Design of an Aircraft Utilizing Fiberglass Reinforced Plastic Primary Structure

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A description is presented of the salient structural features and technical approach employed in the design of a high-performance light aircraft stressed to 8 g limit loads. This design exercise is analytical in scope only and is intended to show the feasibility of constructing primary airframe structures from fibrous composite materials. The results of minimum weight analyses of both fluted core and honeycomb core sandwich panels are presented, and their use in the selection of final sandwich geometry is detailed. Curves are presented showing the variation in elastic properties with changes in fiber orientation angle for typical basic laminate patterns employed in the air-frame components. The tradeoff in fiber orientation angle to obtain the best balance between transverse buckling stiffness and longitudinal stretching stiffness is discussed, together with expected strength performance of the oriented fiber laminates under static and dynamic loadings.

Introduction

THE use of fibrous composite materials in airframe construction has been demonstrated amply by the use of fiberglass reinforced plastic (FRP) throughout the structural spectrum from radomes and fairings to primary wing and fuselage panels (B-58, 707, proposed C-5A and SST designs) and complete aircraft (Piper Aircraft's PA-29 design). Obviously, the use of fibrous composites in the construction of a complete airplane has not approached the level of their use in the fabricating of secondary structural components. This is not caused by any inherent disadvantage in fibrous composite construction, but rather by the requirement that a fresh new approach be taken to the structural design problem, an approach which must encompass 1) the design of the structure, 2) the design of the material in the structure, and 3) the design of the structure/material combination for minimum weight. The paper shows how this approach is followed in the design of a fibrous composite airframe utilizing FRP materials, and also presents a structural weight summary of the design.

The vehicle chosen for the design study is a high-performance light aircraft, as shown in Fig. 1. This allowed a fairly complete structural analysis to be made of an entire airframe without undue complexity. Salient details presented pertain to the primary structural components of the wing, fuselage, and empennage structure. The FRP material was employed for the analysis since it is the only fibrous composite material commercially available at competitive prices compared with conventional aluminum material. Equally important is the fact that it has a long handling and fabricating history, and that its structural performance capability is fairly well established. A number of advantages are gained through the use of FRP construction in primary airframe structure. These can be summarized as those relating to engineering and design, and those relating to service use and maintenance. *Engineering and design:* 1) multicontinuous fiber load paths and bonded joints provide inherent fail-safe characteristics for high load-carrying capability after damage; 2) material properties can be tailored specifically to meet loading intensity and direction; 3) high notch insensitivity with attendant low crack propagation properties; 4) primary structure can be made up of a minimum number of large, one-piece assemblies; and 5) optimum aerodynamic contours can be fabricated easily and will remain in shape at high load levels. *Service use and maintenance:* 1) complete freedom from corrosion, 2) high structural damping properties (especially in sandwich construction) which greatly reduce vibration and noise transmission, 3) low thermal conductivity, 4) high resistance to impact and localized damage, 5) increased flight performance due to reduced parasite drag coefficient of smooth mirror-like skin finish, and 6) ease of minor field repairs.

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Design Concept

The subject flight vehicle is an aerodynamically conventional, low-wing monoplane designed in accordance with the existing Part 3 of Federal Air Regulations. The structural flight regime of this aircraft contains the ultimate design load factors of $+12$ g and -6 g. These high design levels were chosen to determine the structural performance of FRP under uncommonly high flight loadings such as those encountered in severe aerobatic maneuvers. Minimum weight analyses were performed on various panel configurations to assess the effect of orthotropic material tailoring on structural performance. Tradeoffs then were made between the results of these analyses and the requirements of minimum skin gages, material costs, and fabrication costs.

In establishing structural configuration, particular attention was devoted to minimizing the number of mechanical joints and providing maximum fiber load path length and adhesive joint bonding areas. This planning, of course, leads to the predominant use of large-size structural components, which is optimum from the tooling and fabrication standpoint. Sandwich construction was employed to achieve maximum stiffness/weight performance from the FRP material, and to enhance its fatigue endurance by providing full stabilization for the skin surfaces.

