associated with slaving the translational visual and motion system may be less significant than was first thought.

For formation flying studies it is probable the LAFS will be adequate. The motion cueing is within the capability of the existing motion system and the visual display is adequate.

Studies associated with high-speed interception will probably require the type of simulator referred to as the Differential Maneuvering Simulator. Accepting the fact that sustained load factor is not available, a high-quality reproduction of visual information with unlimited field of view seems necessary. The simulator provides this through the use of sophisticated servo driven optical equipment.

This optical manuevering type of simulator seems necessary for studies associated with close air-to-air combat. High-fidelity visual information on the target must be provided with a 360° field of view. It is doubtful that such visual fidelity could be provided in conjunction with a motion system capable of sustaining load factor.

For low-altitude, high-speed flight studies the LAFS has proved adequate. The motion system is capable of providing the motion cues authentically; and although previous simulations have been conducted with in-cockpit displays, a projected terrain-following display system should make VFR simulation possible.

The LAFS has been used in dive bombing investigations. It appears that some improvement in fidelity can be achieved by employing a visual display that allows better perspective simulation. Otherwise, the simulator provides an excellent mechanism for studying dive bombing. It is probable that the simulator with the proposed display can be used in studies associated with air-to-air combat. It would be necessary in this application to develop some logic to drive the target airplane, but this is well within the state-of-the-art.

It is anticipated that by minor refinements of the drive philosophy discussed in the previous sections and rigorous definition of the dynamic characteristics of the washout devices, these simulators will be extremely useful handling qualities research devices.

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Composite Airframe Design

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Manufacturing methods utilizing unidirectional glass reinforcement were applied to design and fabrication of a demonstration wing section representing the T2B airplane wing from station 115 to tip station 207.5. Design was substantiated by detail tests. Fabrication problems and solutions are discussed. Ultimate strengths of full-scale structures were 6% and 30% over the design goal with 40% less weight, 20% less estimated fabrication cost, greater bending stiffness-to-weight ratio, and greater torsional stiffness-to-weight ratio than an aluminum wing.

Introduction

THIS paper discusses the application of unidirectional, S-glass composite to a wing structure using available filament winding fabrication methods. Unidirectional, S-glass laminate has a substantially higher strength-weight ratio, for both tension and compression loads, than either glass fabric laminate or any aluminum alloy. Therefore, unidirectional S-glass roving is a candidate material for strength critical aircraft structures. The filament winding operation provides precise roving placement by machine and utilizes glass filament in an economical form. In addition, the manufacturing methods developed for unidirectional, S-glass aircraft structures will be applicable in principle to applying high modulus fibers to stiffness-critical aircraft structures.

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Index categories: Aircraft Structural Materials; Structural Composite Materials; Optimal Structural Design.

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Design Considerations

The demonstration article for this program represents the T2B aircraft wing outboard of wing station 115 as shown in

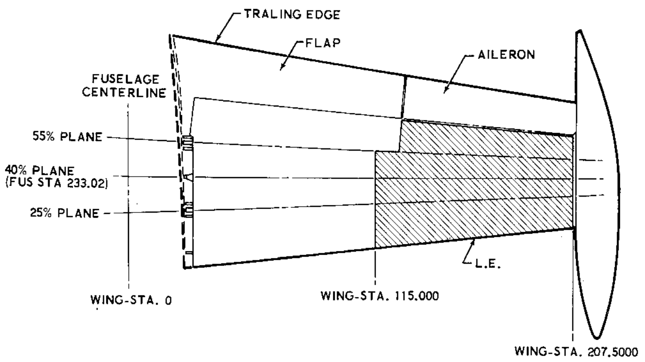


Fig. 1 T2B Wing area represented by demonstration article.

Fig. 1. The demonstration article includes wing hard points and cutouts as indicated in Fig. 2. The structural configuration of four truss spars, rib caps, and spar caps integral with the wing sandwich cover is shown in perspective in Fig. 3. The structural access door is representative of those on the lower wing surface and illustrates the typical door flange, door frame and bolting land.

Available Materials

The selected prepreg roving for filament winding is S-994, 20 end roving with HTS-904 surface finish, or size, impregnated with Shell Chemical Company 58-68R resin with MNA (methyl nadic anhydride) curing agent and BDMA (benzyl dimethylamine) accelerator. The BDMA content was decreased from the normal specification content to obtain maximum resin working life to allow for unanticipated delay in production.

