td><td>12</td><td>4</td><td>2.90</td><td>40,700</td><td>2.88</td><td>19,800</td></tr>
<tr><td>2</td><td>24</td><td>4</td><td>2.92</td><td>43.700</td><td>2.89</td><td>22,100</td></tr>
<tr><td>4</td><td>48</td><td>4</td><td>3.15</td><td>56,000</td><td>2.89</td><td>23,900</td></tr>
</table>
</td>
</tr>
<tr>
<td>Effect of Voids</td>
<td>Premolded glass-fabric facings on glass-fabric honeycomb cores
Wet lay-up glass-fabric facings on glass-fabric honeycomb cores</td>
<td>WADC TR 58-172 (1958)</td>
<td>Maximum Edge Load and/Or Facing Stress (Selected data)
(Compressive loading; normal conditions)
<table>
<tr><th></th><th>Thickness (in.)</th><th>Weight (lb/sq ft)</th><th>Facing stress at failure (PSI)</th><th>Load per inch of edge (lb)</th></tr>
<tr><td>Premolded:</td><td>0.335</td><td>0.693</td><td>27,290</td><td>1,730</td></tr>
<tr><td></td><td>0.334</td><td>0.696</td><td>27,800</td><td>1,727</td></tr>
<tr><td></td><td>0.335</td><td>0.713</td><td>27,870</td><td>1,754</td></tr>
<tr><td>Wet lay-up:</td><td>0.322</td><td>0.699</td><td>-</td><td>1,182</td></tr>
<tr><td></td><td>0.324</td><td>0.685</td><td>-</td><td>1,284</td></tr>
<tr><td></td><td>0.323</td><td>0.669</td><td>-</td><td>1,327</td></tr>
</table>
</td>
</tr>
</table>

SANDWICH CONSTRUCTION (CONT)

<table>
<tr><th>Type of work</th><th>Materials involved</th><th>FPL reports involved</th><th>Typical data -or- Remarks</th></tr>
<tr><td>Effect of Resin Content, Paper Weight and Bonding Adhesive</td><td>Paper honeycomb cores;
3/16" plywood facings</td><td>1796 (1961)</td><td>
Various Effects on Strength of Sandwich Panels (Selected data)
<table>
<tr><th rowspan="3"></th><th colspan="4">Shear stress developed in the bending test (PSI)</th><th colspan="2" rowspan="2">Compressive strength (PSI)</th></tr>
<tr><th colspan="2">Parallel</th><th colspan="2">Perpendicular</th></tr>
<tr><th>Dry</th><th>Wet</th><th>Dry</th><th>Wet</th><th>Dry</th><th>Wet</th></tr>
<tr><td>Variable:</td><td></td><td></td><td></td><td></td><td></td><td></td></tr>
<tr><td>Resin content percent: 5</td><td>49</td><td>14</td><td>15</td><td>5</td><td>40</td><td>9</td></tr>
<tr><td>10</td><td>67</td><td>17</td><td>22</td><td>6</td><td>53</td><td>11</td></tr>
<tr><td>15</td><td>78</td><td>25</td><td>20</td><td>6</td><td>50</td><td>14</td></tr>
<tr><td>Paper weight Pounds: 30</td><td>26</td><td>7</td><td>13</td><td>5</td><td>25</td><td>7</td></tr>
<tr><td>50</td><td>49</td><td>14</td><td>15</td><td>5</td><td>40</td><td>9</td></tr>
<tr><td>65</td><td>63</td><td>13</td><td>20</td><td>7</td><td>54</td><td>13</td></tr>
<tr><td>Bonding Adhesive: Phenolic</td><td>74</td><td>33</td><td>22</td><td>12</td><td>63</td><td>17</td></tr>
<tr><td>Silicate</td><td>78</td><td>25</td><td>20</td><td>6</td><td>50</td><td>14</td></tr>
<tr><td>Urea</td><td>70</td><td>24</td><td>-</td><td>-</td><td>54</td><td>15</td></tr>
</table>
</td></tr>
<tr><td>Effect of Defects on Strength

Properties:
Bending
Tensile
Compressive

Defects:
Wrinkles in facing
Gaps in core
Fabric laps in facing
Repaired unbonded areas
Delamination area
Low resin content
Fold in facing
Variation in plies
Unglued butt joint</td><td>Aluminum facing, end-grain balsa core
Aluminum facing, cotton-cloth honeycomb core
Glass-cloth facing, cellular cellulose acetate core
Glass-cloth facing, glass-cloth honeycomb core
Glass-cloth facing, end-grain balsa core.</td><td>1809 (1956)
1809A(1958)</td><td>General findings:

Poor bonds reduce tensile strength between facing and core, and reduce the compressive strength of a sandwich panel.
Unbonded areas between facing and core reduce edgewise compressive strength. (Same with delaminated areas.)
Wrinkles in glass-cloth facings reduce the binding, edgewise compressive and longitudinal tensile values by amounts in proportion to the depth of the wrinkle. (Shallow fold in cellophane parting film had no weakening effect.)
Fold in glass cloth reduced edgewise compressive strength, bending, longitudinal tensile.
Core gaps was questionable (1/16") or reduced compressive strength resulted (1/8").
As resin content of glass facings decreased, edgewise compressive strength decreased.
Increase in number of plies of glass cloth in facing had no substantial effect on edgewise compressive strength.
Butt-joints in glass-cloth facings had effect on bending and longitudinal tensile strength.</td></tr>
</table>

TABLE D-1. EFFECT OF 36 MONTHS' EXPOSURE AT DIFFERENT CONDITIONS ON THE FLEXURAL STRENGTH OF FOUR REINFORCED PLASTIC LAMINATES

Exposure condition or site	Change from control[1]		
	Normal	Wet	500° F.
	Percent	Percent	Percent
	VIBRIN X-1068		
Normal	+2	+2	-17
Wisconsin	-19	-26	-35
Florida	-33	-38	-34
Panama	-15	-17	-25
	PHENOLIC-ASBESTOS		
Normal	-7	-3	-3
Wisconsin	-11	-6	+7
Florida	-11	-2	+1
Panama	-13	-6	+7
	SHELL X-131 (1 PER CENT BF_3-400)		
Normal	-5	-3	-37
Wisconsin	-15	-9	-31
Florida	-38	-23	-31
Panama	-25	-6	-44
	SHELL X-131 (10 PER CENT DDS AND 1 PER CENT BF_3-400)		
Normal	-6	+4	-39
Wisconsin	-8	-3	-40
Florida	-20	-17	-39
Panama	-12	-2	-45

---

[1] Percentage increase or decrease of average strength values after specimens were reconditioned in a normal or wet atmosphere before test or tested at 500° F., after 1/2-hour exposure at 500° F., as compared to the corresponding value.

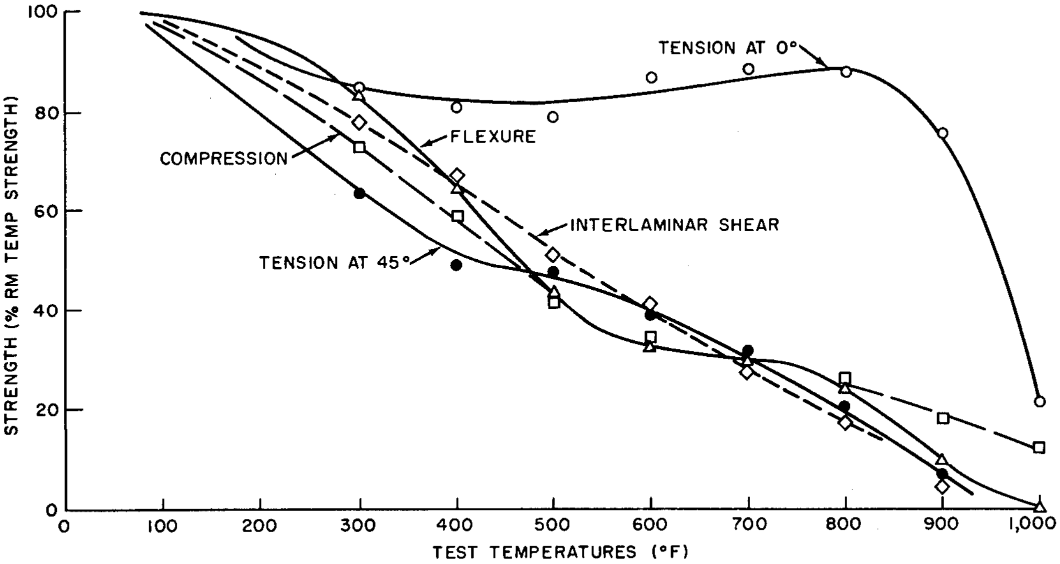


Figure D-1. Strengths vs. Temperature, Half-Hour Exposure - 181-A1100/CTL 37-9X Phenyl-Silane Laminates

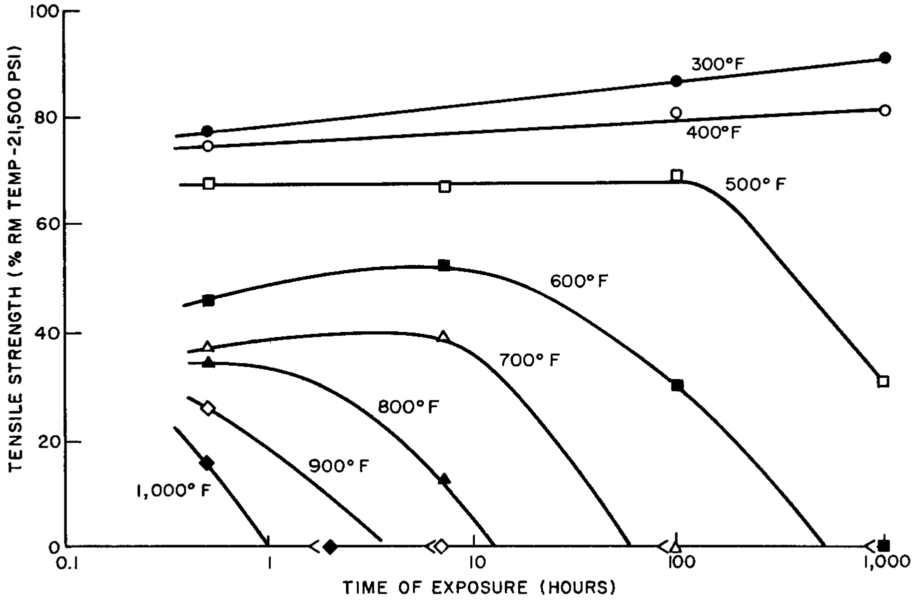


Figure D-2. Tensile Strength vs. Temperature - 181-A1100/Narmco 534 Phenyl-Silane Laminates