Material Properties

Elastic Properties

In order to perform any structural design analysis with an orthotropic material, it is necessary to know how its elastic and strength properties are affected by variations in the amount and arrangement of the constituent materials. In the subject analysis, these constituents are E-glass fibers and epoxy resin.

The basic equations of orthotropic elasticity, as presented in Ref. 1, were utilized in Refs. 2 and 3 to formulate specific

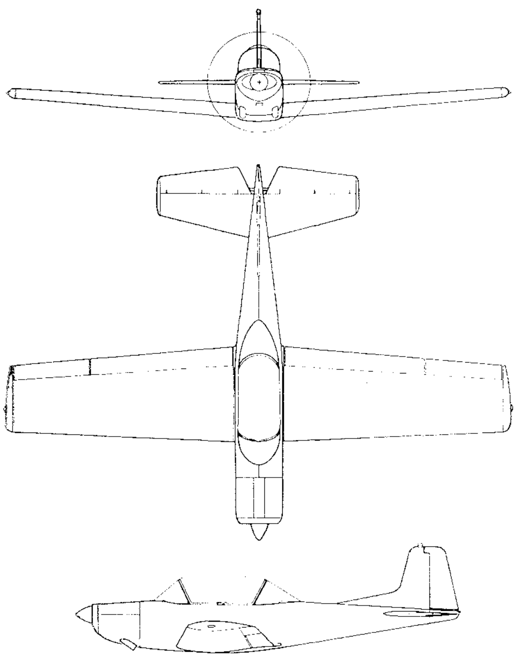

Fig. 1 Three-view of design vehicle.

expressions for Young's moduli, Poisson's ratios, shear moduli, and bending stiffnesses. These expressions relate the elastic properties of a multilayer laminate to a set of orthogonal axes arbitrarily oriented to a loading direction. They apply to laminates made up of layers of nonwoven, unidirectional fibers oriented at various angles to a reference (or load) direction. With very minor modifications, these same equations will apply to laminates made up of layers of conventional woven glass fabric. For the nonwoven unidirectional material, the basic information that must be known are modulus of the fiber material and resin material, Poisson's ratio of the fiber material and resin material, volume percentage of the fiber material in the laminate, and number of layers (or plies) in the laminate and the orientation of each.

This information can be input to a digital computer program that readily calculates the desired elastic properties. Douglas has done this for the three fiber materials, boron, *S* glass, and *E* glass, laminated with epoxy resin. Several basic fiber orientation patterns were analyzed for these fiber systems, and Figs. 2, 3, and 4 show the plots of E_x, E_y, μ_{yx}, and G_{xy} for three of these patterns in *E* glass/epoxy at a fiber volume fraction of 51%. With plots of this type, the structural designer can see how the orthotropic elastic properties vary with fiber orientations, and is able thereby to pick

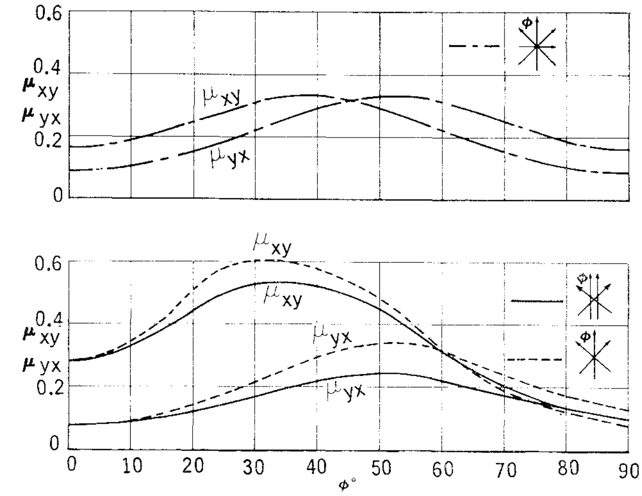

Fig. 3 Variation of Poisson's ratio with fiber orientation angle.