Preimpregnated glass cloth selected for fittings and solid inserts was S901/51 fabric with an Epon 828 resin system for compatible cure with the prepreg roving. The filament wound glass roving layer is nominally 0.008 inch thick. A 0.008 in. prepreg fabric thickness was selected for compatibility in laminates of both fabric and filament winding.

The film adhesives selected for joining separate structural elements were AF-111 and AF-126, proprietary products of the 3M Company. An alternate adhesive, selected to avoid single-source procurement problems, was Epon 9601, a proprietary product of the Shell Chemical Company. The adhesives are compatible to simultaneous cure with the roving and the fabric prepregs.

The aluminum core selected for the cover sandwich was $\frac{1}{8}$ in. hexagonal cell 0.0015-in.-thick foil 5052 aluminum honeycomb (specification MIL-C-7438C) for nearly flat

Table 1 Quantity of parts per concept

	2-dim truss	3-dim truss	Internal post	Honeycomb core	Radial multispar
Spar assembly	5	5	5	5	15
vertical	90	0	60	0	180
diagonal	2	422	0	0	225
cap	10	0	5	5	30
Fittings	13	13	13	13	23
Rib assembly	4	4	4	4	4
caps and honeycomb	16	4	4	4	4
Skins	2	2	2	1	2
parts	5	5	5	4	5
Leading edge parts	4	4	4	3	4
Total parts	151	459	102	39	492

surfaces. The small $\frac{1}{8}$ in. cell size was selected for adequate buckling stabilization of flat, 0.016-in.-thick sandwich facings. For the curved surface at the leading edge, $\frac{3}{8}$ cell 0.0037-in.-thick 5052 aluminum Flexcore, supplied by Hexcel Products, Inc., was selected.

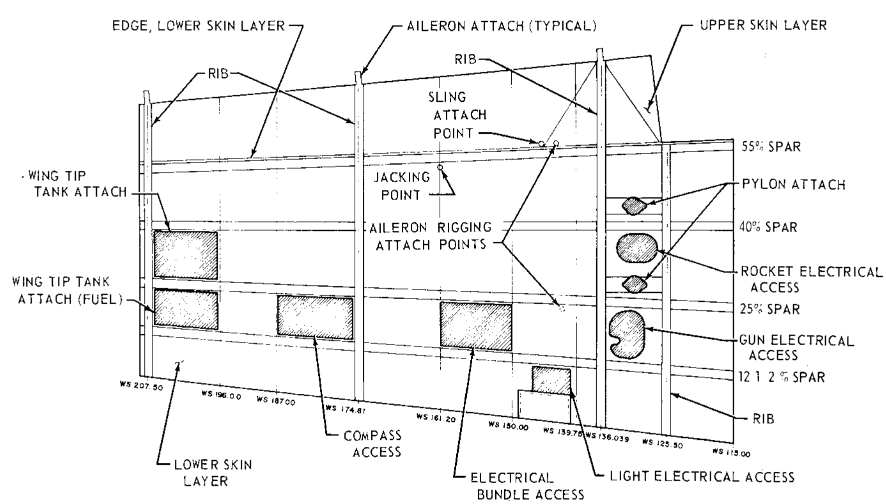
The environmental protective film, used on exterior and interior of the structural cover, is polyvinyl fluoride (PVF) film supplied by DuPont as Tedlar. This material was selected for its low moisture vapor transmission as compared to paint or gel coat and for the excellent resistance to solar radiation, weathering, and to the common fluids around aircraft such as gasoline, hydraulic oil and cleaning compounds.

The mandrel material upon which the filament wound cover was formed was sand-acrylic base resin (Goodrite A73 resin, a product of Goodrich Chemical Company). Mandrel material selection was based on high tensile strength, fast washout characteristics and suitability to sweeping methods of forming the mandrel surface. The sweeping method, used in forming plaster surfaces, can most economically form mandrel surfaces of a single degree of curvature.