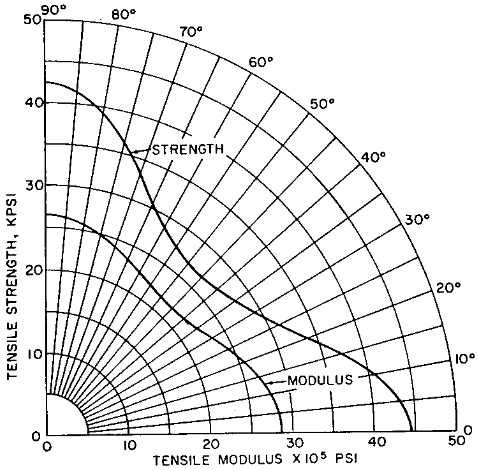


Figure D-3. Directional Tensile Strength and Modulus - 181/Epoxy

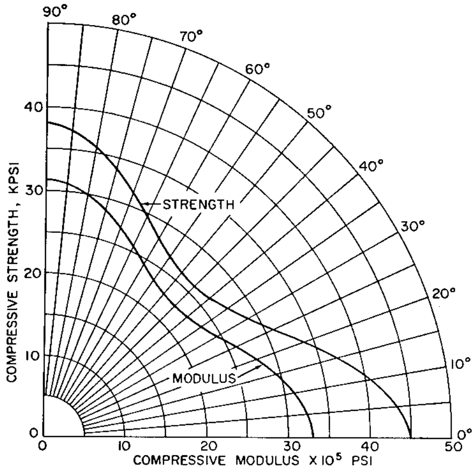


Figure D-4. Directional Compressive Strength and Modulus - 181/Epoxy

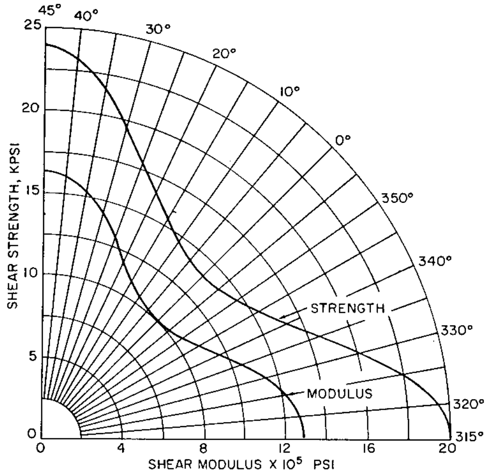


Figure D-5. Directional Shear and Modulus - 181/Polyester

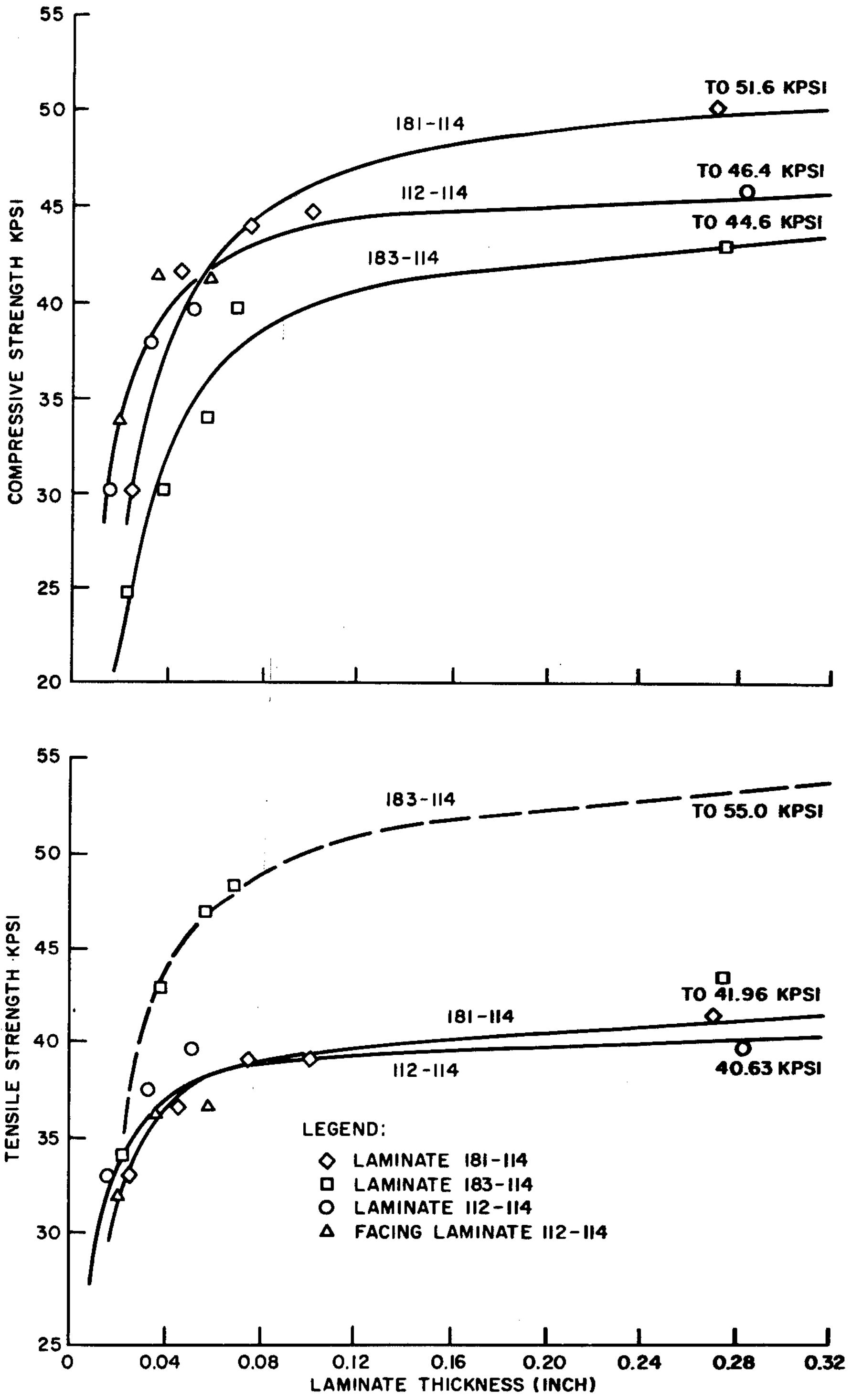


Figure D-6. Effect of Thickness on Strength of Four Laminates

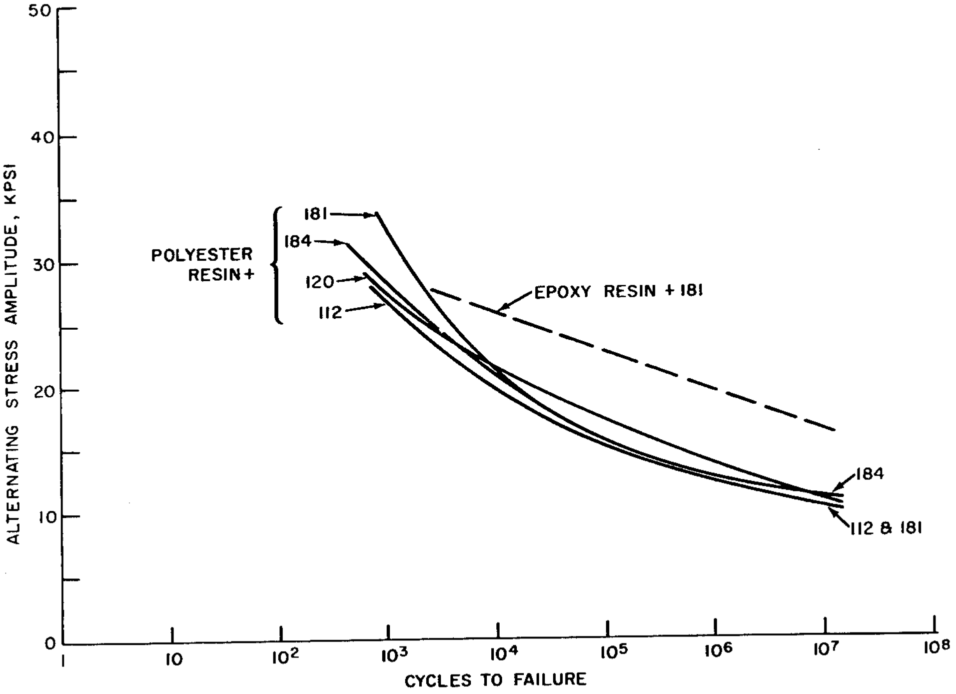


Figure D-7. SN Curves - Unnotched Laminates, Various Cloths

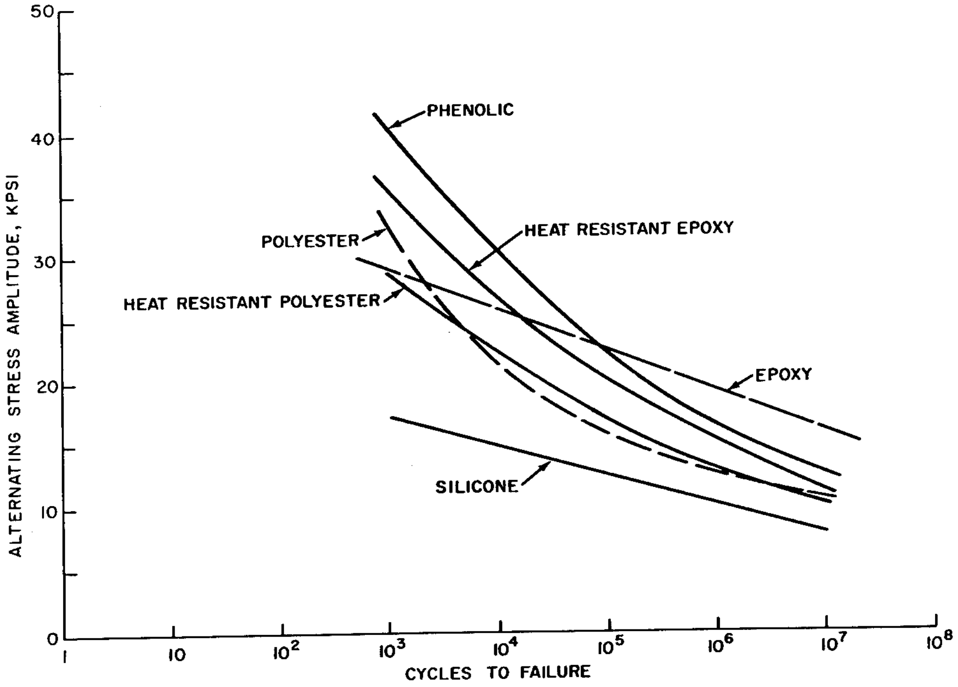


Figure D-8. SN Curves - Unnotched 181 Laminates, Various Resins

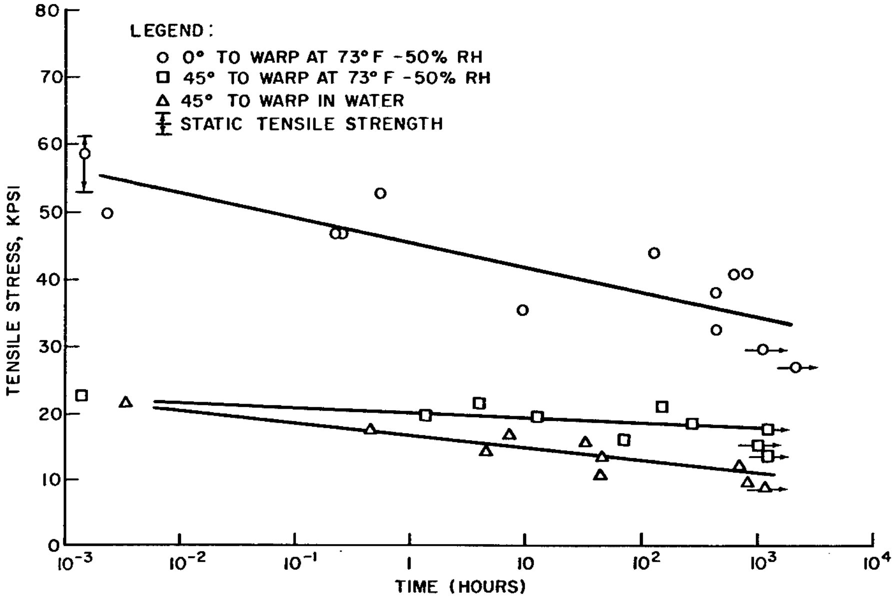


Figure D-9. Tensile Stress Rupture - 181-Volan A/Epoxy Lamintes

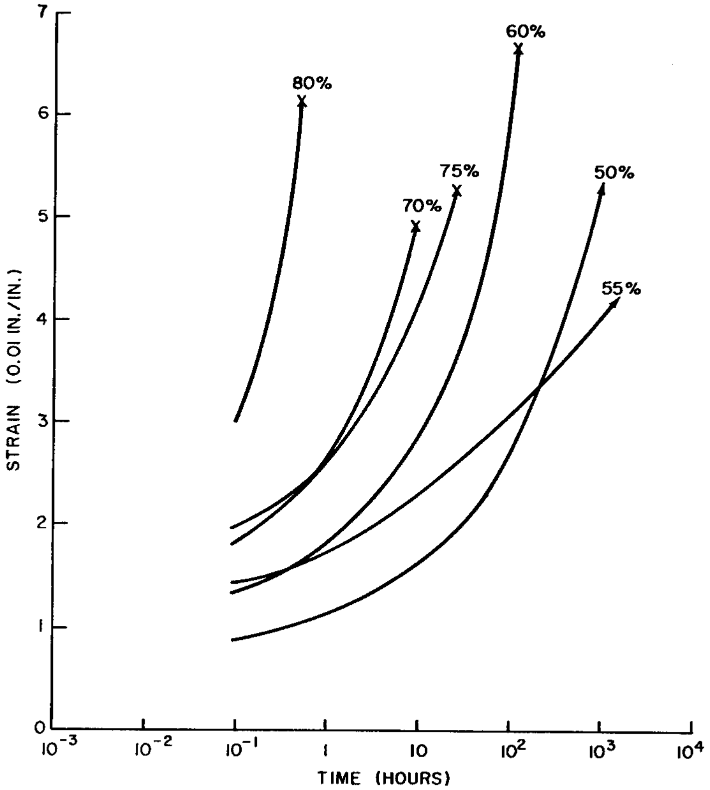


Figure D-10. Tensile Strain - Time Curve - 181-Volan A/Polyester Laminate, 45° to Warp, Various Percents, Static Strength

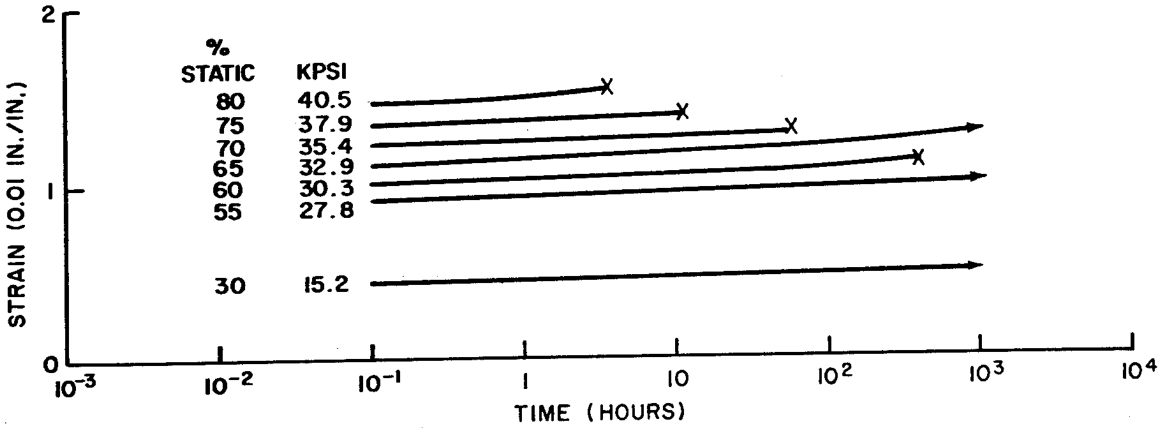


Figure D-11. **Strain - Time Curve - 181-Volan A/Polyester Laminates at Various Percents of Ultimate**

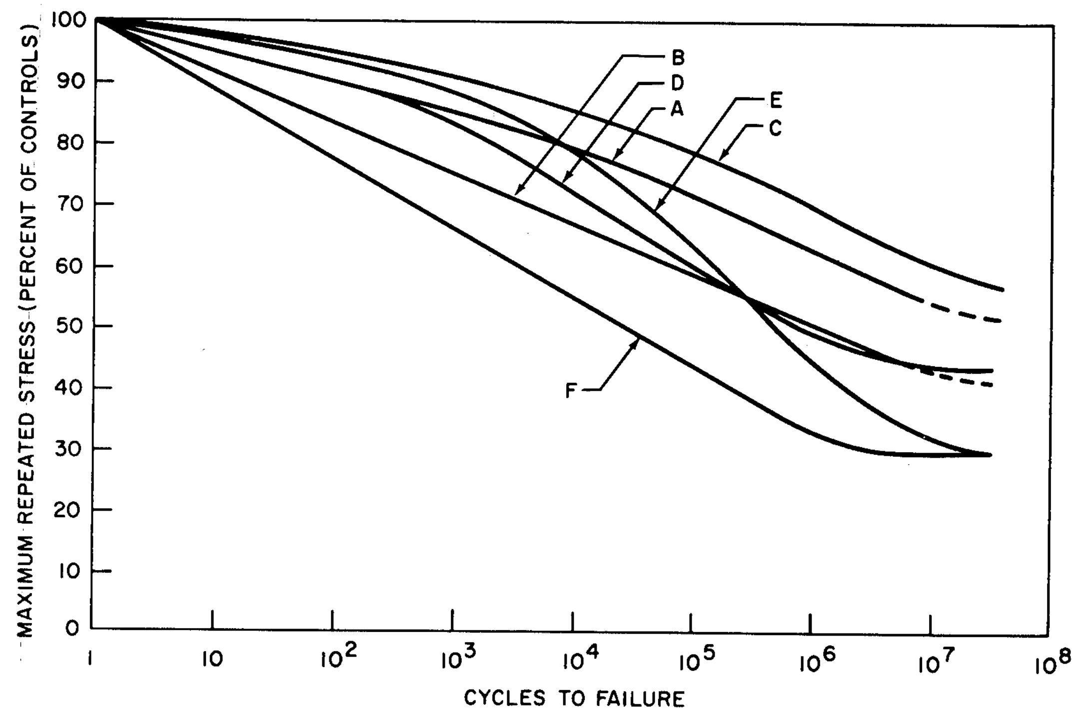
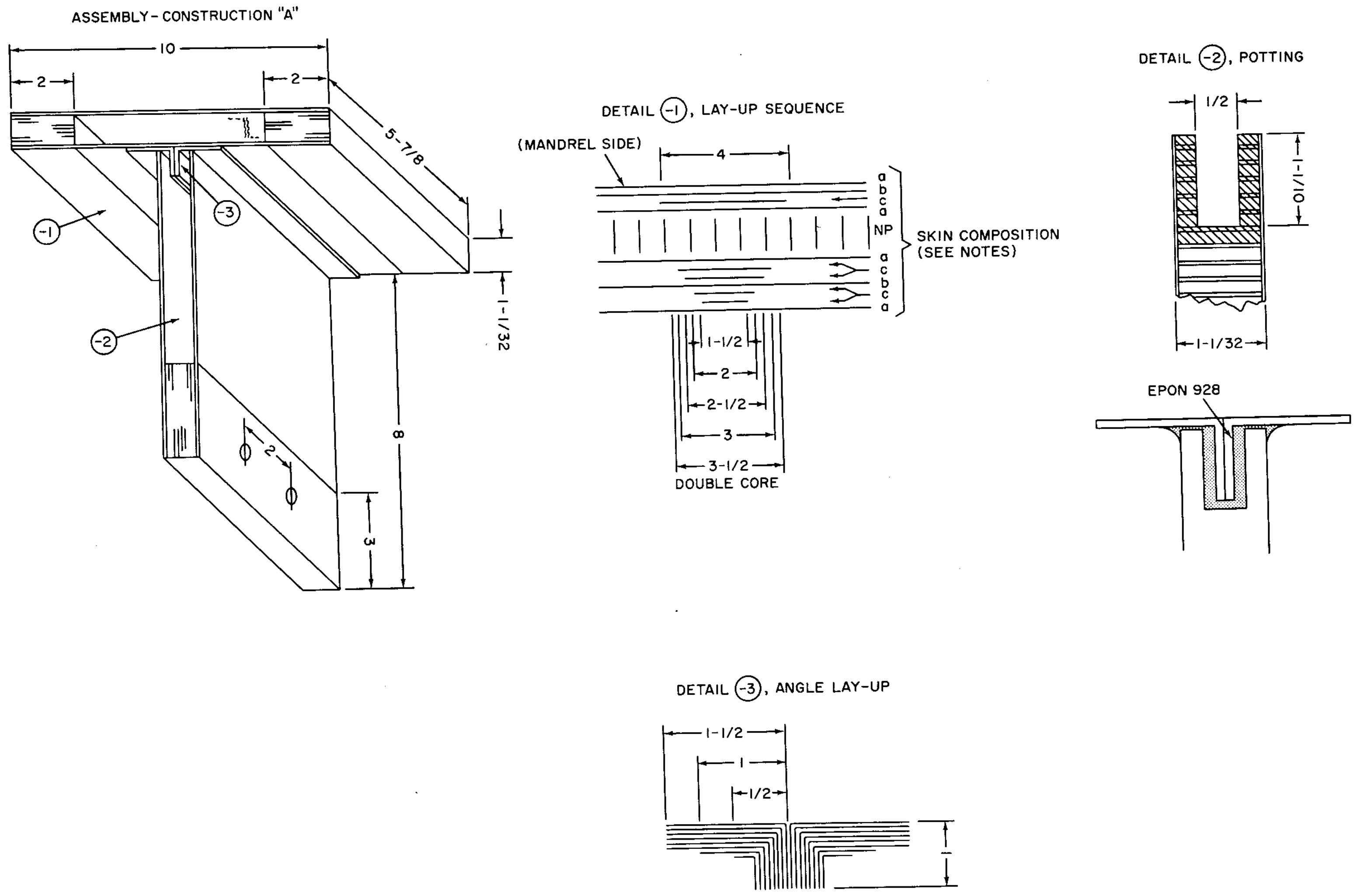


Figure D-12. SN Curves for Sandwich Materials

APPENDIX E BENCH TEST OF HANGER JOINT SANDWICH CONSTRUCTION
(Boeing Airplane Company)

Bench Test Behavior, Hanger Joint Construction A



NOTES:

Bench Test Behavior:

Ultimate load - 2500 pounds Core failed at area marked with jagged line in vertical shear. Core peeled from skin at area indicated. Panel had appearance of poor skin core bond (bag side).

Skin Compositions: Detail (-1)

a - 181 - 60% DER 332/MDA
b - 181 - 40% DER 332/MDA
c - 143 - 50% DER 332/MDA
NP - Nylon phenolic glass honeycomb
4#.core, 1/4" cell, 1" thick

(Skins aged 25 hours @ RT)

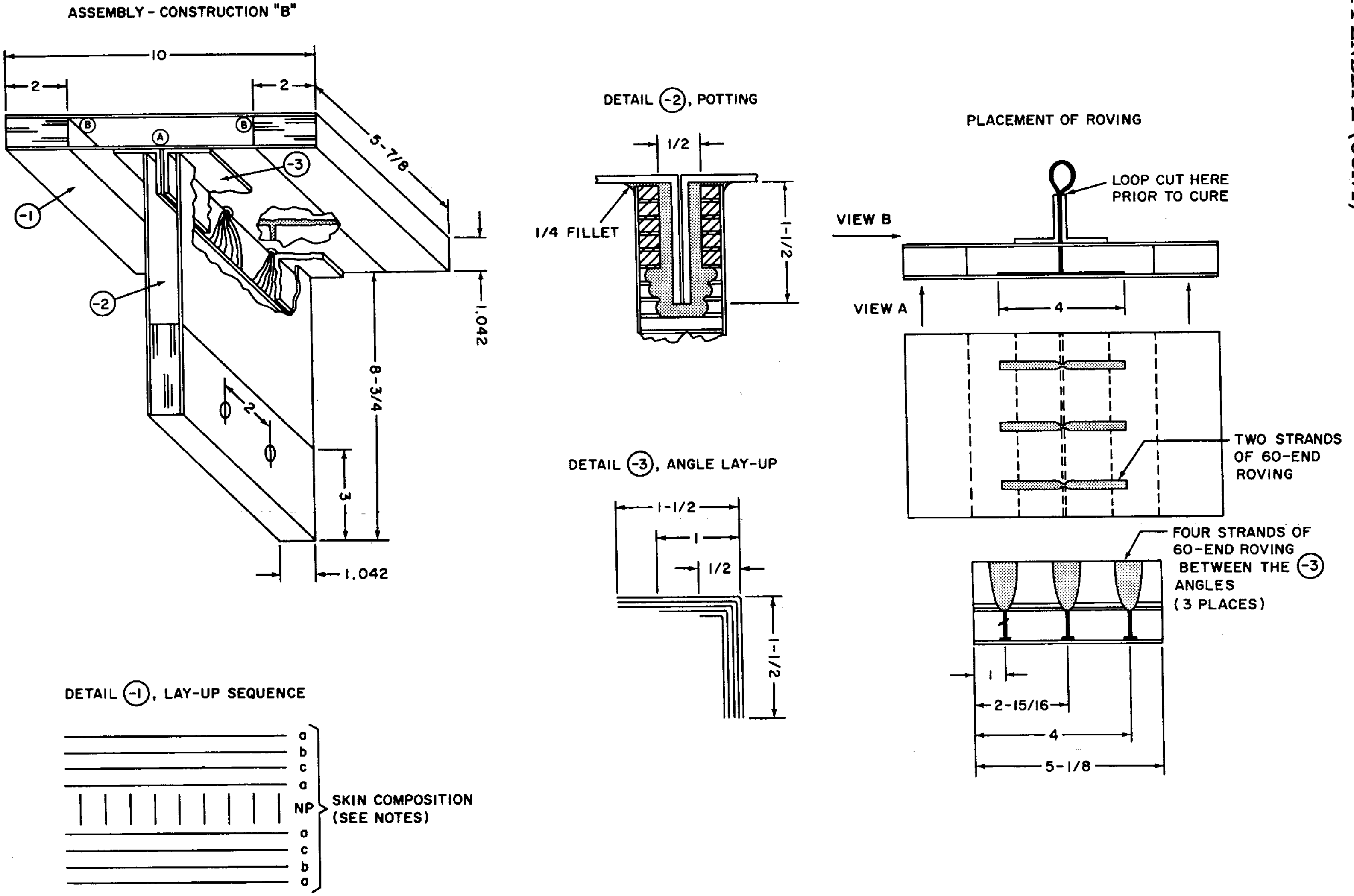
Potting : Detail (-2)

The shaded area was potted with BMS 5-28 material. The slot was filled with Epon 928 and (-3) was inserted into (-2). A 1/4 inch fillet of Epon 928 was formed between (-2) and (-3). The adhesive was allowed to harden overnight at room temperature, plus a cure of 2 hours at 120^{o} F.