the pattern type and fiber angle that best suits his applied loading condition. Plots similar to the foregoing also can be developed easily for laminates made up of woven fabric layers or combinations of woven and nonwoven layers. The actual values of the elastic properties in the laminate will vary slightly from the calculated values, depending upon how the laminate was made. Also, the effects of moisture and weathering must be accounted for in establishing design values from the calculated ones. The main point to be made here is that, for a given laminate fiber pattern and fiber volume fraction, the magnitude and variation of the elastic properties with angular orientation can be calculated. Supplementary laboratory tests at one or two values of ϕ on the shop-fabricated laminate then will determine the percentage reduction to be expected in the curves from processing variables. Finally, the environmental test results are applied to account for moisture and weathering effects, thus establishing the final level of the calculated curves for design purposes.

Strength Properties

The strength properties of a fibrous composite material are much more difficult to calculate than its elastic properties because of the complexity of the material's microstructure and its failure mechanics. In addition, its strength level depends upon the nature of the applied load since, in general, a fibrous composite exhibits unequal tensile and compressive strengths. There is wealth of information available on the strength properties of FRP material, and it will be found in the current issue of *Military Handbook 17*. Unfortunately, this publication limits its scope primarily to woven fabric laminates, and tab-

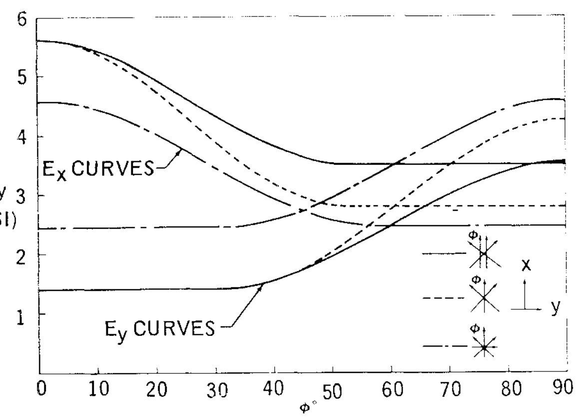

Fig. 2 Variation of elastic moduli with fiber orientation angle.

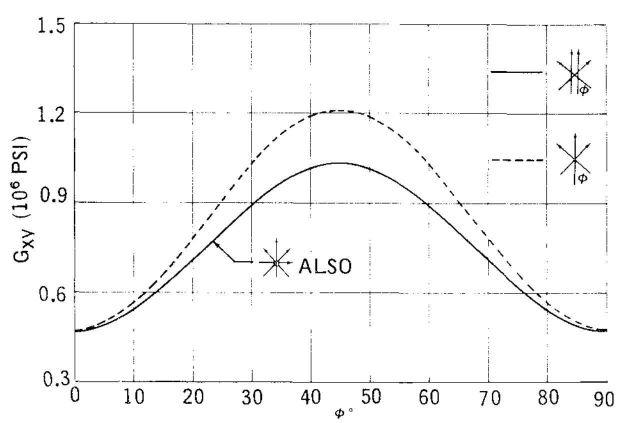

Fig. 4 Variation of shear modulus with fiber orientation angle.