Results of Design Optimization

The relative fabrication cost and five configurations are illustrated and compared in Fig. 4. The cost is based on the number of parts in the assembly listed in Table 1. The three dimensional spar and the radial multispar configurations had too many parts. After four articles were fabricated, an estimate of production costs of the four-spar demonstration article was made and compared with production cost of the current aluminum skin-stringer structure. It was concluded that the demonstration article production cost would be equal to or less than that of the aluminum structure. This is because the number of hand placed parts in the aluminum structure (including rivets) is four times that of the four-spar demonstration article. A simpler, two sandwich spar article would have 30% fewer parts than the

Fig. 2 Schematic showing spar and rib locations and wing access cutouts.



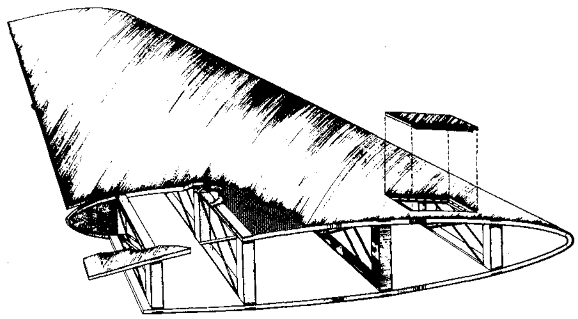


Fig. 3 Filament wound aircraft wing fabrication concept, four spar system.

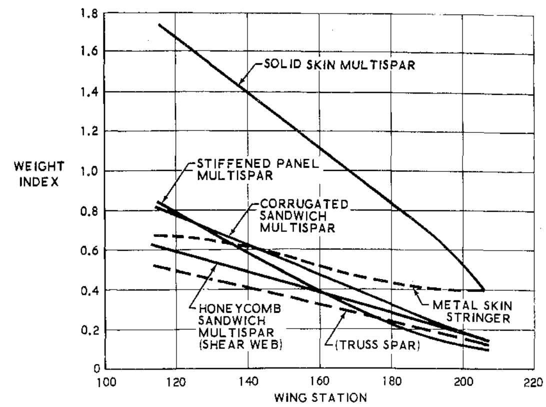


Fig. 5 Weight-index comparisons for various methods of construction.

four-spar article and cost substantially less than the aluminum structure. The percent excess weight (relative to the 2-dimensional truss) were as follows: 3-dimensional truss 18%, honeycomb core 90%, radial multispar 15%. The internal post configuration is a variation of the four truss spar configuration finally selected. A 2 spar or a 3 spar configuration with some weight increase, could also have been used.

The weight index, in cubic inches of material per in. span, of compression cover configurations versus the demonstration article wing stations is shown in Fig. 5. The honeycomb sandwich multispar (truss spar) was used. The metal skin-stringer weight is shown for comparison.

Design for Filament Winding

An outboard portion of a wing, represented by the demonstration article, was designed for filament winding. The demonstration article design has a linear, spanwise taper in plan form and in airfoil thickness, has a sandwich core thickness taper, and has a capability of tapering the cover skin thickness by changing the number of skin winding layers along the span. Representative skin access doors and hard points shown in Fig. 2, authorized for use on the T2B aircraft, are included in the demonstration article. Aileron hinge fittings are combined with the filament wound rib cap.

Internal Structure

The internal structure consists of unidirectional spar caps, (1.5 in. wide and 0.048–0.090 in. thick at intersections), unidirectional rib caps (1.5 in. wide and 0.030 in. thick), and truss spar webs with compression posts and tension diagonals as shown in Fig. 3. Details of the spar rib intersection are shown as Fig. 6. Rib webs were secondarily bonded in place in the assembly after removing the winding mandrel (not shown in the figure). Spar web compression posts are a honeycomb core sandwich with unidirectional, filament wound skins. Spar post sandwich fabrication was accomplished on two long steel plate mandrels. Parallel filament sandwich posts were hoop wound back to back on a steel plate mandrel, then cut to size and fitted with end caps. Sandwich spar web sections were canted hoop wound in pairs, back to back, on a plate mandrel. Inner spar caps of unidirectional roving were wound in place by rotating the spar mandrel with posts and continuous web sections in place. The unidirectional tension diagonals were canted hoop wound over sawtooth winding aids which anchor the end of the diagonals.

Positioning the spars in an assembly mandrel requires assembly mandrel structural elements running spanwise between the spars. It was fortunate that both cover stabilization against buckling and the mechanics of removal of assembly mandrel segments are more suited to the optimum multispar structural configuration than to a multirib structural configuration. Reusable mandrel segments are desired to minimize the labor of preparing the mandrel for fabrication of the next assembly unit.