Fabrication of Angles: Detail (-3)

Angles were prefabricated on matched metal molds and then cured in an autoclave at 45 psi with vacuum, 2 hours at 190^{o} F. After the removal of one peel ply from top and bottom, the angles were bonded back to back with Epon 928. (The peel ply was one layer of 128 cloth laid over each angle prior to match molding.) The (-1) piece was laid up wet with the exception of the solid edge sections. The (-3) angle was placed over the wet (-1) section as indicated. The assembly was covered with perforated PVA film followed by one dry ply of 181 glass fabric. The assembly was cured in an autoclave at 45 psi and vacuum (3 hours at 190^{o} F plus 4 hours at 350^{o} F.)

Bench Test Behavior, Hanger Joint Construction B



NOTES:

Bench Test Behavior:

Ultimate load - 2795 pounds. Core failed in shear along planes passing through point Ⓐ (located just under the (-3) angle), and points Ⓑ (located along the intersection of the solid edge bands and the upper skin).

Skin Compositions: Detail (-1)

Same as for Construction A, except:
Skins aged 30-35 hours at RT

Potting: Detail (-2)

The cross-hatched area was potted with BMS 5-28 material. The slot was filled with Epon 928 (shaded area) and (-3) was inserted into the slot. The adhesive was allowed to harden overnight at RT and was then cured for 2 hours at RT.

Fabrication of Angles: Detail (-3)

Angles contain 5 plies of 181 glass fabric plus 1 peel ply of 128 on both sides. Fabric was impregnated with 40% DER 332/MDA. Angle was cured on teflon coated matched metal tool-in autoclave at 45 psi, evacuated, for 2 hours at 190° F. The 128 glass cloth peel plies were removed, the backs of the angles were coated with the resin mix, the angles with roving between were placed on the uncured (-1) assembly, and all were cured in an autoclave at 45 psi, evacuated, for 2 hours at 190° F plus 4 hours at 350° F.

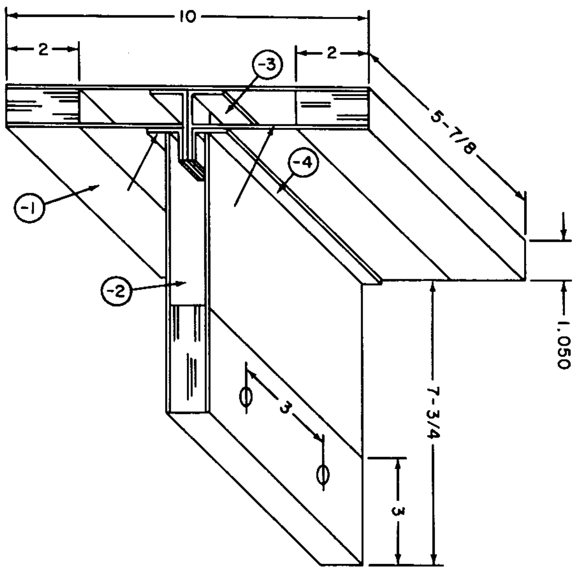
Placement of Roving:

Strands of 60-end glass roving were impregnated with DER 332/MDA and allowed to soak for approximately 3 hours. A loop was formed from two strands of the roving. The roving was pulled through the proper cell of the honeycomb sore, the upper end was flared and positioned between the (-3) angles and the lower end was placed beneath the core. No cutting of core was necessary.

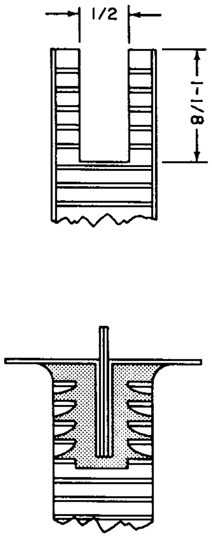
However, the structure is difficult to fabricate.

Bench Test Behavior, Hanger Joint Constructions C and D

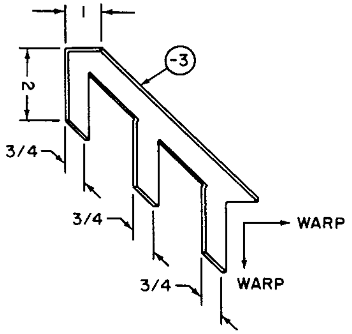
ASSEMBLY - CONSTRUCTION "C" & "D"



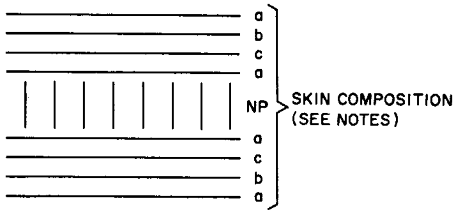
DETAIL -2, POTTING



DETAIL -3 AND -4, ANGLE LAY-UP

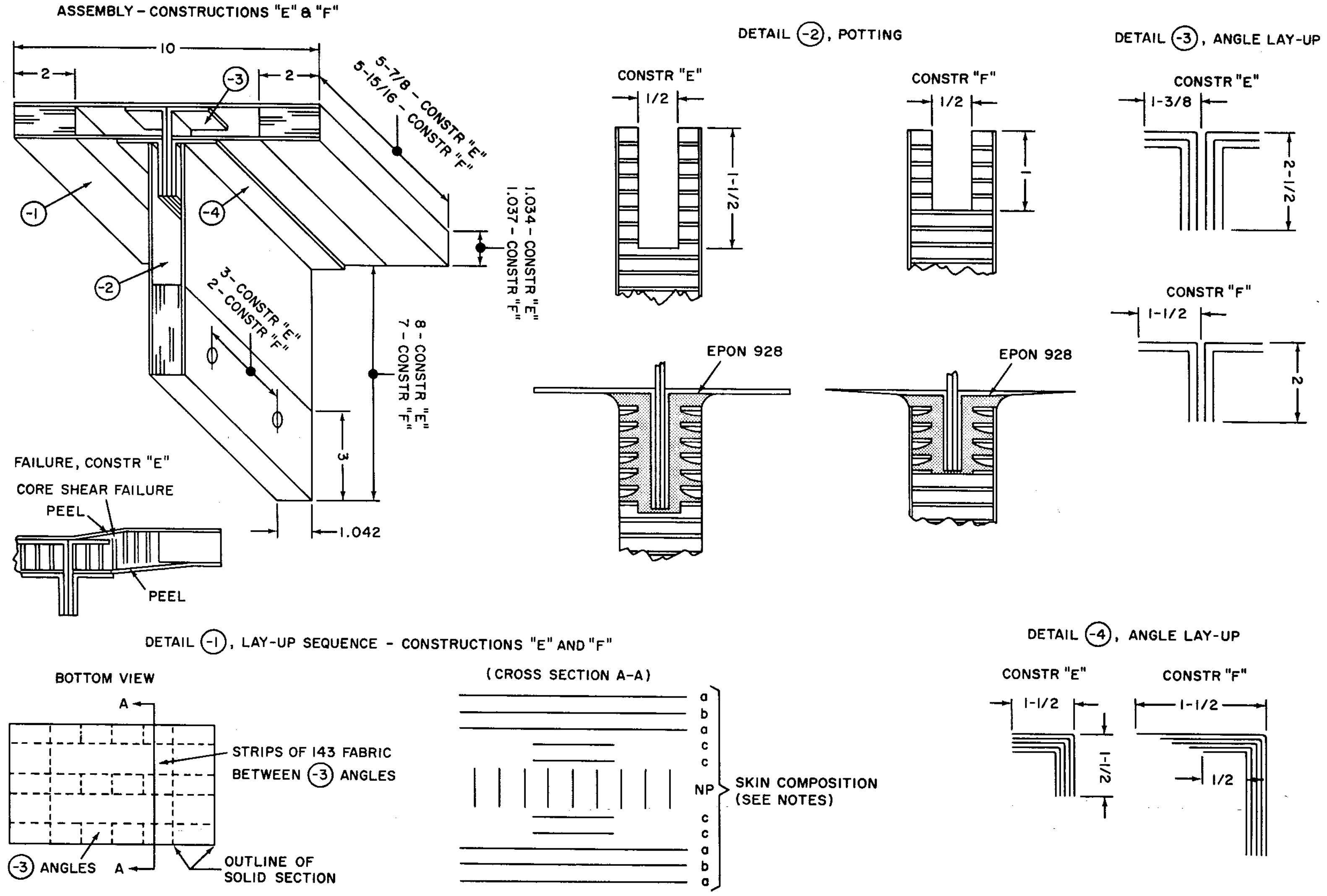


DETAIL -1, LAY-UP SEQUENCE



NOTES:	CONSTRUCTION "C"	CONSTRUCTION "D"
Bench Test Behavior:	Ultimate load - 2550 pounds Bond between (-2) and (-4) failed along the path indicated by arrow. This failure placed uneven stress on the T-section and skin of (-1) peeled from core from (-3) to solid section at area indicated by arrow.	Ultimate load - 1940 pounds The (-3) and (-4) angles were pulled apart at the center because of failure of the lower or inside skin of (-1). The (-1) skin was pulled in two.
Skin Composition: Detail (-1)	Same as Construction A, except: NP - 8# core, 1/2" cell, 1" thick Skins aged 30-35 hours at RT Two separate pieces of core used. Core butted against (-3) and (-4) angle, one piece on each side. (-1) and the (-3) and (-4) were cured in place in one cure operation. Cure in autoclave at 45 psi, vacuum, 2 hours at 190° F plus 4 hours at 350° F.	Same, except that the 8# density core in (-1) was not layered up in two separate pieces. Slots 3/4 inche wide were cut in the core to permit the flanges of (-3) to pass through.
Potting: Detail (-2)	A slot was cut in the edge of (-2) The slot was filled with Epon 928. (-2) was placed over the projections of (-3) and (-4). (-2) was held rigid and at right angles to (-1) in a bonding jig. The adhesive was allowed to harden overnight and was oven-cured at 120° F for 2 hours.	(-2) was bonded to (-1) only through the (-3) and (-4) projections. Parting agent was applied between (-1) and (-2) to prevent bonding between the horizontal surfaces.
Angle Lay-Up: Detail (-3) and -4	(-3) is the same as (-4) except the vertical leg of (-4) is 1 inch tall rather than 2 inches tall. Angles were fabricated on matched metal molds. Angles contain 2 plies of 143 glass fabric plus unimpregnated peel ply of 128 glass fabric on both sides. Fabric 143 impregnated with 50% DER 332/MDA. Cured in autoclave at 45 psi, evacuated, for 2 hours at 190° F. Peel plies removed from all surfaces. Two (-3) angles bonded back to back with Epon 928, cured under pressure for 2 hours at 120° F.	Same

Bench Test Behavior, Hanger Joints E and F



NOTES:	CONSTRUCTION "E"	CONSTRUCTION "F"
Bench Test Behavior:	Ultimate load - 3160 pounds Core failed in vertical shear at area marked Skins peeled from core. Core-skin bond appears to be low due to insufficient pre-impregnation of fabric. Sketch of failure shown.	Ultimate load - 3120 pounds. (-2) pulled away from (-1) leaving (-3) and (-4) angles intact. Failure may have started by the peeling of (-1) skin from core below the (-4) angle. Peeling appeared to have begun at areas where the skin was slotted for the (-3) angles.
Skin Composition: Detail (-1)	Same as Construction A, except: Core recessed to accept (-3) angles; also split to accept the (-3) angle. Double core 3-1/2" wide.	Same: except - Double core 3-1/2" long and 6" wide located at center. Skin aged 30 hours.
Potting: (Detail (-2)	A slot was cut in the edge of (-2). The slot was filled with Epon 928. (-2) was placed over (-4), which was held rigid and at right angles to (-1) in a bonding jig. The adhesive was allowed to harden overnight and was cured in an oven at 120° F for 2 hours.	Same
Angle Lay-Up: Detail (-3)	Angles prefabricated on matched metal molds. Angle contained 3 plies of 181 plus peel ply of 128 on both sides. Fabric impregnated with 40% DER 332/MDA, aged 25 hours at RT. Angles were cured in the autoclave at 45 psi and vacuum (2 hours at 190° F). Peel plies were removed and angles bonded back to back with Epon 928. Cure under pressure at RT, overnight. Angles prefabricated on matched metal molds.	Angles prefabricated on matched metal molds. Angles before cure contained 2 plies of 143 fabric impregnated with 50% DER 332/MDA, aged 30 hours at RT; plus one peel ply of 128 on each side, unimpregnated. Angles were cured in the autoclave at 45 psi and vacuum (2 hours at 190° F). Peel plies were removed and angles bonded back to back with Epon 928. Cured under pressure at RT overnight. Angles were 1" wide.
Detail (-4)	Angle contained 5 plies of 181 plus peel ply of 128 on both sides. Fabric impregnated with 40% DER 332/MDA, aged 25 hours at RT. Cured in autoclave at 45 psi and vacuum (2 hours at 190° F). Peel plies were removed.	Same, except resin aged 30 hours at RT.
Assembly Details: (-1) (-3) and (-4)	Aluminum plate (1/4" thick) was covered with PVA film. Three plies of impregnated 181 placed on mandrel followed by placement of the three (-3) angles. Strips of impregnated 143 placed between the (-3) angles. Extra section of glass core 3-1/2" wide pressed into core 6" wide. Ribbon direction of core parallel to 10" dimension. Slits cut into core to accept (-3), also bottom recessed for same purpose. Core and solid edge blocks placed on mandrel followed by strip of 143 fabric and three plies of 181. Resin (DER 332/MDA) applied to backs of (-4) angles, which were positioned on assembly.	Same; except add: The 181 fabric was slit to accept the (-3) angles. Add also: Perforated PVA film placed over assembly followed by piece of 181 fabric. Cured in autoclave at 45 psi and vacuum for 2 hours at 190° F and 4 hours at 350° F.

APPENDIX F

HISTORICAL REVIEW OF REINFORCED PLASTICS IN AIRCRAFT*

Date	Occurrence
1930	● Glass fiber research initiated by Owens-Illinois and Corning Glass Works.
1935	● The Owens-Corning Fiberglas Corporation was formed.
1940	● FRP industry started.
1941	● A task was set up so that plastics could be examined with the specific purpose to be used in aircraft wherever possible.
	● King Plastics Company, Denver, Colorado was given contract to fabricate plastic seats using combed and carded cotton fiber impregnated with urea resin cured at 2,000 psi.
	● Taylor Fibre received contract to fabricate BT-13 outer wing flap.
	● MacDonald Aircraft Company received contract to fabricate paper-phenolic structural wing box beam for the PT-19.
1942	● Structures and Materials Laboratory recognized a problem in curing of reinforced plastics at high pressure. One of the first contracts to develop a low pressure curing resin system went to Marco Chemical Company. Within approximately six months low pressure curing polyester resin systems were made available. In conjunction with this contract other resin manufacturers almost immediately had available for industry similar type resins (American Cyanamid Company, Bakelite Corporation, Monsanto Chemical Company, Pittsburgh Plate Glass Company, DuPont Company, Libbey Owens Ford Glass Company - Plaskon Division).
April 1942	● A program was set up, to cooperate with all of industry in collecting, at an accelerated rate, data on plastics for aircraft structural use which later would be issued in ANC Bulletins on Design Criteria (to become ANC 17 and 23).

---

*Information supplied by Whittaker Corporation - Narmco Division

Date	Occurrence
June 1942	● Based on material evaluations the best composition of reinforced plastics was determined to be glass fibers and low pressure curing polyester resins.
	● Owens-Corning Fiberglas Company received a contract to evaluate reinforced plastics.
	● By the end of 1942 various different fabricators throughout the country were producing important fiberglas parts for aircraft - (Swedlow Corporation, Lincoln Industries, U.S. Rubber Company, Goodyear Tire and Rubber Company, Formica Insulation Corporation).
1943	● Structures Laboratory personnel at Wright Field made the first fiberglas honeycomb by using large soda straws for the form.
	● Structures Laboratory started in-house projects to build primary structural aircraft parts for the following reasons: ● Relief to strategic materials. ● Low weight in fiberglas had potential to produce more efficient structures. ● Good electrical insulation and also transparent characteristics made reinforced ideal for radomes.
	● First successful major reinforced plastic structural component flight tested.
	● Glass fiber reinforced plastics were first conceived, developed, and designed for light airframe structures by the Air Force, Wright Air Development Center, Structures Laboratory and Materials Laboratory, Ohio in 1943. After analyzing test results on FRP, theoretical calculations indicated that an efficient structure could be designed and fabricated using high strength glass fiber-polyester resin laminate faces with low density core material. A survey of available military aircraft was performed to select some structural component which was reasonably well adapted to redesign in a sandwich structure. The aft section of the Vultee BT-15 basic trainer was selected. This component was completely redesigned and fabricated by the Air Force. The first designed concept was a balsa wood core with glass fiber reinforced plastic skins. These skins were made of five plies of three mil thick glass fabric impregnated with 42 - 45% by weight of polyester resin. The layup was rubber vacuum bag molded in a sheet metal female mold.