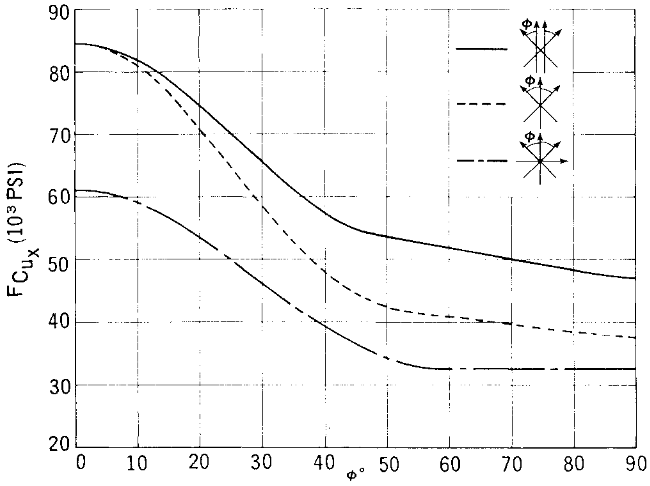

Fig. 5 Variation of compression strength with fiber orientation angle.

ulates strength and elastic properties that have been reduced proportionally to conform to applicable military specifications. The best source of up-to-date information on the mechanical properties of woven fabric laminates is contained in Ref. 4. This publication lists unreduced tensile, compressive, and edgewise shear strengths (wet and dry) for E, D, and S-type woven glass laminates. Also listed are the tensile and compressive moduli values.

Two methods that can be used to predict the variation in laminate strength with changes in layer orientation angle ϕ, for nonwoven filament layers, are the distortional energy analysis and the limiting strain analysis. The first of these is discussed in great detail in Ref. 5 wherein equations are derived which account for the effects of laminating temperature and performance under combined loading conditions, including bending loads. The mechanics of the analysis are quite complex and require the use of a computer to obtain results in any reasonable length of time. The agreement of the analysis with test results appears to be good. The limiting strain analysis is based upon the criterion that, when a given stress is applied to a multilayer laminate, the limiting strains parallel or perpendicular to the fiber direction of any layer must not be exceeded. The limiting strains parallel (L) and perpendicular (T) to the fiber direction are functions of the ultimate strength of each layer in the L and T directions, the angle between the fiber direction of the layer and the loading direction of the laminate, and the modulus of the laminate in the direction of loading (Fig. 2).

In its present form, this analysis predicts compressive strengths that agree very closely with observed test results, but it predicts tensile and shear strengths that are conservative. Figure 5 presents a plot of the ultimate compressive strength of 3 basic laminate patterns that were analyzed for use in one of the wing structure configurations in the subject design. Notice that, for values of ϕ above 55°, the strength level remains essentially constant for each pattern. Similar curves can be constructed for woven fabric laminates and also for combination woven/nonwoven laminates.

Fatigue Properties

Quantitative information cannot be presented at this time because of the relatively small amount of fatigue testing that has been performed on nonwoven oriented (NWO) fiberglass laminates to date. The primary source of NWO fiber fatigue data is the work of K. H. Boller.[6–8] These studies, based on axial loading tests, show the effect of tensile mean stresses and precyclic stresses on the fatigue strengths of E glass/epoxy laminates with NWO fiber orientations of $\pm$ 5°, $\pm$ 10°, $\pm$ 15°, and 0–90°. Summary results showed that:

1) The NWO fiber orientation that showed the best performance was the $\pm 5°$ pattern. With one resin system (3M Company's type 1009) it exhibited an endurance limit of approximately 30,000 psi at zero mean stress; with a second system (3M Company's type 1002) the endurance limit was about 25,000 psi. Optimum resin content for maximum fatigue performance is between 28 and 33%.

2) Endurance limit stress for the 1002 resin laminates of 0–90° orientation was approximately 20,000 psi, whereas the 1009 resin laminates of the same orientation exhibited an endurance limit of about 23,000 psi. All cycling was at zero mean stress.

3) Precycling the laminates at a given stress level for 3 to 1000 cycles, and then testing to failure at some other stress level, caused improvement in fatigue strength in a number of cases over samples not subjected to precycling. A larger statistical base is required to confirm this behavior, but it does tend to indicate that FRP laminates will not fit the usual cumulative damage laws of fatigue used in metal designs.