Design with nonbuckling structural elements and with the relatively low compression modulus of glass filaments requires the use of honeycomb core stabilization in the spar posts and in caps for ribs and spars.

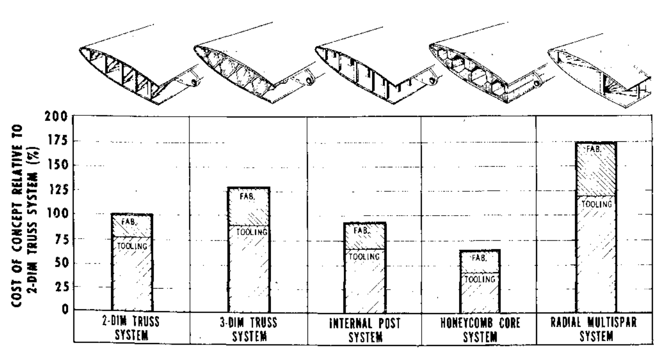


Fig. 4 Comparison of total costs.

External Cover

During this program, membrane shear panels, stiffened membrane shear panels, and honeycomb sandwich panels were tested in edgewise or panel shear. The results indicate that membrane panels cannot buckle without permanent damage to the membrane skin. The need for honeycomb core stabilized sandwich skins, the desirability of collecting longitudinal windings in spar caps, and the need of canted hoop skin windings to allow taper in the span direction of wing skin thickness are the major considerations for the external cover.

Access openings are reinforced by a frame around the cutout. Structural doors employ structural screws in the solid bolting flanges on the door and in the lands on the reinforcing frame around the opening.

Collecting the longitudinal filaments into caps avoids discontinuities at access door cutouts and results in less sandwich core weight to stabilize cover panels between spars.

The inner and outer skins of the cover sandwich are two roving layers oriented at ±45° as shown in Fig. 7. This amount of material was required for design torsional load at wing station 115 but was overstrength (and overweight) outboard of that station. However, two roving layers are considered a minimum thickness to avoid laminate splitting and handling damage. For this reason the demonstration article S-glass panel weight was greater than the optimum for the T2B load as shown on Fig. 9.

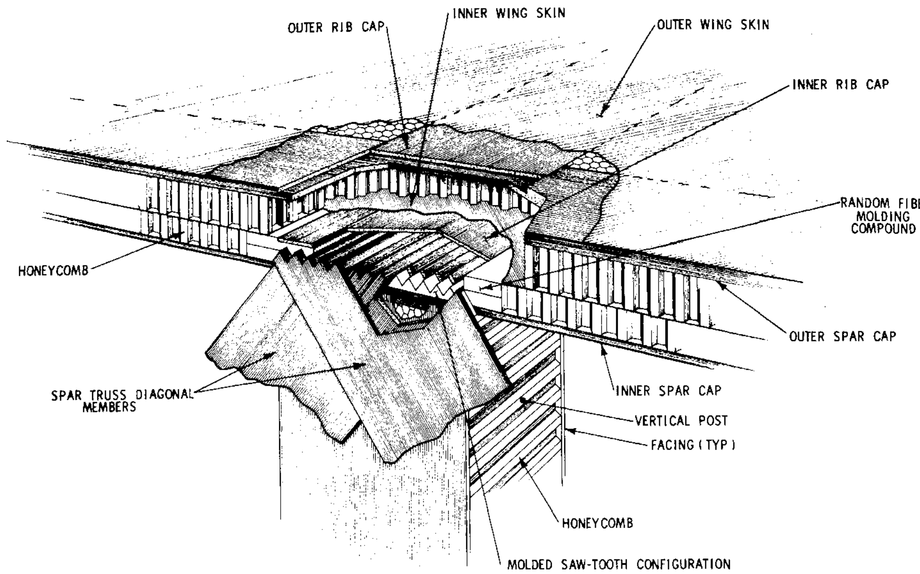


Fig. 6 Typical spar-rib-skin joint configuration at a vertical post.