Date	Occurrence
1943	Cellophane was used to separate the rubber blanket from the inner skin. The static test performed on the first fabricated fuselage demonstrated the very high structural efficiency which had been predicted. From a strength to weight basis, the plastic sandwich structure was approximately 50% stronger than either the metal or wood type construction. In addition to its high structural efficiency, the sandwich section showed a remarkable absence of skin buckling under high torsional load. Severe buckling of aluminum skin would occur at 100% of design load. With the plastic structure at 180% of design load, there was relatively no visual or measurable skin buckling. In order to eliminate the use of wood in this structure, three other fuselages were fabricated using glass fabric honeycomb core.
1943	The properties of the reinforced plastics used in this part were as follows: tensile strength of 40,000 psi, compressive strength of 34,000 psi, flexural strength of 57,000 psi, shear strength of 19,000 psi and modulus of elasticity in flexure of 2,750,000 psi with a specific gravity of 1.8. The theoretical specific strength to weight ratios were higher than aluminum alloys and even the heat treated steels being used in structures. This structural potential was unfortunately drastically reduced by the relatively low modulus of elasticity for the materials. As a comparison, magnesium alloys with approximately the same specific gravity have a modulus of 6.5 million psi. The obvious solution in utilizing the reinforced plastic laminate in primary structures was to stabilize the material. Buckling would not occur until an appreciable portion of the compressive strength was developed. This stabilization was possible by the use of sandwich construction. Two high strength outer faces were separated and supported by bonding them to a much thicker, very light core. The core had to have only the necessary strength in tension, compression, and shear to adequately support the face materials to a high stress level. At that time, cores of suitable physical properties ranged in density from 6 to 10 pounds per cubic foot.
March 1944	● The BT-15 airplane with the plastic fuselage was first flown on March 24, 1944, at Wright-Patterson Air Force Base. This was considered the first successful major structural component of an airplane using reinforced plastics to be developed and flown.
	● The BT-15 airplane with plastic fuselage was flight tested under varying conditions and also varying temperature. Flight tests in low temperature environment were

Date	Occurrence
	conducted in 1946 at Ladd Field, Alaska. The first plastic fuselage tested involved 152 landings with no detrimental effects due to cold weather.
1944	● Aircraft wings for AT-6. In April 1944 preparation for the design and fabrication of the first experimental RP aircraft wing was started.
January 1945	● Actual fabrication of the first wing took place in January 1945 in the engineering shops of the Air Force Structures Laboratory. Since the wing is one of the most highly stressed major structures of an airplane, it represented a more difficult problem than the design of the BT-15 fuselage. The difficulty lay in developing a plastic wing to replace a component originally designed from metal. Obviously, a much more efficient plastic structure could be expected were the aircraft design originally intended for plastic construction, particularly at the attachment of the outer panel to the center section of the wing.
January 1945	The experimental outer panels represented a complete departure from the contemporary design. Instead of using the FRP similar to the standard metal practice of sheet stringer combinations utilized in conventional metal fuselage and wing design, a sandwich construction was used. Total number of FRP structural parts was 6 as compared to 100 parts and thousands of rivets in the metal section.
	Experimental wings were built and subjected first to static tests. The basic design used in the first three experimental wings was essentially the same except for the type of core material employed. In the first experimental wing panel, plain cellulose acetate core was used. This experimental panel passed the design loads requirement in the negative high angle of attack conditions; in the positive low angle of attack the wing failed at 60 percent ultimate design load. Failure of the wing was due to the core not being strong enough in tension to prevent the faces from delaminating. In the design of the second outer wing panel, the use of a 1/4 inch cell size honeycomb core was selected. The second wing panel was static tested in the positive low angle of attack condition and failed at 40% ultimate design load. This failure occurred because the honeycomb core had insufficient strength in the vertical shear plane. These two failures led to the development of using a wrapped acetate core. The wrapped acetate core consisted of 1/4 inch square

Date	Occurrence
	strips of cellular cellulose acetate wrapped spirally with one ply of polyester impregnated 112 glass cloth. This third wing core consisted of only these wrapped CCA strips.
September 1946	● On September 10, 1946 the third experimental wing satisfactorily passed static tests. On a strength to weight basis, this experimental wing was 13 percent stronger than the standard AT-6 metal. The basic materials used in these wings were: type 112 cloth and the then new low pressure curing polyester resins. Vacuum curing procedures were used with rubber or polyvinyl alcohol blankets.
March 1947	● An Air Force specification dated March 10, 1947 (X-26034) was prepared which provided detailed instructions for the fabrication of the AT-6 wing. East Coast Aeronautics, Inc., Pelham Manor, New York, received a contract to fabricate 10 wings. The specific purpose of this contract was to produce wings to be used for flight tests. By 1953 wings were flight tested. One of the wings was actually tested for over 7,000 hours - flight drag tests resulted with the wings producing an airspeed increase of eight knots at 200 knots when compared to standard aluminum airplane - the profile - drag coefficient was 25 percent less for FRP.
1944-1945	● Eagle wing-radar antenna - located below B-29 airplane main wing eroded and damaged during flights through Pacific rains - resulted in expediting developing elastomeric, rubber type, rain erosion coatings applied over radome surface.
March 1948	● Contract was awarded Douglas Aircraft Company to design and fabricate an outer wing panel for the C-54A - FRP replaced metal in a feasibility study to use integral wing antenna which would also be part of the structure.
	● A.F. Structures Lab started project to design, develop and fabricate primary structural aircraft parts made of FRP for use on supersonic aircraft and missiles.
April 1948	● A.F. Structures Lab awarded Narmco contract to design and fabricate P-61 airplane tail booms of FRP sandwich.
	● A.F. Structures Lab awarded Goodyear Aircraft Corporation contract to design and fabricate AT-6 airplane horizontal stabilizers.

Date	Occurrence
May 1948	● Successful FRP flight tests completed on parts for P-86 airplane-nose intake duct, radar dome, dorsal fin, integral antenna in vertical stabilizer and wing tips.
September 1948	● F-86A FRP nose intake duct and forward portion of FRP wing tips eroded during flight through rain- future parts protected by means of applying elastomeric rain erosion coating.
1950	● Epoxy resins synthesized. FRP used in secondary aircraft compound curvature structures; i.e., engine inlets, tip fuel tanks, wing tips and engine cowls.
	● FRP helicopter and aircraft blades developments emphasized due to unique fatigue characteristics - Bell Aircraft, Kaman Aircraft, Curtiss-Wright and Hamilton Standard.
	● FRP armor plate development programs applicable aircraft conducted.
	● FRP filament winding studied.
	● FRP droppable aircraft fuel tanks developed and produced.
	● Lockheed Constellation used the popular and large production 80 ft. diameter radome on the top side as well as large tub shaped belly radome.
1952	● AT-6 sandwich constructed horizontal stabilizer successfully passed static and flight tests.
1955	● Taylorcraft Model 20 airplane used FRP in wings, engine cowling, doors, seats, fuel tanks, instrument panels, fuselage coverings.
	● Vertol H-21 helicopter produced lower cost - equally efficient FRP in fuselage.
1958	● Piper Aircraft started investigation of FRP for primary structures - FRP airplane flew in 1962
1959	● Fairchild surveillance AN/USD-5 drone used FRP integral fuel tank - wings.
1960	● Boeing 727 jet airplanes each contain 5,000 lbs. of FRP lower cost parts and 33 percent lower in weight.

Date	Occurrence
1960	● Convair's C-141 leading edge for the horizontal stabilizer reduced the metal tail weight 100 lbs. - design includes integral de-icing system.
	● Douglas DC-8 jetliners each contain 2,000 lbs of FRP - which includes unique structural parts of spar and vertical tail section.
	● High strength and high modulus fibers being developed - (S-glass, Boron).
	● FRP structural components for high performance aircraft - B-58, DC-8, F8U and C-141.
	● Grumman Aircraft Corp. Hawkeye E-2a used rotating 15 ft. diameter radome located above wing - actually aids aircraft lift.
	● Boeing B-52 uses relatively large FRP parts, such as wing tips.
	● North American Aviation stabilizers for T-2A airplane.
	● Ryan Co. transonic Q-2C target missile uses 42 FRP components.
	● Mississippi State University Marvelette airplane used in conducting laminar boundary layer control flight tests.
	● Piper Aircraft Co. flew fiberglas airplane.
	● Jet transport each using approximately 2-1 tons of FRP - access doors, fairings, radomes, tail cones, etc.
	● Kaman Aircraft Corp. helicopter model HH-43B all fiberglas blades had successful flight tests - offered major advances when compared to metals in fatigue resistance as well as reducing manufacturing and servicing costs.

BIBLIOGRAPHY

NOTE: Reports cited are not available from PLASTEC.

Ref. No.

1 Abelin, Rudolf and E. A. Wohlberg (Malmo Flygindustri, Sweden) "Glassfiber reinforced plastics in highly stressed aircraft parts." In: PROCEEDINGS, 17TH ANNUAL MEETING, REINFORCED DIVISION, the Society of the Plastics Industry, Chicago, February 1962. Section 2-A

2 Aerojet-General Corporation, Azusa FEASIBILITY OF FILAMENT-WOUND AIRCRAFT WING CONSTRUCTION, Proposal No. SM-63131, to Bureau of Naval Weapons, April 1964 (Proprietary)

3 Aerojet-General Corporation, Sacramento DEVELOPMENT OF IMPROVED RESIN SYSTEMS FOR FILAMENT WOUND STRUCTURES, by J. J. Cooney. Task VI, Final Report 0627-VI-F, August 1964. Contract No. 63-0627c (FBM) (BuWeps)

4 Aerojet-General Corporation, Von Karman Center PRIMARY AIRCRAFT COMPONENTS OF FILAMENT-REINFORCED PLASTIC, a proposal to Research and Technology Division, Air Force Systems Command. Proposal No. SM-64082, July 1964. (Proprietary)

5 Aerojet-General Corporation, Von Karman Center FILAMENT-WOUND COMPRESSION PANELS FOR AIRCRAFT WING CONSTRUCTION, by J. C. Ekvall. Report No. 2885 (Special), July 1964 (Proprietary)

6 Aerojet-General Corporation, Von Karman Center STRUCTURAL MATERIALS HANDBOOK (Proprietary)

7 Air Force Materials Laboratory, WPAFB PLASTICS IN THE AIR FORCE, by D. L. Schmidt and R. C. Tomashot. Report ML-TDR 64-76, April 1964. AD 601 555

8 Air Forces Materials Laboratory CONFERENCE ON STRUCTURAL PLASTICS, ADHESIVES, AND FILAMENT WOUND COMPOSITES, 11-13 December 1962 Sidney Litvak, Chairman. Technical Documentary Report No. ASD-TDR-63-396, April 1963

9 Air Force Materials Laboratory, WPAFB HIGH MODULUS GLASS FIBERS FOR REINFORCED PLASTICS, by G. P. Peterson. WADD Technical Report No. 60-735, September 1961. (Compilation of papers and general discussion, ASD/Univ of Dayton Joint Meeting on High Modulus Fibers for Reinforced Plastics, 12 October 1960.)

Ref. No.

10 Air Force Materials Laboratory, WPAFB
Wright Air Development Center/University of Dayton
STRUCTURAL ADHESIVES AND SANDWICH CONSTRUCTIONS, WADC/UNIVERSITY OF DAYTON JOINT SYMPOSIUM, November 1957. Administered by T. J. Martin.
WADC Technical Report 57-512, November 1957. AD 142 152

11 Air Material Command, WPAFB
EVALUATION OF SYNTHETIC CORE MATERIALS FOR SANDWICH TYPE AIRCRAFT STRUCTURES,
Engineering Division Memo Report No. TSEAL2-45119-1-11, 1 March 1945

12 Air Material Command, WPAFB
Engineering Division Memo Report No. ENG-56-M4690, 20 December 1943.

13 Aleksandrov, A. Y., L. E. Bruykker, and others
DESIGNING SANDWICH PANELS, FTD-TT-62-206, 15 June 1962. Unedited rough draft translation, Foreign Technology Division, WPAFB.

14 American Aviation Publications, Inc.
"Annual world aviation encyclopedia." In: AMERICAN AVIATION, Vol 28, No. 5, October 1964.

15 Anderson, J. A. and J. A. McCarthy (3M, Reinforced Plastics Division)
"Prepreg reinforced plastics in fatigue applications."
In: PROCEEDINGS, 18TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1963. Section 8-C

16 Anonymous
"New RP technology gives the 727 a big lift." In: MODERN PLASTICS, August 1964, p. 101 to 104

17 Anonymous
"Military aircraft - facts, figures - and 167 'Flight Internations' scale drawings." In: FLIGHT INTERNATIONAL, 20 August 1964

18 Anonymous
"More lift with RP prop." IN: MODERN PLASTICS, May 1963

19 Anonymous
"Plastics in aircraft." In: BRITISH PLASTICS, August 1962

20 Armed Services Technical Information Agency (Now Defense Documentation Center)
H-16 and H-21 HELICOPTERS, A REPORT BIBLIOGRAPHY, ARB No. 11005, July 1962. AD 447 051

21 Armour Research Foundation
MECHANISM OF REINFORCEMENT OF FIBER-REINFORCED STRUCTURAL PLASTICS AND COMPOSITES, by J. S. Islinger, K. Gutfreund, I. M. Daniel and A. J. Durelli. WADC Technical Report 59-600, Part III, April 1961. Contract AF 33(616)-6987 (Materials Central)

Ref. No.

22 Army Air Forces
MOLDED GLASS FIBER SANDWICH FUSELAGES FOR BT-15 AIRPLANE, by G. B. Rheinfrank Jr., and W. A. Norman. Technical Report No. 5159, 8 November 1944. OTS-PB-2293.

23 Avco Corporation
PROPOSED SPECIFICATION - NONDESTRUCTIVE DETECTION OF UNBOND IN COMPLEX HONEYCOMB STRUCTURE Unnumbered, undated. ULTRASONIC CONTACT TESTING OF FIBERGLASS PLATES, PROCEDURE FOR, RAD-P58067. NOTES ON PREPARATION AND APPLICATION OF NONDESTRUCTIVE TEST SPECIFICATIONS FOR PLASTICS, unnumbered, undated. (Proprietary)

24 Baer, Eric
ENGINEERING DESIGN FOR PLASTICS. Reinhold Publishing Corporation, New York, 1964.

25 Bell Aerosystems Company, Buffalo
THERMAL STRESS DETERMINATION TECHNIQUES FOR SUPERSONIC TRANSPORT AIRCRAFT STRUCTURES, PART II - DESIGN DATA FOR SANDWICH PLATES AND CYLINDERS UNDER APPLIED LOADS AND THERMAL GRADIENTS, by R. A. Gellatly and R. H. Gallagher. Report No. ASD-TDR-63-783, Part II, January 1964. Contract AF 33(657)-8936 (Aeronautical Systems Division) AD 431 959

26 Boeing Company, Airplane Division
HIGH PERFORMANCE GENERAL PURPOSE ATTACK AIRCRAFT STUDY, by R. W. Rockwell. Document No. D6-8635, July 1964. Contract NOmr-4536 (00) (BuWeps and Office of Naval Research) CONFIDENTIAL REPORT

27 Boeing Company, Airplane Division, Renton
LONG RANGE HEAVY LOGISTICS TRANSPORT SYSTEM AIRFRAME AND PROPULSION STUDY PROGRAM (CX-HLS) Document No. D6-10482-2, September 1964. Final Report, Task No. 3. Contract AF 33(615)-1923. REPORT CONFIDENTIAL

28 Boeing Company (Transportation Division)
"A report on structural adhesive bonding." In: BOEING AIRLINER, February 1959.

29 Boeing Company, Seattle
FILAMENT-WOUND CYLINDERS UNDER AXIAL, TORSIONAL, AND PRESSURE LOADS, by R. E. Sherrer and C. W. Harter. Document No. D2-23221, November 1964. (Proprietary)

30 Boeing Company, Vertol Division
MANUFACTURE OF POSITIVE PRESSURE CURED, GLASS FIBER FACED, ALUMINUM HONEYCOMB CORE, SANDWICH ASSEMBLIES, by G. W. Campbell. Report No. D8-0106, 24 November 1964.

Ref. No.

31 Boeing Company, Vertol Division
THE DEVELOPMENT OF "FIBERGLAS" REINFORCED PLASTIC BLADES FOR CH-47A HELICOPTER, by T. Tarczynski. Reprint, VERTI-FLITE, October 1963.

32 Boeing Company, Vertol Division
HELICOPTER ROTOR BLADE EROSION PROTECTIVE MATERIALS. TCREC Technical Report 62-111, December 1962. Contract DA 44-177-TC-836 (Army Transportation Research Command) AD 411 780

33 Boeing Company, Vertol Division
HELICOPTER ROTOR BLADE EROSION PROTECTIVE MATERIALS, Appendix V - Master Code List for all Metals and Nonmetals Subjected to Flat-Panel Sand Impingement Tests. TCREC Technical Report 62-111, December 1962. Contract DA 44-177-TC-836. AD 412 502

34 Boeing Company, Vertol Division
YHC-1B PLASTIC BLADE DEVELOPMENT, by T. Tarczynski and V. Hardy Report No. R-301, October 1962

35 Boeing Company, Vertol Division
BENCH FATIGUE TEST OF YHC-1B PLASTIC ROTOR BLADE ROOT END SECTION, by J. Nonemaker. Report No. R-282, 16 May 1962

36 Boeing Company, Wichita
QUALIFICATION OF SOURCES FOR STRUCTURAL REINFORCED PLASTIC PARTS, by L. Soos, Jr. Document No. D-16538, 22 May 1956; revised by W. D. Boyle, 14 January 1964 (Proprietary)

37 Boeing Company, Wichita
STATIC LOAD DESTRUCTION TEST OF ST-2413 AIRFOIL ASSEMBLIES, by C. H. Price. Test Report No. ST-62-11, 9 March 1962. (Proprietary)

38 Boeing Company, Wichita
FABRICATION PROCESS FOR ADHESIVE BONDING, by J. J. Elton. Document No. D3-4077, 7 November 1961. (Proprietary)

39 Boeing Company, Wichita
B-52 A-F CHIN RADOME INVESTIGATION, by J. J. Elton. Document No. D3-3643, 25 April 1961. (Proprietary)

40 Boeing Company, Wichita
MECHANICAL AND PHYSICAL TESTING PROCEDURES FOR STRUCTURAL PLASTICS, by G. D. Nichols. Document No. D3-3189, 2 September 1960 (Proprietary)

41 Boeing Company, Wichita
BETA BACKSCATTER RESIN CONTENT AND THICKNESS, by C. E. Kimball. Report No. D3-2834, 18 January 1960 (Proprietary)

42 Boeing Company, Wichita
COMPARATIVE WEATHER RESISTANCE OF LAMINATES. Report No. AP-3-7, August 1959

Ref. No.

43 Boeing Company, Wichita
RELATIVE RESISTANCE OF EIGHT POLYESTER RESINS TO HIGH HUMIDITY AND TO ELEVATED TEMPERATURES, by E. W. Morgan, Jr. Document No. D3-1686, 11 June 1958.

44 Boeing Company, Wichita
RESISTANCE OF VARIOUS LAMINATES TO ELEMENTAL EFFECTS. Final Summary Report 4, Contract AF 33(616)-8141. Including reports: BAC 5426, BAC 5449, D3-1686, AP-3-7, AP-2-7, P-2-341, P-2-381.