At best, however, these results can only serve as guides because they are based upon fiber patterns not employed in the subject design. It is the fiber pattern that primarily determines the level and shape of the S-N curve; and these properties of the S-N curve plus the applicable fatigue damage criteria allow the fatigue life of the structure to be determined.

Airframe Configuration

Minimum Weight Analysis

Truss core sandwich

Two basic sandwich structure configurations were analyzed for use in the wing of the subject design. These were truss

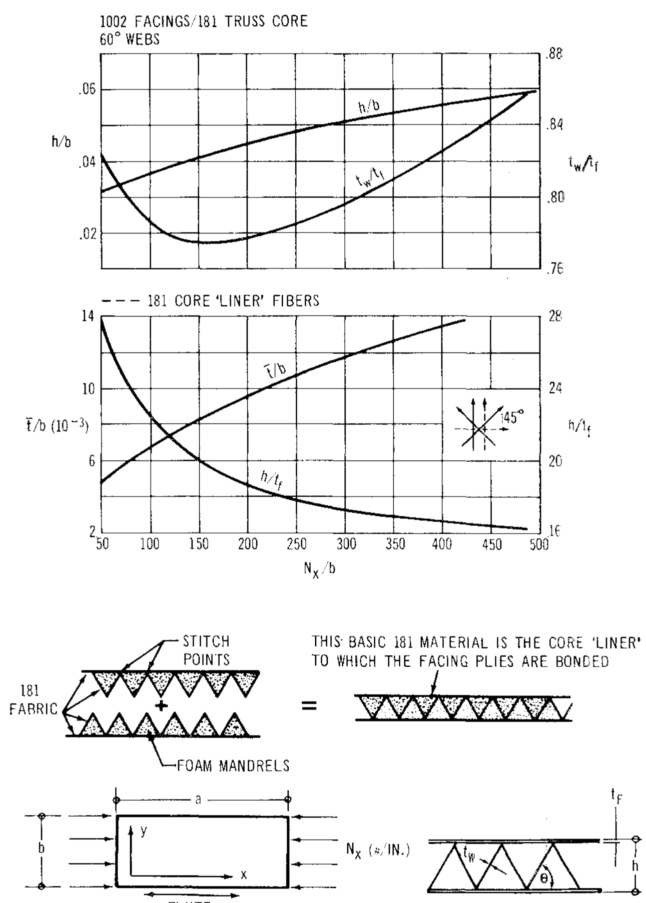

Fig. 6 Minimum weight truss core sandwich panel dimensions and geometry.

core sandwich paneling and honeycomb core sandwich paneling. A minimum weight analysis was made of each type in order to determine optimum proportions under edgewise compression loading. The minimum weight criterion is based upon the simultaneous occurrence of the over-all panel buckling and local instability modes of failure.

For the truss core configuration, the analytical approach of Anderson,[9] was employed after being modified suitably to account for the orthotropic nature of the core and facing materials. References 10 and 11 were used to provide the detail analytical background to Anderson's work. The results of his work had to be modified in two main areas to account for material orthotropy. These were in the calculation of the over-all buckling characteristics of the sandwich plate, and in determining the local buckling interaction between facing and core flute elements. The governing parameter that determines the minimum weight performance of a truss core sandwich panel is the ratio of the core web flexural modulus perpendicular to the loading direction to the facing flexural modulus in the same direction. The truss core shape is formed in two sections (see Fig. 6), with each section consisting of two layers of style 181 woven fabric stitched together around triangular-shaped polyethylene foam mandrels. The sections then are bonded together to form the completed core slice. Additional facing plies then are bonded on either side of the core as dictated by the stress analysis. Where the edgewise bending load on the panel is not severe, one of the core sections can be omitted entirely, resulting in a semi-sandwich panel, or merely a stiffened FRP skin wherein the truss flutes are the stiffeners. Panel notation and geometry also are shown in Fig. 6, together with the plot of minimum weight sandwich dimension parameters vs load intensity N_x/b. In this analysis, the contribution of the 181 core section liner to the elastic properties of the nonwoven oriented facing plies has been accounted for. The parameter $\bar{t}$ shown on the plot of $\bar{t}/b$ is the sandwich cross-sectional area (excluding foam) per inch of panel width.