The upper aft cover of the demonstration article contained linear core tapers in the span and in the chord directions. The core for this cover was formed by two planar cuts on a HOBE and by hand sanding to blend the intersection of the two planes. Grooves were routed into the outer surface of the honeycomb to accommodate structural elements, such as rib and spar caps, to provide a smooth, external, aerodynamic surface. A guided, portable router with a valve stem cutter was used on the expanded and bonded honeycomb to make the grooves. Where tolerances on groove placement or location on the finished part will permit, machining the grooves in the HOBE was desirable. Around the wing leading edge the use of flexible aluminum honeycomb makes fabrication easier (Fig. 8).

The design of the assembly winding mandrel was most dependent on the internal structural configuration. If a washout mandrel was used, the mandrel could be washed out of openings at the rear spar, the root, the tip, or through lower skin access doors. A reusable mandrel was preferred for production use to avoid recurring labor costs for mandrel preparation.

In the tapered demonstration article the only practical opening for a removable assembly mandrel was at the root end. Rib webs must be omitted until mandrel removal. The inside surface between spars was smooth except for three rib caps, 1.5 in. wide by 0.030 in. thick. The rib caps are located inside the cover inner skin to avoid additional machining on the cover honeycomb core and to allow the rib cap windings to be wound and cured in a recess or groove in the winding mandrel. The mandrel was made collapsible to release from the internal surface.

Joints and Adhesive Bonds

High-peel-strength adhesive film was used to join basic structural elements where peel and interlaminar shear forces exist. In the filament winding process, roving was placed on a mandrel in the prepreg or wet state so the roving layers conform, under curing pressure, to the mandrel surface. This inherently avoided a tooling problem of close tolerance fit associated with assembly of precured parts. It was desirable to make all joints in a similar manner. For example, the lower cover access door frames on the demonstration article were molded in place on the assembly mandrel in order to assure conformance of the door frame to the inner skin to which it was bonded. In sequence, the outer skin was wound in place on the door frame.

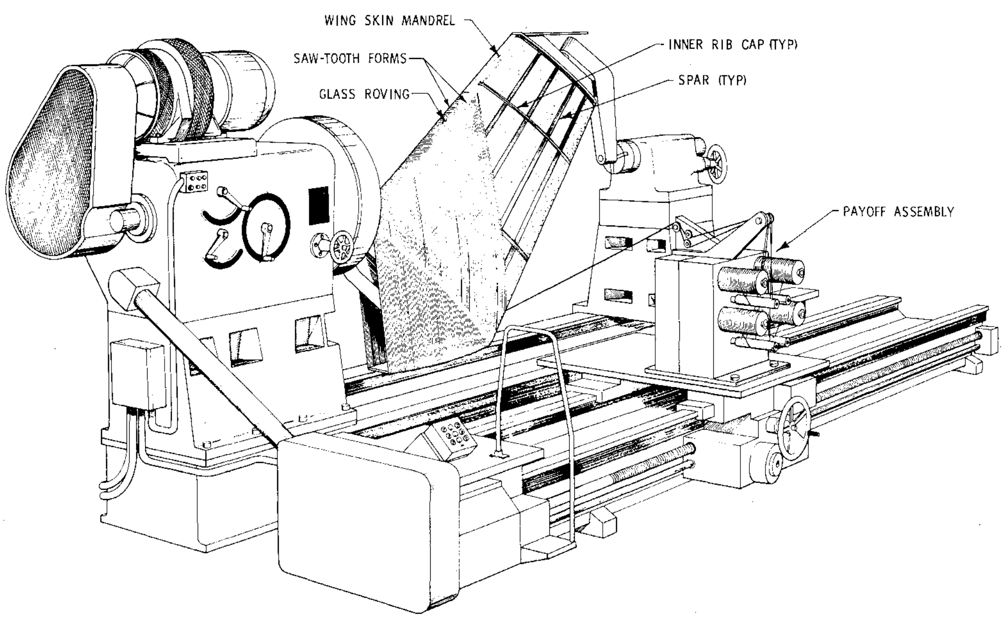


Fig. 7 Winding glass roving on the wing skin mandrel.

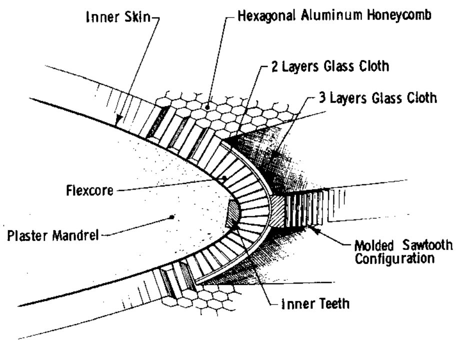


Fig. 8 Typical cross-sectional view of leading edge of wing prior to final skin wrapping.