45 Boyd, A. R. and L. D. Moore (Ferro Corporation)
"Properties and application of nonwoven unidirectional materials." In: PROCEEDINGS, 18TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1963. Section 10-D

46 Brookfield, K. J. and D. Pickthall (Fiberglass Ltd.)
"The importance of the resin system and the method of cure on the ultimate strength developed in wound structures." In: PROCEEDINGS, 18TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1963. Section 20-B

47 Brunswick Corporation
AN EVALUATION OF NEW AND POTENTIALLY HEAT RESISTANT GLASS REINFORCED PLASTICS. QPR No. 1 (1 June - 31 August 1964), by S. A. Moorefield, 4 September 1964. BPSN: 64-6899-7381-738101. Contract AF 33(615)-1610 (Air Force Materials Laboratory)

48 Brunswick Corporation, Defense Products Division.
POLYIMIDE GLASS FABRIC LAMINATES, Technical Bulletin No. 1003 GLASS REINFORCED ALUMINUM PHOSPHATE, Technical Bulletin No. 1005

49 Brunswick Corporation, Defense Products Division
DEVELOPMENT OF A 1200°F RADOME, by V.A. Chase and R. L. Copeland. Unpublished report; on Contract AF 33(657)-11469 (Air Force Avionics Laboratory)

50 Burge, R. E. and S. H. Christie (Shell Chemical Company)
"New epoxy resin systems for filament winding." In: PROCEEDINGS, 19TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1964. Section 14-E

51 Catchpole, E. J. (Bonded Structures, Ltd., Cambridge, England)
DEVELOPMENT OF SYNTHETIC RESIN ADHESIVES. Reprint, Technical and Management Papers Presented at the 20th Annual Conference of the Society of the Plastics Industry - Western Section, Palm Springs, May 1963

52 Chemical and Engineering News
THE GLASS INDUSTRY, a C & EN Special Report, 16 November 1964

Ref. No.

53 Dallas, R. N. (Lockheed Aircraft Company)
"Filament winding the Lockheed Star Fighter radome." In: PROCEEDING, SAMPE FILAMENT WINDING CONFERENCE, 28-30 March 1961, Pasadena

54 Data Publications
ARMY AVIATION; Feature Activity Report; Vol 9, No. 8, August 1964.

55 Davis, J. R. (Dama Plastics, Inc.)
"A helicopter blade - a composite plastics construction." In: PROCEEDING, FOURTH INTERNATIONAL REINFORCED PLASTICS CONFERENCE, British Plastics Federation, Cafe Royal, London. 25-27 November 1964.

56 Davis, J. W., J. A. McCarthy and J. N. Schurb (3M, Reinforced Plastics Division)
"The fatigue resistance of reinforced plastics." In: MATERIALS IN DESIGN ENGINEERING, December 1964. Vol 60, No. 7. p. 87 ff.

57 DeBell and Richardson, Inc.
STUDIES OF HOLLOW MULTIPARTITIONED CERAMIC STRUCTURES, by W. J. Eakins and R. A. Humphrey.
Final Report, 7 April 1963 to 15 September 1964.
Contract NASw-672 (NASA)

58 Defense Documentation Center
RAIN EROSION OF MATERIAL IN SPACECRAFT AND AIRCRAFT. A CUSTOM ABSTRACT SEARCH (U), by Gordon Willey DDC ARB-16 993, June 1963. CONFIDENTIAL REPORT
AD-337 131

59 Defense Documentation Center (Formerly Armed Services Technical Information Agency) COMPOSITE MATERIALS FOR AIRPLANE CONSTRUCTION, a Report Bibliography, October 1962.
ARB No. 12, 189. AD 447 040

60 Douglas Aircraft Company, Inc.
STRUCTURAL DESIGN FOR ACOUSTIC FATIGUE, by M. J. Cote. Report No. ASD-TDR-63-820, October 1963. Contract AF33(657)-8217 (Air Force Flight Dynamics Laboratory) AD 425 406

61 Douglas Aircraft Company, Inc., Aircraft Division
CX-HLS STUDY PROGRAM, FINAL REPORT. PART IV (U) ANALYSIS OF RECOMMENDED DESIGN - TASK 3. VOLUME V (U) STRUCTURE AND MATERIALS, by H. W. Adams. DAC Report 47582, Part IV, Vol V, 14 September 1964. Contract AF 33(615)-1924 (Research and Technology Division, WPAFB).
CONFIDENTIAL REPORT

62 Douglas Aircraft Company, Inc., Missile and Space Systems Division
ELASTIC CONSTANTS AND ANALYSIS METHODS FOR FILAMENT WOUND SHELL STRUCTURES, by L. B. Greszczuk.
SM-45849, January 1964.

Ref. No.

63 Douglas Aircraft Company, Inc., Missiles and Space Systems Division
THE EVALUATION OF HEXCEL RRP HONEYCOMB CORE FOR FLATWISE COMPRESSION STRENGTH, by F. J. Schneider and N. N. Doyle. Report MP 1744, 12 December 1963.

64 Douglas Aircraft Company, Inc.
SONIC FATIGUE DAMPING MATERIALS, by P. R. McGowan and J. M. Snider. Final report LB-31451, September 1963. Contract NOw-62-1071c. AD 600 170

65 The Dow Chemical Company, Texas Division
PLASTICS IN AIRCRAFT, by E. L. Pendleton and L. J. Windecker. Report No. TOP 549-1. December 1964. (Proprietary)

66 Dyna/Structures Incorporated
BIBLIOGRAPHY OF METHODS OF STRUCTURAL OPTIMIZATION FOR FLAT SANDWICH PANELS, by J. R. Vinson and S. Shore Report No. 64-N-001, December 1964.

METHODS OF STRUCTURAL OPTIMIZATION FOR FLAT SANDWICH PANELS, by J. R. Vinson and S. Shore. Report No. 64-N-002, December 1964.

DESIGN PROCEDURES FOR STRUCTURAL OPTIMIZATION OF FLAT SANDWICH PANELS, by J. R. Vinson and S. Shore. Report No. 64-N-003, December 1964.

Contract N-56-44970 (Naval Air Engineering Center)

67 Eastwood, N.
"Fluted-core fabrics." In: TRANSACTIONS AND JOURNAL - THE PLASTIC INSTITUTE. Vol 32, No. 100, August 1964, p 261 - 264.

68 Fechek, Frank and Robert Tomashot (Aeronautical Systems Division, WPAFB)
"Reinforcements - Air Force approach to planned composite properties." In: PROCEEDINGS, 18TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1963. Section 4-E

69 Fetherston, W. H. (The Boeing Airplane Company)
"Reinforced plastics and typical application on Boeing Transport planes." In: PROCEEDINGS, 17TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry. Chicago, February 1962

70 Fischer, Lawrence (Grumman Aircraft Engineering Corporation)
"Design of glass-reinforced plastic structures." In: PROCEEDINGS, 15TH ANNUAL TECHNICAL AND MANAGEMENT CONFERENCE, Reinforced Plastics Division, the Society of the Plastics Industry, Inc., Chicago, February 1960. Section 3-D.

Ref. No.

71 Forest Products Laboratory
DESIGN CRITERIA FOR LONG CURVED PANELS OF SANDWICH CONSTRUCTION IN AXIAL COMPRESSION, by E. W. Kuenzi. Report No. 1558, January 1959.

72 Forest Products Laboratory
FATIGUE OF SANDWICH CONSTRUCTIONS FOR AIRCRAFT; FIBERGLAS-HONEYCOMB CORE MATERIAL WITH FIBERGLAS-LAMINATE OR ALUMINUM FACINGS, TESTED IN SHEAR, by Fred Werren. Report No. 1559-C, March 1956. Information reviewed and reaffirmed, 1962.

73 Forest Products Laboratory
CELLULAR-HARD-RUBBER CORE MATERIAL WITH ALUMINUM OR FIBERGLAS-LAMINATE FACINGS, TESTED IN SHEAR, by Fred Werren. Report No. 1559-E, October 1948; Information reviewed and reaffirmed, 1962.

74 Forest Products Laboratory
FATIGUE OF SANDWICH CONSTRUCTIONS FOR AIRCRAFT; ALUMINUM FACING AND ALUMINUM HONEYCOMB CORE SANDWICH MATERIAL TESTED IN SHEAR, by Fred Werren. Report No. 1559-H, March 1956. Information reviewed and reaffirmed, 1962

75 Forest Products Laboratory
FATIGUE OF SANDWICH CONSTRUCTIONS FOR AIRCRAFT, by Fred Werren Report No. 1559-J, April 1952. Information reviewed and reaffirmed, 1958.

76 Forest Products Laboratory
DIRECTIONAL PROPERTIES OF GLASS-FABRIC-BASE PLASTIC LAMINATE PANELS OF SIZES THAT DO NOT BUCKLE, by Fred Werren and C. B. Norris. Report No. 1803, March 1956

77 Forest Products Laboratory
DIRECTIONAL PROPERTIES OF GLASS-FABRIC-BASE PLASTIC LAMINATE PANELS THAT DO NOT BUCKLE (Supplement to), by Fred Werren. Report 1803-A. Information reviewed and reaffirmed March 1956.

78 Forest Products Laboratory
DIRECTIONAL PROPERTIES OF GLASS-FABRIC-BASE PLASTIC LAMINATE PANELS OF SIZES THAT DO NOT BUCKLE (Supplement to), by A. D. Freas and Fred Werren. Report No. 1803-B, November 1955.

79 Forest Products Laboratory
DIRECTIONAL PROPERTIES OF GLASS-FABRIC-BASE PLASTIC LAMINATE PANELS OF SIZES THAT DO NOT BUCKLE (Supplement to), by Fred Werren and Marvin Gish. Report No. 1803-C, May 1957.

80 Forest Products Laboratory
EFFECT OF DEFECTS ON STRENGTH OF AIRCRAFT-TYPE SANDWICH PANELS, by B. G. Heebink and A. A. Mohaupt. Report No. 1809, March 1956.

Ref. No.

81 Forest Products Laboratory
EFFECT OF DEFECTS ON STRENGTH OF AIRCRAFT-TYPE SANDWICH PANELS, by A. A. Mohaupt and B. G. Heebink. Report No. 1809-A, November 1951. Information reviewed and reaffirmed, 1958

82 Forest Products Laboratory
WRINKLING OF THE FACINGS OF SANDWICH CONSTRUCTION SUBJECTED TO EDGEWISE COMPRESSION, by C. B. Norris, W. S. Ericksen, H. W. March, C. B. Smith, and K. H. Boller. Report No. 1810, April 1961.

83 Forest Products Laboratory
EFFECT OF PRESTRESSING IN TENSION OR COMPRESSION ON THE MECHANICAL PROPERTIES OF TWO GLASS-FABRIC-BASE PLASTIC LAMINATES (Supplement to), by Fred Werren. Report No. 1811-A, June 1951. Information reviewed and reaffirmed 1958.

84 Forest Products Laboratory
EFFECT OF SHEAR STRENGTH OF MAXIMUM LOADS OF SANDWICH COLUMNS, by K. H. Boller and C. B. Norris. Report No. 1815, March 1955. Information reviewed and reaffirmed, 1960.

85 Forest Products Laboratory
STRENGTH OF ORTHOTROPIC MATERIALS SUBJECTED TO COMBINED STRESSES, by C. B. Norris. Report No. 1816, March 1955 (Information reviewed and reaffirmed 1960)

86 Forest Products Laboratory
SHORT-COLUMN COMPRESSIVE STRENGTH OF SANDWICH CONSTRUCTIONS AS AFFECTED BY THE SIZE OF THE CELLS OF HONEYCOMB-CORE MATERIALS, by C. B. Norris and W. J. Kommers. Report No. 1817, March 1956.

87 Forest Products Laboratory
MECHANICAL PROPERTIES OF PLASTIC LAMINATES, by Fred Werren. Report No. 1820. Information reviewed and reaffirmed September 1958.

88 Forest Products Laboratory
MECHANICAL PROPERTIES OF PLASTIC LAMINATES, by Fred Werren. Report No. 1820-A. Original report dated 1953. Information reviewed and reaffirmed 1960.

89 Forest Products Laboratory
MECHANICAL PROPERTIES OF PLASTIC LAMINATES (Supplement to), by Fred Werren, Report No. 1820-B, September 1955. Information reviewed and reaffirmed 1960.

90 Forest Products Laboratory
MECHANICAL PROPERTIES OF PLASTIC LAMINATES (Supplement to), by R. L. Youngs. Report No. 1820-C, November 1956.

91 Forest Products Laboratory
MECHANICAL PROPERTIES OF PLASTIC LAMINATES, by G. H. Stevens. Report No. 1820-D, January 1958.

Ref. No.

92 Forest Products Laboratory
MECHANICAL PROPERTIES OF CROSS-LAMINATED AND COMPOSITE GLASS-FABRIC-BASE PLASTIC LAMINATES, Report No. 1821, February 1951. Information reviewed and reaffirmed 1958.

93 Forest Products Laboratory
MECHANICAL PROPERTIES OF CROSS-LAMINATED AND COMPOSITE GLASS-FABRIC-BASE PLASTIC LAMINATES (Supplement to), by A. D. Freas and Fred Werren. Report No. 1821-A, March 1953. Information reviewed and reaffirmed 1959.

94 Forest Products Laboratory
MECHANICAL PROPERTIES OF A LAMINATE DESIGNED TO BE ISOTROPIC, by Fred Werren and C. B. Norris. Report No. 1841. October 1959

95 Forest Products Laboratory
MECHANICAL PROPERTIES OF ALUMINUM HONEYCOMB CORES, by E. W. Kuenzi. Report No. 1849, September 1955. Information reviewed and reaffirmed, 1962.

96 Forest Products Laboratory
TENSILE PROPERTIES OF GLASS-FABRIC LAMINATES WITH LAMINATIONS ORIENTED IN ANY WAY, by E. C. O. Erickson and C. B. Norris. Report No. 1853, November 1955. Information reviewed and reaffirmed 1960.

97 Forest Products Laboratory
MECHANICAL PROPERTIES OF ALUMINUM MULTIWAVE CORES, by E . W. Kuenzi and V. C. Setterholm. Report No. 1855, September 1956.

98 Forest Products Laboratory
ELASTIC BUCKLING OF A SIMPLY SUPPORTED RECTANGULAR SANDWICH PANEL SUBJECTED TO COMBINED EDGEWISE BENDING, COMPRESSION, AND SHEAR, by W. R. Kimel. Report No. 1859, November 1956.

99 Forest Products Laboratory
POISSON'S RATIOS FOR GLASS-FABRIC-BASE PLASTIC LAMINATES, by R. L. Youngs. Report No. 1860, January 1957

100 Forest Products Laboratory
MECHANICAL PROPERTIES OF GLASS-FABRIC HONEYCOMB CORES, by E. W. Kuenzi. Report No. 1861, March 1957.

101 Forest Products Laboratory
TENSILE STRESS-RUPTURE AND CREEP CHARACTERISTICS OF TWO GLASS-BASE PLASTIC LAMINATES, by K. H. Boller. Report No. 1863, June 1959.

102 Forest Products Laboratory
COMPRESSIVE BUCKLING CURVES FOR SIMPLY SUPPORTED SANDWICH PANELS WITH GLASS-FABRIC-LAMINATE FACINGS AND HONEYCOMB CORES, by C. B. Norris. Report No. 1867, December 1958.

Ref. No.

103 Forest Products Laboratory
TORSION OF SANDWICH PANELS OF TRAPESOIDAL, TRIANGULAR, AND RECTANGULAR CROSS SECTIONS., by Shun Cheng. Report No. 1874, June 1960.

104 Forest Products Laboratory
TORSION OF SANDWICH PANELS OF TRAPEZOIDAL, TRIANGULAR, AND RECTANGULAR CROSS SECTIONS (Supplement to). Derivation of Differential Equation and its Application to Rectangular Panels with Loads Applied at Corners, by Shun Cheng. Report No. 1874-A, November 1960.

105 Forest Products Laboratory
EFFECT OF CORE THICKNESS ON SHEAR PROPERTIES OF ALUMINUM HONEYCOMB CORE, by P. M. Jenkinson and E. W. Kuenzi. Report No. 1886, July 1962.

106 Forest Products Laboratory
MECHANICAL PROPERTIES OF SEVERAL HONEYCOMB CORES, by G. H. Stevens and E. W. Kuenzi. Report No. 1887, July 1962.

107 Forest Products Laboratory
STRUCTURAL SANDWICH DESIGN CRITERIA, by E. W. Kuenzi. Report No. 2161, October 1959.

108 Forest Products Laboratory
EFFECTS OF WEATHERING ON THE MECHANICAL PROPERTIES OF FOUR REINFORCED PLASTIC LAMINATES, by K. E. Kimball. WADC-TR-55-319, Supp 4, October 1962. (Prepared for Directorate of Materials and Processes, WPAFB)

109 Forest Products Laboratory
STRENGTH PROPERTIES OF REINFORCED PLASTIC LAMINATES AT ELEVATED TEMPERATURES (CTL 37-9X Resin and 181-A1100 Glass Fabric), by K. H. Boller. ASD Technical Report 61-482, October 1961. (For Air Force Materials Laboratory)

110 Forest Products Laboratory
RESUME OF FATIGUE CHARACTERISTICS OF REINFORCED PLASTIC LAMINATES SUBJECTED TO AXIAL LOADING, by K. H. Boller. ASD-TDR-63-768, December 1963. For Air Force Materials Laboratory, WPAFB.

111 Forest Products Laboratory
STRENGTH PROPERTIES OF REINFORCED PLASTIC LAMINATES AT ELEVATED TEMPERATURES (Narmco 534 Resin and 181-A1100 Glass Fabric), by K. H. Boller. RTD-TDR-63-4091, January 1964. (For Air Force Materials Laboratory)

112 Forest Products Laboratory
EFFECT OF TENSILE MEAN STRESSES ON FATIGUE PROPERTIES OF PLASTIC LAMINATES REINFORCED WITH UNWOVEN GLASS FIBERS, by K. H. Boller. For Air Force Materials Laboratory. ML-TDR-64-86, June 1964.

Ref. No.