Honeycomb core sandwich

Honeycomb core sandwich panels were optimized for minimum weight under edgewise compression in a manner similar to the truss core analysis. The honeycomb panel analysis is more complex, however, since there are three local instability modes (i.e., intracell buckling, face wrinkling, and shear crimping) which can exist in a honeycomb panel under edgewise compression. Reference 12 established a solution to this problem, and its analysis was employed in the subject design. To utilize this analysis, it was necessary to have the stress-strain and stress-tangent modulus characteristics of the facing material. For the core, it is only necessary to know the shear modulus of the material from which the core is made. Two fiber patterns (NWO material) were chosen initially for the critical wing section based upon the need to obtain the highest possible facing shear modulus (G_{xy}) consistent with a reasonable value of spanwise Young's modulus (E_x). This led to the choice of a laminate pattern #1 with $E_x = 3.62(10)^6$ psi and $E_y = 1.73(10)^6$ psi. It then was anticipated that this pattern might not possess a sufficiently high enough value of E_y to prevent large transverse panel bending deflections in the wing root area caused by wing flexure crushing loads. A fifth NWO ply therefore was added in the transverse direction, resulting in a laminate pattern #2 with $E_x = 3.18(10)^6$ and $E_y = 2.51(10)^6$. The stress-strain characteristics of these two patterns then were determined, and aluminum honeycomb of 3003 alloy was chosen for the core. This choice of core material was made as a result of the prohibitively high cost of fiberglass honeycomb core and the anticipated weathering deficiencies of paper core. Of the many aluminum alloy cores available, the 3003 type was the least expensive and its properties were more than adequate for the subject design. The results of the analysis are plotted in Fig. 7 and show the effect of the two basic fiber patterns on the minimum weight sandwich dimensions as a function of edge loading intensity N_x/b. For the honeycomb sandwich, t_c is the core thickness (height), t_w refers to the thickness of the honeycomb cell walls, and d is the inscribed diameter of the honeycomb cell. The remaining parameters are the same as in the truss core analysis.

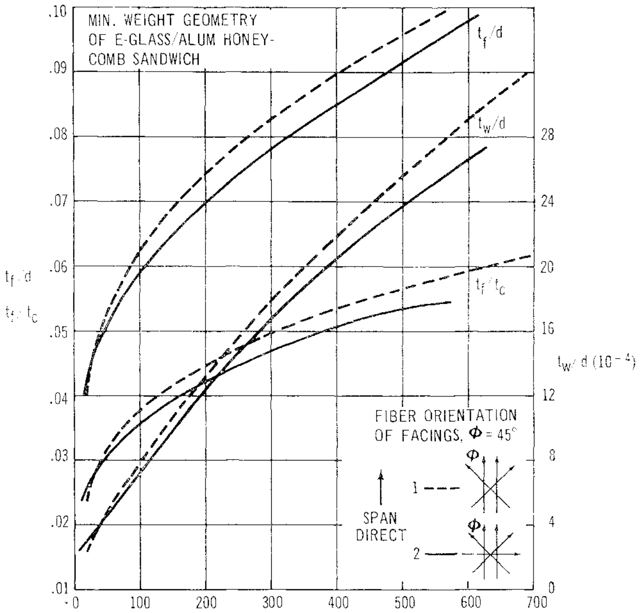

Fig. 7 Minimum weight honeycomb core sandwich panel dimensions.