Bonded joints in composite construction provide the highest joint efficiency. Stress concentrations and the need for high bearing allowables in pin joints should be avoided where possible. Field break joints may be mechanically fastened. However, bonding methods suited to field use should be considered. For example, a butt splice joint with one or more splice layers of fabric can be separated or opened by cutting the splice fabric. After replacing a damaged structural component, the joint may be remade by stripping off the remaining splice fabric, cleaning and bonding in place new splice fabric. Mechanical bonding pressure in the field may be applied by a sponge rubber pad, a pressure bag or a vacuum bag. Heat may be applied by heat lamps with joint temperature measured by thermocouple. No field break joints were in the demonstration article.

Of the two available methods of making hard points or fittings, the hand layup of fabric (or parallel filament mat) was selected over molding (using chopped fiber molding compound) because of the high cost of matched molding dies and of the relatively low tensile strength of molded parts.

Filament winding was used to make the demonstration article aileron hinge fittings by winding the rib caps, of unidirectional roving, around the hinge fitting bolting block. Failure in test occurred in the hinge bolt, not in the composite fitting. Except for this one successful fitting application, filament winding could not be incorporated into fittings in the demonstration article.

Winding Methods and Mandrels

The glass filament winding was accomplished by turning the mandrel and paying off the roving from a traversing winding head and by paying the roving off a winding head which circles a fixed mandrel (a ring winder).

The unidirectional placement of the glass fibers during filament winding is desired because it achieves maximum laminate strength. The filaments were hoop wound on the wing at the desired angle by offsetting or canting the mandrel to the winding axis as shown in Fig. 7. A sawtooth winding aid or mandrel insert (or an alternate method) was used to prevent filament slippage during canted hoop winding as shown in Fig. 8. Canted hoop winding was a rapid, continuous method for fabrication of the skin and other structural elements.

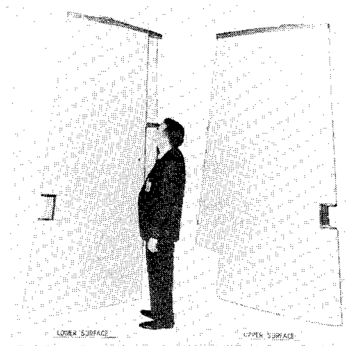


Fig. 10 Demonstration article.

An assembly mandrel with sand acrylic resin washout mandrel material was used for this manufacturing process development program. The mandrel was designed to be collapsible and removable from the completed cured structure, as for a production program, to gain applicable production experience. Internal mandrel parts locate the spars and fittings prior to winding the external skin at completion of the assembly. Relatively simple steel plate mandrels were selected for fabrication of spars.

Vacuum or pressure bag (autoclave) resin curing processes are necessary on the relatively flat areas on aircraft surfaces. Elevated temperature cure is universally used to obtain high strength composites and to improve elevated temperature strength in the composite.

Fabrication of one $92\frac{1}{2}$ in. long demonstration article (Fig. 10, top and bottom views) followed fabrication and static test of three 39 in. long articles (the inboard half of the demonstration article).

Compression Strength-Weight Efficiency

The relative compression strength-to-weight efficiency of a composite structure loaded in compression was a multidimensional problem involving, 1) panel size, 2) provision of buckling stabilization of thin structural elements by a honeycomb core sandwich (as in the demonstration article) or by a corrugated core sandwich, and 3) the filament orientation in the sandwich skin (to resist compression load or combined compression and shear as in the demonstration article).

The demonstration article was designed with 43% of the S-glass skin material oriented at 0° (in spar caps) and the balance distributed at ±45° to meet torsion and bending load requirements.

Figure 9 illustrates the relative weight of sandwich panels for varying panel compression load intensity (in pounds per in. panel width) and for several reinforcing materials. The size of sandwich panel was 12 in. by 21 in. as in the demonstration article at wing station 115.

The flight weight portion of the demonstration article weighed 136.5 lb. This is 40% less than the 227 lb. weight of the same portion of aluminum structure designed to similar criteria.