113 Forest Products Laboratory
MECHANICS OF ADHESIVE BONDED LAP-TYPE JOINTS: SURVEY AND REVIEW. ML TDR-64-298, October 1964. Contract 33(657)63-358 (Air Force Materials Laboratory)

114 General Dynamics Corporation/Convair
ADVANCED NONMETALLIC PRIMARY STRUCTURE, by J. W. Lyman. Document No. GD/C-64-267, September 1964. (Proprietary)

115 General Dynamics Corporation/Convair
GLASS FIBER REINFORCED PLASTIC BOX BEAM: Technical Proposal. by J. W. Lyman. Document No. GD/C-64-204, July 1964. Prepared for: Naval Air Engineering Center. (Proprietary)

116 General Dynamics Corporation/Convair
SONIC FATIGUE EVALUATION OF FIBERGLAS- POLYURETHANE FOAM SANDWICH IN THE PROPELLER PLANE OF THE MODEL 48A "COIN" AIRCRAFT, by G. L. Getline, 22 May 1964. Document No. DF-48A-102. (Proprietary)

117 General Dynamics Corporation/Convair
FIBERGLASS STRUCTURE, CONVAIR MODEL 48A, LIGHT ARMED RECONNAISSANCE AIRPLANE, Document No. GD/C-64-029-10a, March 1964. Prepared for BuWeps. (Proprietary)

118 General Dynamics Corporation/Convair
ADVANCED NON-METALLIC FABRICATION, by W. E. Hoyt and E. D. Green. Final R & D Report, Task No. 8501-404, December 1963. (Proprietary)

119 General Dynamics Corporation/Convair
PROPOSAL TO DEVELOP PROTOTYPE STRUCTURE OF THE GENERAL DYNAMICS/CONVAIR MODEL 48 FROM SANDWICH-CORE FILAMENT WOUND FIBERGLASS, Document No. 63A0-03, 31 October 1963. In conjunction with R. P. R. Manufacturing Company, Santa Monica. (Proprietary)

120 General Dynamics Corporation/Convair
STRENGTH WEIGHT STUDY, MODEL 48 FIBERGLASS AIRFRAME STRUCTURE, by J. W. Lyman. Document No. PD-48-18, 1 October 1963; and Addendum 1, July 1964 (Proprietary)

121 General Dynamics Corporation/Convair
RESEARCH STUDY OF AIRFRAME COST FACTORS, THIRD REPORT, by E.I. Siden, R.L. Benson, M.R. Yale, J.F. Balata. ERR-SD-028, 30 December 1960 AD 253 993

122 General Dynamics Corporation/Fort Worth
INTERLAMINAR SHEAR STRENGTH TESTS ON FIBROUS REINFORCED COMPOSITES BY PORTA-SHEAR, by L. L. Patton. Report No. MR-D-102, 14 August 1964

Ref. No.

123 General Dynamics Corporation/Fort Worth
ADHESIVE TORSIONAL SHEAR TEST, by L. R. Lunsford, Report ERR-FW-134, 15 January 1964. Contract AF 33(657)-11214 (Air Force Materials Laboratory)

124 General Dynamics Corporation/Fort Worth
STRENGTH OF BONDED STRUCTURAL COMPONENTS EXPOSED TO 325°F THROUGH 10,000 HOURS, by C. H. Mooney, Jr. Report SR/D 60-17, 15 January 1964. Contract AF 33(657)-11214 (Air Force Materials Laboratory)

125 General Dynamics Corporation/Fort Worth
EFFECTS OF ELEVATED TEMPERATURE AND REDUCED ATMOSPHERIC PRESSURE ON ADHESIVES, POTTING COMPOUNDS AND SEALANTS, by J. P. Thomas and R. J. Stout. Report ERR-FW-129, 14 September 1962. Contract AF 33(657)-7248

126 General Dynamics Corporation/Fort Worth
MATERIAL - SCREENING SOLID LAMINATE - RADOMES - EVALUATION OF, by J. R. Peoples, Jr. FGT-2434, Sec 1, 10 September 1962. Contract AF 33(600-36200

127 General Dynamics Corporation/Fort Worth
MATERIAL - ADHESIVES - FMS-0015 (SHELL 422J OR HEXCEL HP-422) AND FMS-0016 (PLASTILOCK 620-626) - JP-4 FUEL IN SANDWICH PANELS - EFFECTS OF FGT-2855, 27 Feb 1962. Contract AF 33(657)-7248

128 General Dynamics Corporation/Forth Worth
EMPIRICAL DATA - HONEYCOMB SANDWICH ELEVATED TEMPERATURE PROPERTIES, DETERMINATION OF, by A. L. Narlo. FTDM 2497, 31 January 1962 Contract AF 33(657)-7248

129 General Dynamics Corporation/Fort Worth
MATERIALS - GLASS CLOTH REINFORCED PLASTICS - ROOM AND ELEVATED TEMPERATURE - PROPERTIES OF, by S. V. Glorioso. Final Report FGT-2186, 26 July 1960. Contract AF 33(657)-7248.

130 General Dynamics Corporation/Fort Worth
MATERIAL-GLASS CLOTH REINFORCED PLASTICS - ROOM AND ELEVATED TEMPERATURE - PROPERTIES OF, by S. V. Glorioso. FGT-2186, 1 May 1959. Contract AF 33(600)-32841.

131 General Dynamics Corporation/Fort Worth
STRUCTURAL DESIGN OF ORTHOTROPIC SANDWICH RADOMES, by G. C. Grimes. From WADC TR 58-272, Vol 1, pp 177 to 198.

132 General Electric Company, Space Sciences Laboratory
HOLLOW GLASS FIBER REINFORCED LAMINATES, by B. W. Rosen and others. Final report, 15 June 1963 - 15 August 1964. Contract Now-63-0674-c

Ref. No.

133 B. F. Goodrich Company
DESIGN INFORMATION FROM ANALYTICAL AND EXPERIMENTAL STUDIES ON FILAMENT WOUND STRUCTURES SUBJECTED TO COMBINED LOADINGS, by R. A. Harrington and others. ABL Library No. 038328. February 1964. For Hercules Powder Company, Allegany Ballistics Laboratory. ABL subcontract No. 89

134 Goodyear Aerospace Corporation
PROPOSAL FOR GLASS-FIBER-REINFORCED PLASTIC BOX BEAM PROGRAM, Naval Air Engineering Center. Document No. GAP-2769, 9 July 1964. (Proprietary)

135 Goodyear Aerospace Corporation
PROPOSAL FOR STUDY OF FIBERGLASS MATERIALS FOR AIRCRAFT APPLICATIONS, Volume I - Technical. Report No. GAP-2856, 3 September 1964. (Proprietary)

136 Goodyear Aerospace Corporation
LIGHT ARMED RECONNAISSANCE AIRPLANE (GA - 39) GER-11463, 5 March 1964 (Proprietary)

137 Goodyear Aircraft Corporation, Akron
CONCERNING THE OPTIMIZATION OF FIBERGLAS REINFORCED PLASTICS FOR STABILITY APPLICATIONS, by E. Rottmayer. GER 11097, 18 April 1963. (Proprietary)

138 Gorcey, R., J. Glyman and E. Green
ADVANCED FABRICATION TECHNIQUES. American Society of Mechanical Engineers, Paper Number 61-AV-13.

139 Grumman Aircraft Engineering Corporation
STRESS ANALYSIS OF FIBERGLASS VERTICAL TAIL SURFACES, by J. Floyd and others. Report No. 3804.221, 1 August 1964

140 Grumman Aircraft Engineering Corporation
AN ENGINEERING EVALUATION OF AIRPLANE GUST LOAD ANALYSIS METHODS, VOLUME III, RESULTS, by J. B. Smedfjeld, Max Chernoff and Dino George. Report No. ADR 06-14-63-3, February 1964. Confidential Report. AD 352 532 Contract Now 60-0449-c

141 Grumman Aircraft Engineering Corporation
AN ENGINEERING EVALUATION OF AIRPLANE GUST LOAD ANALYSIS METHODS, VOLUME II - CALCULATION PROCEDURE, by Dino George, Virginia Kellar and J. B. Smedfjeld. Report No. ADR 06-14-63-2, November 1963. Contract NOw 60-0449-c Confidential Report AD 352 531

142 Grumman Aircraft Engineering Corporation
STRESS ANALYSIS OF ROTODOME, by W. Hauck. Report No. 3808.24, 20 February 1961.Contract NOa(s)57-628c Confidential Report

Ref. No.

143 Hackman, L. E. (North American Aviation), R. J. Molella (Naval Air Engineering Center), C. L. Stotler and D. G. Worthinton (North American Aviation) STRUCTURAL FIBER GLASS AIRCRAFT COMPONENT-PROGRAM RESULTS. AIAA Paper No. 64-142, July 1964.

144 Hackman, L. E. and J. E. Richardson (North American Aviation) "Design optimization of aircraft structures with thermal gradients." In: JOURNAL OF AIRCRAFT, January-February 1964, Vol 1, No. 1.

145 Hackman, L. E. (North American Aviation), R. J. Molella (Naval Air Engineering Center), C. L. Stotler and D. G. Worthington (North American Aviation) DEVELOPMENT OF A STRUCTURAL FIBER GLASS AIRCRAFT COMPONENT. AIAA Paper No. 64-162, (AIAA General Aviation Aircraft Design and Operations Meeting, Wichita, 25-27 May 1964)

146 Haferkamp, H. (Technical University, Hanover) "Non-destructive testing of glass fibre reinforced plastics." In: PROCEEDINGS, REINFORCED PLASTICS CONFERENCE, the British Plastics Federation, Reinforced Plastics Group, London, 25-27 November 1964.

147 Hayes International Corporation, Birmingham REINFORCED PLASTIC LANDING GEAR FOR UH-1 HELICOPTER, by R. D. Holmes and E. L. Moak. Preprint, TRECOM Report, 15 July 1964. Contract DA 44(177)-AMC-120(T)

148 Hayes International Corporation FEASIBILITY OF REINFORCED PLASTICS FOR PRIMARY STRUCTURE OF ARMY AIRCRAFT, by L. R. Anderson and others. TRECOM Technical Report 63-15, March 1963. Contract DA 44-177-TC-756 (Army Transportation Research Command).

149 Hayes International Corporation MECHANICAL PROPERTIES OF PRE-IMPREGNATED POLYESTER LAMINATES, by L. R. Anderson and C. B. Reymann. Engineering Report No. 431, 8 October 1959. (Proprietary)

150 Henderson, J. P. (Air Force Materials Laboratory) NEW TECHNIQUES AND MATERIALS FOR DAMPING. The American Society of Mechanical Engineers, ASMR Publication No. 64-MD-22. 18 March 1964.

151 Hercules Powder Company, Inc., Bacchus Works GENERALIZED PLANE STRAIN OF A PRESSURIZED SPIRALLOY TUBE, by B. W. Shaffer (New York University), 1962 AD 282 756

152 Hercules Powder Company, Research Center INFLUENCE OF ENVIRONMENT ON ROVING AND LAMINATE PROPERTIES, by J. T. Paul, Jr., and J. B. Thomson. Unnumbered, undated. From work carried out under Contract AF 04/647-243

Ref. No.

153 Hess, Goldsmith and Company, Inc.
HOW TO SELECT FIBER GLASS FABRICS. Unnumbered document; undated.

154 Hess, Goldsmith and Company, Inc
PRICES AND SPECIFICATIONS IN INDUSTRIAL GLASS FABRICS. Trade literature, 1 June 1964. Industrial Products Division.

155 Hess, Goldsmith and Company, Inc.
I - 540, HIGH TEMPERATURE FINISH FOR PHENOLIC RESIN-FIBERGLASS LAMINATES. Bulletin F-3, September 1963.

156 Hess, Goldsmith and Company, Inc.
ADHESION OF EPOXY RESINS TO GLASS FABRICS THROUGH FINISHES, by E. L. Lotz and S. J. Milletari. Reprint of Paper #215A, 19th Annual Technical Conference, Society of Plastics Engineers, Los Angeles, February 22 - March 1, 1963.

157 Hexcel Products Incorporated
HONEYCOMB SANDWICH DESIGN, Brochure "E", 1964.

158 Hexcel Products Incorporated
MECHANICAL PROPERTIES OF HEXCEL HONEYCOMB MATERIALS. Technical Services Bulletin No. 120, 1 January 1964; revised 20 February 1964.

159 Hexcel Products Incorporated
ENGINEERING DATA; Technical Service Bulletins; Brochures

160 Holland, K. M. and R. P. Forsberg (Hexcel Products, Inc.)
'Developments in reinforced plastics honeycomb core materials." In: PROCEEDINGS, 12TH ANNUAL MEETING OF THE REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1957. Section 9-E

161 Honeycomb Company of America, Inc.
HONEYCOMB: HONEYCOMB CORE MATERIALS, HONEYCOMB PANELS, COMPLETE HONEYCOMB ASSEMBLIES. Trade literature. Product Information Bulletins: P-1, A-2, P-2, P-3, P-4, P-5. Undated

162 Hughes Aircraft Company
ULTRASONIC TESTING OF ADHESIVE BONDS USING THE FOKKER BOND TESTER, by D. F. Smith and C. V. Cagle. Reprint of paper presented at the Spring National Meeting of the Society of Non-Destructive Testing, March 1964.

163 IIT Research Institute
EFFECT OF CYCLE PROFILE ON THE BIAXIAL COMPRESSIVE FATIGUE PERFORMANCE OF FILAMENT WOUND LAMINATES, by R. H. Cornish, B. W. Abbott and C. K. Cole. First Quarterly Report, M6081. August 1964. Contract No. NObs 90329 (Bureau of Ships, Navy).

164 Jones, W. P. and F. L. Hampson (Fothergill and Harvey. Ltd.)
"A new lightweight structure material." In: ENGINEERING MATERIAL DESIGN, June 1962, p. 404 ff.

Ref. No.

165 Jube, G. (Sub-Aviation, Paris)
"Are reinforced plastics structures heading in the right direction?" In: PROCEEDINGS, FOURTH INTERNATIONAL REINFORCED PLASTICS CONFERENCE, British Plastics Federation, Cafe Royal, London. 25 - 27 November 1964.

166 Kaman Aircraft Corporation
AN EXPLORATORY INVESTIGATION INTO THE DESIGN OF AIRCRAFT STRUCTURES UTILIZING FIBROUS COMPOSITE MATERIALS, by Benjamin Posniak. Report No. R-537, 16 September 1964. (Proprietary)

167 Kaman Aircraft Corporation
FIBERGLASS REINFORCED PLASTICS SHAFTS PARAMETER DEVELOPMENT, by B. Posniak. Report No. R-437 A, April 1964 (Proprietary)

168 Kaman Aircraft Corporation
COMPILATION AND ANALYSIS OF TEST DATA ON FIBER-GLASS-REINFORCED PLASTICS, by R. C. Anderson. TRECOM Technical Report 64-9, March 1964. Contract DA 44-177-AMC-90(T).

169 Kardashov, D. A.
BONDING AGENTS AND THE TECHNOLOGY OF ADHESIVE BONDING. Translation, FTD-TT-62-1617, 6 June 1963 Foreign Technology Division, WPAFB

170 Layton, Paul (Owens-Corning Fiberglas Corporation)
"Current (status) use of reinforced plastic in commerical aircraft." In: PROCEEDINGS, 17TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1962. Section 8-E

171 Lightning and Transients Research Institute
LIGHTNING PROTECTION: REVIEW OF PAST LTRI RESEARCHES RELATING TO KC - 135 AND SIMILAR AIRCRAFT. L & T Report 387; ASD Technical Note 61-92, September 1961. Contract AF 33(616)-7828. (Air Force Materials Laboratory) AD 263 066

172 Lockheed Aircraft Corporation, Missiles and Space Company
BUCKLING OF AXIALLY COMPRESSED SANDWICH CYLINDERS, by B. O. Almroth. Document No. 6-62-64-9, July 1964. (Lockheed independent development program).

173 Lockheed Aircraft Corporation, Missiles and Space Division
ANALYSIS OF FILAMENT-WOUND PRESSURE VESSELS, by Oscar Hoffman. LMSD-480823, 31 May 1960; revised 15 April 1961. Contract NOrd 17017

174 Lockheed - California Company, Division of Lockheed Aircraft Corporation
COIN, LIGHT ARMED RECONNAISSANCE AIRPLANE: COLUME 3.1.2.1, LOCKHEED NONMETALLIC AIRPLANE. LR 17621, 9 March 1964 (Proprietary)

Ref. No.

175 Lockheed-California Company, Division of Lockheed Aircraft Corporation AN ENGINEERING EVALUATION OF METHODS FOR THE PREDICTION OF FATIGUE LIFE IN AIRFRAME STRUCTURES, by W. J. Crichlow, A. J. McCulloch, Louis Young, and M. A. Melcon. ASD-TR-61-434, March 1962. Contract AF 33(616)-6574. (Flight Dynamics Laboratory, WPAFB) AD 276 249

176 Lopez, A. R. (Whittaker Corporation, Advanced Structures Division) "Newest helicopter blade facility is the largest." In VERTI-FLITE, October 1964.

177 Lubin, George, and others (Grumman Aircraft), J. E. Fitzgerald (Brunswick) "Design and construction of a 24-foot rotodome." In: PROCEEDINGS, 16TH ANNUAL CONFERENCE, SOCIETY OF THE PLASTICS INDUSTRY, Reinforced Plastics Division. 7-9 February 1961, Chicago. Section 7-C

178 Madden, J. J. (Union Carbide Plastics Company) "Filament wound properties vs. resin structure and process history." In: PROCEEDINGS, 18TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1963. Section 2-C

179 Marshall, D. W. (Pittsburgh Plate Glass Company) "Glass fibers for reinforced plastics - a survey of the technical literature." In: PROCEEDINGS, 16TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1961.

180 Martenson, J. A. (U. S. Polymeric Chemicals, Inc.) "Improved filament winding materials, their properties and methods of evaluation." In: PROCEEDINGS, 20TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1965. Section 16-D

181 The Martin Company DESIGNING WITH REINFORCED PLASTICS, by J. G. Crockett. Report No. N. D. OR-1297.

182 Massachusetts Institute of Technology, Plastics Research Laboratory A STUDY OF RIGID POLYURETHANE FOAM AS A STRUCTURAL MATERIAL FOR RADOMES, by R. E. Chambers. Unnumbered report, 9 November 1961. AD 271 567.

183 Materials in Design Engineering PARTS, FORMS AND PROCESSES, A MATERIALS HANDBOOK; prepared by the editors. Undated.

184 McDonnell Aircraft Corporation INVESTIGATION OF MECHANICAL PROPERTIES OF FOAM-IN PLACE SANDWICH AT ELEVATED TEMPERATURES, by D. W. Johnson. Report No. A485, 10 March 1964. Contract AF 33(657)-11215 (Air Force Materials Laboratory)

185 McDonnell Aircraft Corporation MECHANICAL PROPERTIES OF NON-WOVEN MONOFILAMENT UNIDIRECTIONAL AND BIDIRECTIONAL LAMINATES OF PARALLITE CORPORATION, Report No. A258, 10 December 1963. Contract AF 33(657)-11215 (Air Force Materials Laboratory)

Ref. No.