Table 1 Ideal minimum weight dimensions

Truss core, in.	Honeycomb core (face pattern #1), in.
$h = 1.020$	$t_c = 0.79$
$t_f = 0.039$	$t_f = 0.030$
$t_w = 0.029$	$d = 0.48$
	$t_w = 0.0004$

Wing Structure

The total wing bending moment at the fuselage side is carried by the wing box structure between the front and rear spars, since the portion of the wing forward of the front spar is terminated at the fuselage side. Thus, the critical edgewise compression loading in the upper skin panel at the design flight condition will be carried by the width of this box, and will determine the minimum weight sandwich panel dimensions at that point. The design flight condition for spanwise wing bending is the maneuvering flight condition that produces a spanwise bending moment of 640,000 in.-lb at the fuselage side. The value of N_x/b associated with this moment load (the chordwise bending moment is neglected) is entered in Figs. 6 and 7 and the ideal minimum weight dimensions for the truss core and honeycomb core compression panels are found to be those shown in Table 1. Since the actual facing thicknesses and web thickness (for the truss core panel) must be consistent with laminated thickness of the selected fiber patterns, the actual fabricated sandwich dimensions would be those listed below, in which the h and t_c dimensions have been rounded off to the closest convenient size, and the honeycomb cell diameter has been reduced to 0.25 in. to preclude excessive dimpling of the facings during fabrication (Table 2). The d and t_w dimensions for the honeycomb core

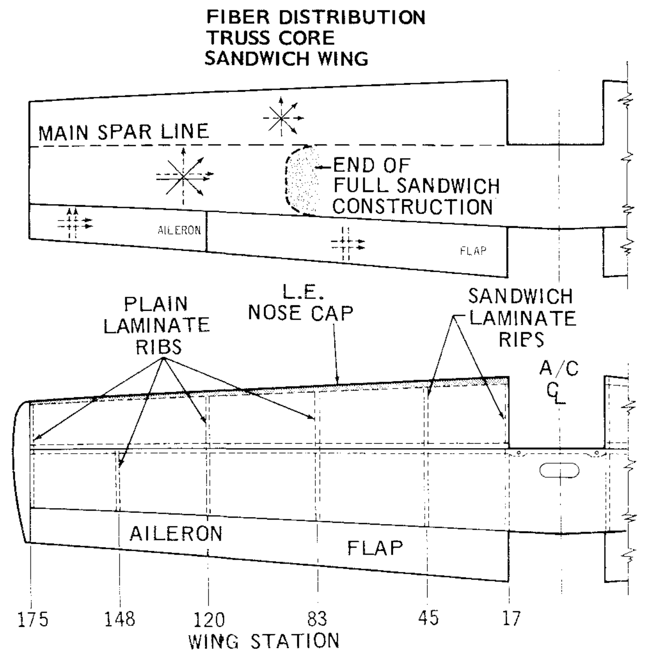

Fig. 8 Glass fiber orientations in the wing, and structural rib locations.

are equivalent to Hexcell Products' $\frac{1}{4}$-3003-0.001-NP grade aluminum core material. It was felt that the 0.001-in. thick cell wall was the thinnest that would be acceptable from a fabrication and service standpoint. The face pattern #1 was found to be satisfactory after supplementary calculations showed that crushing deflections of the panel were not excessive because of the support of the rib at wing station 45. This rib closes off the fuel tank bay that is forward of the front spar, and picks up the landing gear strut loads that are input aft of the front spar.

Obviously, the minimum weight dimensions of the sandwich panels will vary as the structural loadings decrease in the outboard direction. This would require that the core thickness progressively decrease toward the wing tip, but the added cost of tapering both the truss and honeycomb core material is prohibitive. Thus, the same core thickness is retained from root to tip, and nonoptimum weight reductions are made by reducing skin thicknesses within the limitations imposed by minimum gage requirements and allowable stresses. Figure 8 shows the final distribution of laminate fiber patterns for the truss core wing design. Each NWO fiber layer (ply) is approximately 0.009 in. thick, and the truss core 181 liner thickness is 0.010 in. Below is a summary of detail design comments for the truss core wing, and a tabular comparison of final wing weights and stress levels (see Table 3):