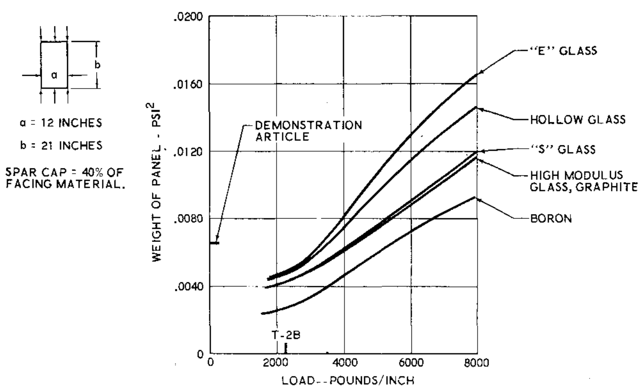


Fig. 9 Honeycomb panel weight vs panel load.

Substantiation of Design

The critical structural elements of the detail design were substantiated by structural tests. The most important of the test results are summarized in Table 2 and indicate acceptable margins of strength. Note that average values are listed because minimum guaranteed strengths have not

Table 2 Bench test substantiation of design

	Item	Test value average or noted	% of design	Design value	Comment
1.	Spar cap tensile strength	194,000 psi	121	160,000 psi	NOL ring test
2.	Spar cap compression strength	112,000 psi	140	80,000 psi	Necked coupon
3.	Spar diagonal tension load	7,050 lb	176	4,000 lb	Equals 47% F_{TV}
4.	Spar post compression load	11,360 lb	285	4,000 lb	Post test
5.	Cover sandwich skin compression	30,000 psi	124	24,000 psi	Coupon, 2 plies at ±45°
6.	Cover sandwich skin shear	43,000 psi	187	23,000 psi	1 panel with cutout
7.	Aileron hinge load	5,262 lb	138	3,800 lb	2 tests valid
8.	Forward tip tank fitting load	3,050 lb	98	3,119 lb	Reinforcement added
9.	Aft tip tank fitting load	3,645 lb (1st rev)	101	3,625 lb	A 2nd revision was made later
10.	Spar-rib tests				
	Spars	3,106 lb	124	2,600 lb	3 tests
	Ribs	7,760 lb	127	6,000 lb	3 tests
11.	Cap compression after exposure,				
	Jet fuel soak, 28 days	99,106 psi	124	80,000 psi	Necked coupon
	Sun and water, 28 days	90,442 psi	113	80,000 psi	Necked coupon

been determined. The percent margins over design values will decrease when compared to minimum guaranteed strength. The three 39-in.-long analog structures were fabricated first to illustrate repetitive fabrication experience. These structures carried, in structural test, 69%, 89%, and 130% of the ultimate load design goal (T2B wing condition 3207L) as fabrication and test problem areas were solved. The demonstration article, shown in Fig. 10, had premature cap buckling (beyond the area included in the analog articles) at limit load or 67% of ultimate load. The demonstration article achieved 106% of the ultimate design load goal or 159% of design operating load. High static strength of filament wound structure has been substantiated by detail tests.

Under a prior, separate contract with the Naval Air Development Center, similar glass-fiber-reinforced-plastic box beam structures were designed, fabricated using hand lay-up techniques, and both static and fatigue tested to provide structural data on the behavior of representative wing-type composite structures. The beams were a basic two-spar configuration with unidirectional S-glass skin-sandwich surface panels. S-glass and epoxy resin was used for the fabrication of all parts. The fatigue life requirement was equivalent to 750 loading blocks of 20 flight hours each. After 1698 blocks (over two times required life) cycling was discontinued with the conclusion that the fatigue strength of this type of GFRP structure is more than adequate to meet military requirements.

Conclusion

Over-all, these favorable test results clearly substantiate the value of the unidirectional laminate manufacturing methods developed and applied. The results indicate a way to successful production application. Several advantages are indicated: a demonstrated 40% weight savings; a decreased estimated cost; a greater bending stiffness and torsional stiffness-to-weight ratio than comparable aluminum skin-stringer construction designed to similar criteria.

The program was considered a complete success in that it proved that properly evolved manufacturing processes could produce highly efficient, lightweight and stiff aircraft structures at competitive cost if one designs for proper utilization of composite materials at the outset rather than subscribing to the process of substituting composite elements for metal structural elements.