186 McDonnell Aircraft Corporation
INVESTIGATION OF HEATING AND FREEZING EFFECTS ON HONEYCOMB PANELS, by H. J. Lause, Report No. A077, 10 September 1963. Contract AF 33(657)-11215 (Air Force Materials Laboratory)

187 McDonnell Aircraft Corporation
AXIAL COMPRESSION TEST - UPPER TORQUE BOX SKIN WHEEL BUMP FAIRING, by D. D. Jones. Report No. 9362, 10 January 1963. Contract AF 33(657)-7749. (Air Force Materials Laboratory)

188 McDonnell Aircraft Corporation
INVESTIGATION OF MECHANICAL PROPERTIES OF FOAM-IN-PLACE SANDWICH AT ELEVATED TEMPERATURES, by D. W. Johnson Report 513-323, 21 June 1963. AD 433 317

189 McDonnell Aircraft Corporation
DETERMINE EFFECT OF MIL-L-7808 OIL ON POLYESTER - GLASS LAMINATE AND MIL-S-8802 FUEL TANK SEALANT COMPOUND, by E. M. Kern. Report No. 9104, 10 October 1962. Contract AF 33(657)-7749. (Air Force Materials Laboratory)

190 McDonnell Aircraft Corporation
COMPARISON OF GLASS CLOTH - 181/150 AND 181/225. Report No. 8892, 10 July 1962. Contract AF 33(657)-7749 (Air Force Materials Laboratory)

191 McGraw-Hill Book Company, Inc.
TECHNIQUES FOR AIRBORNE RADOME DESIGN, by Lt. W. F. Creswell and Lt. Richard Quinn. WADC TR 57-67. September 1957. Contract AF 33(616)-3279

192 McGuire, R. L., Jr. (Flight Dynamics Laboratory, WPAFB)
PHILOSOPHY FOR THE DESIGN OF PRIMARY AIRCRAFT WITH FIBROUS REINFORCED COMPOSITES (RC) Memorandum report.

193 Minnesota University
DAMPING AND FATIGUE PROPERTIES OF SANDWICH CONFIGURATIONS IN FLEXURE, by Leon Keer and B. J. Lazan. ASD Technical Report 61-646, November 1961. Contract AF 33(616)-6828. (Air Force Materials Laboratory, WPAFB) AD 272 016

194 Morgan, Phillip
GLASS REINFORCED PLASTICS, Iliffe Books Ltd., London, 1961

195 National Advisory Committee for Aeronautics
LOCAL INSTABILITY OF THE ELEMENTS OF A TRUSS-CORE SANDWICH PLATE, by Melvin S. Anderson (Langley Aeronautical Laboratory). NACA TN 4292, July 1958

196 National Advisory Committee for Aeronautics
HANDBOOK OF STRUCTURAL STABILITY. PART III - BUCKLING OF CURVED PLATES AND SHELLS, by George Gerard and Herbert Becker. (New York University) NACA TN 3783, August 1957.

Ref. No.

197 National Advisory Committee for Aeronautics
PRELIMINARY INVESTIGATION OF THE STRENGTH AND ENDURANCE OF PLASTIC-IMPREGNATED FIBERGLASS COMPRESSOR BLADES, by D. F. Johnson and A. J. Meyer, Jr. NACA RM E54127a, 10 January 1955

198 National Advisory Committee for Aeronautics
EFFECTS OF RESIN COATING METHODS AND OTHER VARIABLES ON PHYSICAL PROPERTIES OF GLASS-FABRIC REINFORCED POLYESTERS, by B. M. Axilrod, J. E. Wier, and J. Mandel (National Bureau of Standard). NACS RM54G26, 23 August 1954.

199 National Advisory Committee for Aeronautics
PROPERTIES OF HONEYCOMB CORES AFFECTED BY FIBER TYPE, FIBER ORIENTATION, RESIN TYPE, AND AMOUNT, by R. J. Seidl, D. J. Fahey, and A. W. Ross. (Forest Products Laboratory). Technical Note NACA TN 2564, November 1951.

200 National Advisory Committee for Aeronautics
EFFECTS OF HUMIDITY DURING FABRICATION ON SOME PHYSICAL PROPERTIES OF GLASS-FABRIC UNSATURATED-POLYESTER LAMINATES, by J. E. Wier, D. C. Pons, and B. M. Axilrod (National Bureau of Standards) NACA RM 51C21, 18 July 1951

201 National Advisory Committee for Aeronautics
EFFECT OF CELL SHAPE ON COMPRESSIVE STRENGTH OF HEXAGONAL HONEYCOMB STRUCTURES, by L. A. Ringelstetter, A. W. Voss, and C. B. Norris (Forest Products Laboratory) NACA TN 2243, December 1950

202 National Advisory Committee for Aeronautics
EFFECTS OF MOLDING CONDITIONS ON SOME PHYSICAL PROPERTIES OF GLASS-FABRIC UNSATURATED-POLYESTER LAMINATES, by J. E. Wier, D. C. Pons, and B. M. Axilrod (National Bureau of Standards), NACA RM50J19, 9 November 1950.

203 National Advisory Committee for Aeronautics
ANALYSIS OF SHEAR STRENGTH OF HONEYCOMB CORES FOR SANDWICH CONSTRUCTION, by Fred Werren and C. B. Norris (Forest Products Laboratory). NACA TN 2208, October 1950.

204 National Research Council of Canada
AN IMPROVED METHOD FOR PRODUCING FOAM-FILLED FIBERGLASS-REINFORCED PLASTIC PROPELLER BLADES, by B. Neal and W. J. Edwards (National Aeronautical Establishment). Aeronautical Report LR-346, June 1962. AD 283 935

205 National Research Laboratories, Canada
ON THE ADHESION OF ICE TO VARIOUS MATERIALS, by J. R. Stallabrass and R. D. Price. LR-350, July 1962 AD 285 054

Ref. No

206 Naval Air Engineering Center
SURVEY OF FLIGHT-LOAD PARAMETERS OF SERVICE AIRCRAFT (U). Fifth Summary Report, by E. M. Spector and J. J. Kelly. Report No. NAEC-ASL-1070, 29 June 1964. Confidential Report AD 352 291

207 Naval Air Engineering Center, Aeronautical Materials Laboratory
PROBLEM ASSIGNMENT NO. 9-09, INVESTIGATION OF REINFORCED PLASTICS INCLUDING LAMINATES AND SANDWICH FOR AIRCRAFT CONSTRUCTION, UNDER BUREAU OF NAVAL WEAPONS WEPTASK 03 017/200 1/R007 04 01; PROGRESS REPORT NO. NAEC-AML-1956, FORWARDING OF. Ref: M-52-EKR:lrs, 10320/5 (3083), 7 June 1964. AD 443 066

208 Naval Air Engineering Center, Aeronautical Materials Laboratory
RESULTS OF QUALIFICATION TESTS OF SIX SAMPLES OF HEXAGRID ALUMINUM HONEYCOMB CORE MATERIAL UNDER MIL-C-7438C, SUBMITTED BY GENERAL GRID CORPORATION, A SUBSIDIARY OF AMERICAN CYANAMID COMPANY, HAVRE DE GRACE, MARYLAND Report No. NAEC AML 1783, 17 October 1963. AD 422 613

209 Naval Air Engineering Center
REPORT OF PROCEEDINGS, GLASS FIBER REINFORCED PLASTICS CONFERENCE, 27, 28 January 1964.

210 Naval Air Engineering Center
GLASS FIBER REINFORCED PLASTICS CONFERENCE, 27, 28 January 1964. Work Statements.

211 Navy Department, Bureau of Ships
REINFORCED PLASTICS, PREVENTIVE MAINTENANCE AND REPAIR MANUAL, NavShips 250-346-4, August 1959

212 Naval Reserach Laboratory
MAXIMUM STRAINS IN THE RESIN OF FIBERGLASS COMPOSITES, by J. A. Kies. NRL Report 5752, 26 March 1962.

213 Naval Research Laboratory
PREDICTION OF FAILURE DUE TO MECHANICAL DAMAGE IN THE OUTER HOOP WINDINGS IN FIBERGLASS PLASTIC PRESSURE VESSELS, by J. A. Kies. NRL 5736, 18 January 1962. AD 271 693

214 New York Naval Shipyard, Materials Laboratory
THE SHEAR PROPERTIES OF REINFORCED PLASTIC SQUARE PANELS THAT BUCKLE, by H. V. Cordiano and P. Abramov. Final report, S-R 007 03 04, 15 November 1960 AD 246 786

215 North Atlantic Treaty Organization (NATO)
PRATIQUE DU SANDWICH NID D'ABEILLES ALLIAGE LEGER, by C. Thomas NATO Report 217, October 1958. AD 254 481

216 North Atlantic Treaty Organization (NATO)
APPLICATION OF STRUCTURAL ADHESIVES IN AIR VEHICLES, by D. L. Grimes, Advisory Group for Aeronautical Research and Development. NATO Report 181, (Presented at Seventh Meeting of the Structures and Materials Panel, Rome, April 1958.

Ref. No.

217 North American Aviation, Inc.
ESTABLISHMENT OF THE APPROACH TO, AND DEVELOPMENT OF, INTERIM DESIGN CRITERIA FOR SONIC FATIGUE, by G. E. Fitch and others. Technical Documentary Report No. ASD-TDR-62-26, June 1962. Contract AF 33(616)-7694
AD 284 597

218 North American Aviation, Inc.
FABRICATION OF GLASS FABRIC, TEMPERATURE RESISTANT POLYESTER LAMINATES FOR STRUCTURAL USE, Process Specification LA0105-007, 15 March 1961. AD 283 149 L

219 North American Aviation, Inc.
EVALUATION OF HIGH TEMPERATURE REINFORCED STRUCTURAL PLASTICS, by J Yurk. NA-60-964, 13 January 1961 AD 285 232 L

220 North American Aviation, Inc.
LAMINATES, GLASS FABRIC-PHENYL SILANE RESIN, FOR STRUCTURAL USE. Material Specification, LB0130-110, 5 October 1960. AD 283 240 L

221 North American Aviation, Inc.
LAMINATE: GLASS-CLOTH-BASE, ELEVATED-TEMPERATURE-RESISTANT, STRUCTURAL. Material Specification, NA2-4184C, 22 May 1956. AD 283 164 L

222 North American Aviation, Inc.
LAMINATE: GLASS-CLOTH-BASE, ELEVATED-TEMPERATURE-RESISTANT, STRUCTURAL. Material Specification, NA2-4184C, 12 May 1954 AD 283 164 L

223 North American Aviation, Inc., Columbus Division
FIBER GLASS PERFORMANCE CHARACTERISTICS, Applied Mechanics Technical Note, AM TN-1-64, undated (Proprietary)

224 North American Aviation, Inc., Columbus Division
GENERALIZED HONEYCOMB THERMAL STRESS STUDY, by L. E. Hackman and C. W. Annis. Document No. NA61H-755, undated. (Proprietary)

225 North American Aviation, Inc., Columbus Division
PROPOSAL FOR DESIGN AND FABRICATION OF NON-METALLIC BOX BEAMS, by L. E. Hackman Document No. NA64H-700, 17 July 1964 (Proprietary)

226 North American Aviation, Inc., Columbus Division
GLASS FIBER REINFORCED PLASTIC (U), Document No. NA 64H-366, 19 June 1964. Confidential Report (Proprietary)

227 North American Aviation, Inc., Los Angles Division
MATERIALS APPLICATION REPORT, B-70 WEAPON SYSTEM, by R. L. Schleicher. Report No. NA-58-432, 1 December 1958. Contract AF 33(600)-36599. (Materials Section, WPAFB) Confidential Report

Ref. No.

228 Northrop Corporation, Radioplane Division (Now Northrop Ventura)
GLASS FIBER REINFORCED PLASTIC FOR XQ-10 DEVELOPMENT PROGRAM, by R. G. Reeder. AMC Technical Report NR 59-7-263. Report Period: 12 October 1953 to 1 March 1959. Radioplane Report No. 2018. Contract AF(600)-25933 (Manufacturing Methods Division, WPAFB)

229 Northrop Corporation - Ventura Division
RESEARCH AND DEVELOPMENT CAPABILITIES, GLASS FIBER REINFORCED PLASTICS (GRP) FOR LOAD-BEARING AIRCRAFT STRUCTURES, by H. E. Renner and R. H. Smith. Report No. NVR 3606, undated (Proprietary)

230 Nourse, J. H. and J. L. Amick
A REVIEW OF THE TECHNICAL LITERATURE FOR THE ELASTIC ANALYSIS OF FILAMENT-WOUND PRESSURE VESSELS. No. 1 Report, July 1964. For Hercules Powder Company, Allegany Ballistics Laboratory. Personal Service Contract No. 4110-09003

231 Office of Naval Research
MATERIAL REPORTED BY OLIN-MATHIESON CHEMICAL CORPORATION. Memorandum, 3 December 1964. Ref: ONR: 425

232 Oklahoma University Research Institute
STRENGTH PROPERTIES AND RELATIONSHIPS ASSOCIATED WITH VARIOUS TYPES OF FIBERGLASS REINFORCED FACING SANDWICH STRUCTURE, by T. B. Warren and others. (Final report in publication, December 1964.) Contract DA 44-177-AMC-893(T) (Army Transportation Research Command)

233 Oklahoma University Research Institute
RESEARCH IN THE FIELD OF FIBERGLASS-REINFORCED SANDWICH STRUCTURE FOR AIRFRAME USE, by G. Nordby, J. V. Noyes and W. C. Crisman. Final Report, TRECOM Technical Report 64-37, July 1964. Contract DA 44-177-AMC-98(T) (Army Transportation Research Command)

234 Oleesky, S. S. and J. G. Mohr
HANDBOOK OF REINFORCED PLASTICS, of The Society of the Plastics Industry, Inc. Reinhold Publishing Corporation, New York, 1964

235 Owens-Corning Fiberglas Corporation
EVALUATION OF LIGHT WEIGHT FABRICS, by D. G. Mettes (Granville). Memorandum for record, 3 September 1964

236 Owens-Corning Fiberglas Corporation
GLASS REINFORCEMENTS FOR FILAMENT WOUND COMPOSITES, by E. M. Lindsay and J. C. Hood. Final Technical Engineering Report TR-64-8-104, December 1963. Contract AF 33(657)-9623 (AF Materials Laboratory)

237 Owens-Corning Fiberglas Corporation
INDUSTRIAL GLASS FABRIC DESIGN ENGINEERING SYMPOSIUM, May 1960

Ref. No.

238 Owens-Corning Fiberglas Corporation
A COMPARATIVE PROPERTIES ANALYSIS. Trade literature, No. NY-AD-64-3A

239 Owens-Corning Fiberglas Corporation
TEXTILE FIBER MATERIALS FOR INDUSTRY. Publication No. 1-GT-1375.

240 Owens-Corning Fiberglas Corporation
FIBERGLAS REINFORCED PLASTICS AS A STRUCTURAL MATERIAL FOR LIGHT MILITARY AIRCRAFT. Document No. OC-6206. undated.

241 Petker, I. and D. H. Eilfort (Aerojet-General, Azusa)
"Resin systems for filament-wound high external load-bearing structures." In: PROCEEDINGS, 19TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1964. Section 14-B

242 Philco Corporation Research Laboratories, Newport Beach
STRUCTURAL BEHAVIOR OF COMPOSITE MATERIALS, by S. W. Tsai. NASA-CR-71, July 1964. Contract NAS 7-215

243 Picatinny Arsenal, Plastics and Packaging Laboratory
SYMPOSIUM ON ADHESIVES FOR STRUCTURAL APPLICATIONS, 27-28 September 1961. Sponsored by U. S. Army Ordnance Corps.

244 Pinckney, R. L. (The Boeing Company)
RECENT APPLICATIONS OF COMPOSITE STRUCTURE IN MODERN COMMERCIAL AIRCRAFT. AIAA Paper No. 64-164. (Presented at AIAA General Aviation Aircraft Design and Operations Meeting, 25-27 May 1964, Wichita, Kansas)

245 Pittsburgh Plate Glass Company, Glass Research Center
BASIC STUDY ON HOLLOW FIBERS, by J. A. Burgman. Contract NOw 64-0080-C (Bureau of Naval Weapons)
Quarterly Reports:
1. 1 February through 30 April 1964.
2. 1 May through 31 July 1964

246 Plastics Technical Evaluation Center
FILAMENT WINDING BIBLIOGRAPHY: EVALUATED AND ANNOTATED, by A. M. Shibley, PLASTEC Report 19, November 1964.

247 Plastics Technical Evaluation Center
A REVIEW OF PLASTICS FOR TOOLING: MATERIALS, TECHNIQUES, TOLL DESIGN, by N. A. Baldanza. PLASTEC Report 15 March 1964. AD 601 391

248 Plastics Technical Evaluation Center
A SURVEY OF FILAMENT WINDING: MATERIALS, DESIGN CRITERIA, MILITARY APPLICATION by A. M. Shibley, H. L. Peritt and Merrill Eig. PLASTEC Report 10, May 1962. AD 284 629

Ref. No.

249 Posniak, Benjamin (Kaman Aircraft Corporation)
"Development of a directed fiber F.R.P. helicopter rotor blade." In: PROCEEDINGS, 17TH ANNUAL MEETING OF THE REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1962. Section 2-D

250 Raech, Harry and J. M. Kreinik (FMC Corporation)
"Prepreg materials for high performance dielectric applications." In: PROCEEDINGS, 20TH ANNUAL TECHNICAL AND MANAGEMENT CONFERENCE, Reinforced Plastics Division, the Society of the Plastics Industry. 1 - 5 February 1965, Chicago. Paper No. 11A

251 Rand Corporation, Santa Monica
A REVIEW AND EVALUATION OF FILAMENT-COMPOSITE MATERIALS (U), by WL. Micks. Memorandum No. RM-4166-PR, October 1964. Confidential Report (Proprietary)

252 Rand Corporation, Santa Monica
AIRCRAFT AIRFRAME COST ESTIMATING TECHNIQUES, by J. M. Carrier and R. W. Smith. RM-3375-PR. November 1962. AD 293 863

253 Raymond Development Industries, Inc.
INTEGRALLY WOVEN FLUTED CORE PANELS, Trade literature, 1961, unnumbered.

254 Rock Island Arsenal, Design Engineering Branch
SANDWICH DESIGN AIDS, by C. M. Fitzpatrick. Report No. 62-2355. July 1962.

255 Rock Island Arsenal, Design Engineering Branch
A STRUCTURAL APPLICATION OF ALUMINUM HONEYCOMB SANDWICH CONSTRUCTION, by C. M. Fitzpatrick. Technical Report No. 61-2509, June 1961. AD 260 559

256 Rosato, D. V. and C. S. Grove, Jr.
FILAMENT WINDING: ITS DEVELOPMENT, MANUFACTURE, APPLICATIONS, AND DESIGN. Interscience Publishers, New York, 1964.

257 Royal Aircraft Establishment
TEMPERATURE LIMITATIONS OF POLYMERIC MATERIALS, PART 7 - TRANSPARENCIES, by G. Wood. Technical Note CPM. 53, April 1964. AD 443 635

258 Royal Aircraft Establishment/Farnborough
THE SHEAR STIFFNESS OF A CORRUGATED WEB, by K. I. McKenzie. Report No. Structures 275, June 1962

259 Royal Aircraft Establishment/Farnborough
STRUCTURAL WEIGHT ESTIMATIONS FOR NOVEL CONFIGURATIONS, by M. E. Burt. Report Structures 270, December 1961. AD 271 622

260 Royal Aircraft Establishment/Farnborough
RAIN EROSION, PART IV. AN ASSESSMENT OF VARIOUS MATERIALS, by R. B. King. Report CHEM. 521, September 1960. AD 249 682 (Proprietary)

Ref. No.

261 Schmidt, D. L. and R. C. Tomashot (Air Force Materials Laboratory, WPAFB)
"Astronautics and rocketry - Part I In: REINFORCED PLASTICS, July-August 1964. Part II - In: REINFORCED PLASTICS, September-October 1964

262 Shell Chemical Company, Plastics and Resins Division
EPON RESINS FOR REINFORCED STRUCTURES. Technical Bulletin SC: 64-48.

263 Siefert, R. F. (Pittsburgh, Plate Glass Co.)
"Hollow-glass filaments." In: DESIGN NEWS, 27 May 1964

264 Sikorsky Aircraft, Division of United Aircraft Corporation
THE APPLICATION OF RELIABILITY CONCEPTS TO FATIGUE LOADED HELICOPTER STRUCTURES, by H. T. Jensen. Paper delivered to American Helicopter Society, 18th Annual Forum, Washington, 3 May 1962. AD284 471

265 Snyder, F. S. and R. E. Drake (Piper Aircraft Corporation)
EXPERIENCE WITH REINFORCED PLASTIC PRIMARY AIRCRAFT STRUCTURES. Society of Automotive Engineers publication, No. 622C

266 The Society of the Plastics Industry, Asbestos Technical and Standards Committee
ASBESTOS-REINFORCED PLASTICS, a Bulletin, submitted by D.V. Rosato (Telecomputing Corporation), Chairman.

267 Solar, Subsidiary of International Harvester Company
EXPLORATION AND EVALUATION OF NEW GLASSES IN FIBER FORM, by G. K. Schmitz. Final Report, RDR 1266-7, 12 April 1963. Contract NOnr 3694 (00) (X) (Naval Research Laboratory)

268 Sonneborn, R. H., A. G. Diets and A. D. Heyser
FIBERGLAS REINFORCED PLASTICS, Reinhold Publishing Corporation, New York, 1954.

269 Southern Research Institute
THE DEVELOPMENT AND EVALUATION OF HEAT-RESISTANT, RAIN-EROSION RESISTANT COATINGS, by R. B. Perkins. Bimonthly Progress Reports: 4. -16 April 1964. Contract N156 - 43450

270 Southern Research Institute
THE THERMOPHYSICAL PROPERTIES OF PLASTIC MATERIALS FROM -50°F to Over 700° F, by C. D. Pears, W. T. Engelke and J. D. Thornburgh. ML-TDR-64-87, Part I, August 1964. Contract AF 33(657)-8594

271 Stanford University, Department of Aeronautics and Astronautics
IMPERFECTIONS, A MAIN CONTRIBUTOR TO SCATTER IN EXPERIMENTAL VALUES OF BUCKLING LOAD, by W. H. Horton and S. C. Durham. TRECOM Technical Report 64-31, May 1964. Contract DA 44-177-AMC-115(T) (Army Transportation Research Command)

Ref. No.

272 Sterman, S. and J. G. Marsden (Union Carbide Corporation, Tonowanda) "Silane coupling agents as 'integral blends' in resin-filler systems." In: PROCEEDINGS, 18TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1963. Section 1-D

273 J. P. Stevens and Company, Inc., Industrial Glass Fabrics Department STEVENS STYLE 181/VOLAN, AND 181/S-550; LAMINATE PROPERTIES: FIBER GLASS FABRIC. Product Data Sheet, dated 25 March 1964.

274 J. P. Stevens and Company, Inc., Industrial Glass Fabric Department STEVENS STYLE 7581/VOLAN, AND 7581/S-550, LAMINATE PROPERTIES, FIBER GLASS FABRIC. Product Data Sheet, dated 25 March 1964.

275 J. P. Stevens and Company, Inc., Industrial Glass Fabrics Department INDUSTRIAL GLASS FABRICS, Product Data Sheet, 301-1, 9 March 1964.

276 J. P. Stevens and Company, Inc., Industrial Glass Fabrics Department HIGH MODULUS FIBER GLASS FABRICS. Trade literature, dated March 1963.

277 J. P. Stevens and Company, Inc., Industrial Glass Fabrics Department FIBER GLASS AND THE WEAVER, by I. J. Gusman. (Reprint from THE GLASS INDUSTRY)

278 J. P. Stevens and Company, Inc., Industrial Glass Fabrics Department STEVENS FIBER GLASS FABRICS FOR INDUSTRY. Brochure, unnumbered, dated 15 December 1960.

279 J. P. Stevens and Company, Inc., Industrial Glass Fabric Department FINISH S-910, Product Data Sheet (Preliminary), undated.

280 J. P. Stevens and Company, Inc., Industrial Glass Fabrics Department FINSIH S-920, Product Data Sheet (Preliminary), undated.

281 Street, S. W. (Reichhold Chemicals, Inc.) "A new fast curing hardener for epoxy resins." In: PROCEEDINGS, 18TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1963. Section 1-F

282 Stulen, F. L. (Parsons Corporation) "Reinforced plastic helicopter blades." In: PROCEEDINGS, 16TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1961. Section 10-A

283 Stuttgart Technical University PROPELLERS AND HELICOPTER BLADES OF STRAND-REINFORCED SYNTHETIC RESIN MATERIAL, by Ulrich Hutter. From: Special issue from the Year Book 1960 of the WGL. DK 621.074.4; 533.695.8; 769.57; 677.521. Translated by Alis Himsl.

Ref. No.

284 Syracuse University Research Institute
MATERIALS DESIGN HANDBOOK, DIVISION 1, STRUCTURAL PLASTICS, by C. S. Grove, Jr. and R. Ford Pray, III. ML-TDR-64-141, May 1964. Contract AF 33(616)-7736 (Air Force Materials Laboratory)

285 Thiokol Chemical Corporation, Wasatch Division
JOINT DESIGN, FIBERGLASS REINFORCED PLASTIC SEGMENTED MOTOR CASES, by John Hinchman. Reprint of paper presented at the 20th Interagency Solid Propellant Meeting, Philadelphia, 13-15 July 1964

286 Turner, A. W. (Lockheed Flight Test Division)
"Lightning and aircraft - Part one - a basis for discussion."
In: THE AIR LINE PILOT, October 1964, p. 7 ff.

287 Union Carbide Plastics Company
RESEARCH ON IMPROVED EPOXY RESINS. Contract NOnr4172 (00) (X) (Naval Research Laboratories) Quarterly Progress Reports:
1 - 1 August 1963, by J. J. Madden, D. P. Norris, R. F. Sellers, S. G. Smith
2 - 1 November 1963, by J. J. Madden, D. P. Norris, R. F. Sellers, S. G. Smith
3 - 1 February 1964, by D. P. Norris, R. F. Sellers, S. G. Smith
4 - 1 May 1964, by W. P. Mulvaney, R. F. Sellers, S. G. Smith
5 - 1 August 1964, by J. R. Harvey, W. P. Mulvaney, R. F. Sellers, S. G. Smith, Jr.

288 United Kingdom, Ministry of Aviation
BIBLIOGRAPHY ON GLASS FIBER AND PLASTIC LAMINATES, TIL/BIB/51, March 1961 AD 257 587

289 Whitaker, R. A. (Thompson Ramo Wooldridge, Inc.)
REINFORCED PLASTIC FABRICATION PROCESSES. Society of Automotive Engineers, Inc., publication, No. 684C.

290 Whittaker Corporation, Narmco Research and Development Division
OPTIMUM DESIGN FOR BOX BEAMS OF REINFORCED PLASTICS CONSTRUCTION, Narmco Proposal No. 100-2290, 23 October 1964. Prepared for: Naval Air Engineering Center. (Proprietary)

291 Whittaker Corporation, Narmco Research and Development Division
HIGH-TEMPERATURE ADHESIVES AND RESINS, by P. M. Hergenrother, J. L. Kerkmeyer and H. H. Levine. Quarterly Progress Reports: Contract NOw 64-0524C.
1 - (Not available for review)
2 - 15 July 1964 through 15 October 1964

292 Whittaker Corporation, Narmco Research and Development Division
MECHANICAL RELATIONSHIP OF REINFORCEMENTS AND THE BINDER MATRIX, by Juan Haener. Technical Progress Report No. 1, September 1964. Contract DA 44-177-AMC-208(T) (Army Transportation Research Command)

Ref. No.

293 Whittaker Corporation, Narmco Research and Development Division. APPENDIX, NASA BOX BEAM ANALYSIS. NRD document 100-2159, March 1964. (Hand-written copy reviewed.) (Proprietary)

294 Whittaker Corporation, Narmco Research and Development Division DETERMINATION OF THE PERFORMANCE OF PLASTIC LAMINATES UNDER CRYOGENIC TEMPERATURES, by N. O. Brink. Report No. ASD-TDR-62-794, February 1963. Contract AF 33(616)-8289 (Air Force Materials Laboratory)

295 Wood, R. P., J. S. White, and T. E. Phillips (Owens-Corning Fiberglas Corporation) "Fibrous glass reinforcements for epoxies." In: PROCEEDINGS, 14TH ANNUAL MEETING, REINFORCED PLASTICS DIVISION, the Society of the Plastics Industry, Chicago, February 1959. Section 13-D

296 Cornell Aeronautical Laboratory, Incorporated PROCESS AND FABRICATION STUDIES OF STRUCTURAL, GLASS REINFORCED PLASTIC WING, by N. E. Wahl and S. Willimas. Interim progress report, 17 April 50 - 17 June 50. Report No. BE-688-M-1. Contract NOa(s) 11016.

297 Cornell Aeronautical Laboratory, Incorporated MATERIAL TESTS, DESIGN AND STRESS ANALYSIS OF A GLASS REINFORCED PLASTIC WING, by S. P. Williams. Interim progress report, 17 June - 1 Dec 1951. Report No. BE-688-M-2, 3 March 1952. Contract NOa(s) 11016.

298 Cornell Aeronautical Laboratory, Incorporated THE DESIGN, FABRICATION AND TESTING OF A STRUCTURAL GLASS REINFORCED PLASTIC WING, by W.W. Bird, N.E. Wahl and S.P. Williams. Final engineering report. Report No. BE-688-M-3, December 1955. Contract NOa(s)-11016.

299 Lockheed Aircraft Corporation, Missile and Space Division RAIN EROSION AT SUBSONIC AND SUPERSONIC SPEEDS: AN ANNOTATED BIBLIOGRAPHY, by A.A. Beltran. Special Bibliography No. SB-62-6, March 1962. Contract NOrd 17017. AD 276-495

300 Materials in Design Engineering (Staff) "Epoxy Plastics." Materials and Process Manual, No. 227, January 1965. MATERIALS IN DESIGN ENGINEERING, Vol 61, No. 1, January 1965.

301 Wright-Patterson Air Force Base, Directorate of Flight and All-Weather Testing FUNCTIONAL SERVICE TEST OF FIBERGLASS WING, by W.E. Brown. WADC Technical Report No. 55-442, 6 Sept 1962.

AVAILABILITY OF OTHER PLASTEC REPORTS

Number	Identification	OTS Price
PLASTEC Report 1 AD 244 104	STATE OF THE ART - FLAKE-GLASS LAMINATES, by Allen M. Shibley, October 1960	$2.75
PLASTEC Report 2 AD 245 498	OXYGEN-ALCOHOL TEST FOR INSULATING MATERIALS - A MEMORANDUM ON A SCREENING TEST FOR ROCKET-MOTOR BLAST TUBES, by Harold F. Mannheimer and Allen M. Shibley, October 1960	$1.00
PLASTEC Report 3 AD 247 865	RECENT DEVELOPMENTS IN CASTING RESINS AND TECHNOLOGY FOR ELECTRICAL ENCAPSULATION APPLICATIONS, by Arnold E. Molzon, November 1960	$1.00
PLASTEC Report 4 AD 273 400	DESIGN CRITERIA FOR PLASTIC PACKAGE-CUSHIONING MATERIALS, by Mario E. Gigliotti, December 1961	$2.75
PLASTEC Report 5A AD 419 399	DIRECTORY IN PLASTICS - KNOWLEDGEABLE GOVERNMENT PERSONNEL, by Norman E. Beach, April 1963	$3.50
PLASTEC Report 6 AD 261 549	STATE OF THE ART - METHODS OF BONDING FLUOROCARBON PLASTICS TO STRUCTURAL MATERIALS, by Marjorie C. St. Cyr, May 1961	$0.75
PLASTEC Report 7 AD 276 142	GUIDE TO SPECIFICATIONS FOR RIGID LAMINATED PLASTICS, by Ruth S. Kobler and Cecilia U. McNally, March 1962	$5.00
PLASTEC Report 8 AD 264 775	SUBJECT INDEX, BIBLIOGRAPHY, AND CODE DESCRIPTION OF TECHNICAL CONFERENCE PAPERS ON PLASTICS: MARCH 1960 - FEBRUARY 1961, by Arnold E. Molzon, July 1961	$1.75
PLASTEC Report 9 AD 294 117	REDUCTION OF REFLECTIVITY FROM TRANSPARENT MATERIALS: A MEMORANDUM IN EVALUATION OF TECHNIQUES APPLICABLE TO PLASTIC HELICOPTER CANOPIES, by Norman E. Beach, July 1962	---
PLASTEC Report 10 AD 284 629	A SURVEY OF FILAMENT WINDING: MATERIALS, DESIGN CRITERIA, MILITARY APPLICATIONS, by Allen M. Shibley, Harvey L. Peritt and Merrill Eig, May 1962	---
PLASTEC Report 11 AD 282 795	SUBJECT INDEX, BIBLIOGRAPHY, AND CODE DESCRIPTION OF TECHNICAL CONFERENCE PAPERS ON PLASTICS: 15 FEBRUARY 1961 - 23 FEBRUARY 1962, by Arnold E. Molzon, June 1962	$1.50
PLASTEC Report 12 AD 288 682	EFFECTS OF THE SPACE ENVIRONMENT ON PLASTICS: A SUMMARY WITH ANNOTATED BIBLIOGRAPHY, by Arthur H. Landrock, July 1962	$2.25
PLASTEC Report 13 AD 431 603	FLUIDIZED-BED COATING WITH PLASTICS: TECHNOLOGY AND POTENTIAL FOR MILITARY APPLICATIONS, by Arthur H. Landrock, January 1964	$3.50
PLASTEC Report 14 AD 423 560	SUBJECT INDEX, BIBLIOGRAPHY, AND CODE DESCRIPTION OF TECHNICAL CONFERENCE PAPERS ON PLASTICS: 10 FEBRUARY 1962 - 10 FEBRUARY 1963, by Arnold E. Molzon, August 1963	$2.50
PLASTEC Report 15 AD 601 391	A REVIEW OF PLASTICS FOR TOOLING: MATERIALS, TECHNIQUES, TOOL DESIGN, by Nicholas T. Baldanza, March 1964	$3.00
PLASTEC Report 16 AD 605 396	PLASTIC GEARS: A MEMORANDUM ON FEASIBILITY FOR USE IN AMMUNITION ITEMS, by John Nardone, July 1964	$1.00
PLASTEC Report 17 AD 606 561	SUBJECT INDEX, BIBLIOGRAPHY, AND CODE DESCRIPTION OF TECHNICAL CONFERENCE PAPERS ON PLASTICS: 28 Feb. 1963 - 1 Mar. 1964 by Arnold E. Molzon, July 1964	$2.50
PLASTEC Report 18 AD 609 526	GLASS/RESIN INTERFACE: PATENT SURVEY, PATENT LIST, AND GENERAL BIBLIOGRAPHY, by William J. Eakins, September 1964	$1.25
PLASTEC Report 19 AD 457 593	FILAMENT WINDING, BIBLIOGRAPHY: EVALUATED AND ANNUTATED, by Allen M. Shibley, December 1964	---
PLASTEC Note 1 AD 261 550	THE APPLICATION OF NONDESTRUCTIVE TESTING TO PLASTICS, by Alfred M. Anzalone, July 1961	$1.25
PLASTEC Note 2 AD 268 266	INDEXED REFERENCES PERTAINING TO DEGRADATION AND FRACTURE OF PLASTICS, by Arnold E. Molzon, August 1961	$1.00
PLASTEC Note 3 AD 402 225	DEFENSE SPECIFICATIONS AND STANDARDS FOR AND RELATING TO REINFORCED PLASTICS, by Norman E. Beach, March 1963	$0.50
PLASTEC Note 4 AD 275 832	PLASTICS IN THE MEDICAL INDUSTRY: A CROSS-INDEXED BIBLIOGRAPHY, by Arnold E. Molzon, March 1962	$1.25
PLASTEC Note 5 AD 276 001	HEALTH HAZARDS AND TOXICITY OF PLASTICS: A CROSS-INDEXED BIBLIOGRAPHY by Arnold E. Molzon, March 1962	$0.75
PLASTEC Note 6 AD 410 401	GOVERNMENT SPECIFICATIONS AND STANDARDS FOR PLASTICS, COVERING DEFENSE ENGINEERING MATERIALS AND APPLICATIONS, by Norman E. Beach, May 1963	$1.75
PLASTEC Note 7 AD 423 546	LITERATURE SURVEY ON THERMAL DEGRADATION, THERMAL OXIDATION AND THERMAL ANALYSIS OF HIGH POLYMERS, by David W. Levi, June 1963	$2.75
PLASTEC Note 8 AD 425 147	FILAMENT WINDING IN MILITARY APPLICATIONS, A DISCUSSION OF PROBLEMS ASSOCIATED WITH FILAMENT WOUND MOTOR CASES, by Allen M. Shibley, September 1963	$1.00