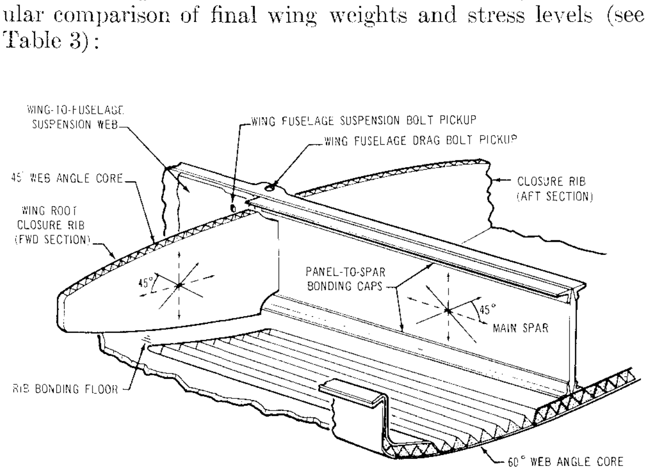

Fig. 9 Interior detail of wing root area (upper skin panel removed).

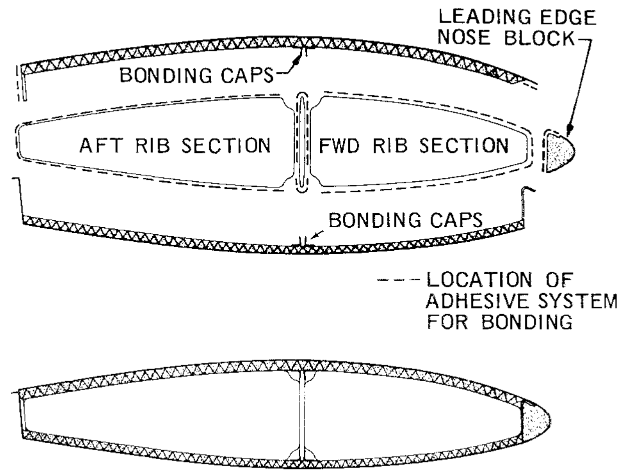

Fig. 10 Assembly of the wing at a rib station.

1) The wing section forward of the front spar carries no bending load, but does carry torsional shear. It therefore requires no full sandwich construction and effectively can carry shear stresses in the single external skin stiffened by the flute webs. Only 45° NWO fiber plies need be added to keep the skin's shear modulus high.

2) In the wing box area from Station 90 outboard, the bending loads can be carried adequately by the single skin semi-sandwich with the NWO fiber plies retained for the necessary torsional stiffness.

3) Unidirectional NWO fibers were used in the facings instead of woven fabric, in order to achieve better resin content control (the NWO material is prepreg, *B*-staged tape), higher fatigue properties, and a minimum allowable skin thickness of 0.020 in. on top of the core liner at a slightly lower weight than with equivalent woven fabric.

4) The shear tie between the inner and outer sandwich skins is stiffened locally along the main spar line by increasing the flute web thickness in this area. This is done simply by stitching extra plies of 181 fabric into the webs along this line when the core is made up.

5) Sandwich panel facing thicknesses must be increased appropriately inboard of Station 17 to transmit the bending stresses around the control system cutout at the rear pilot's locations. This cutout is shown in Fig. 8, which also shows the wing rib locations.

6) The bonding caps (see Figs. 9 and 10) are angle sections molded from style 143 woven fabric, with the warp direction oriented perpendicular to the length of the cap. This places the primary stiffness of the cap in the chordwise direction for efficient transfer of shear from spar web to cover panels, and minimizes axial loading of the cap caused by wing bending.

7) A 45° flute web angle is used in the sandwich construction of the two spars and the inboard ribs. This is done because these components are loaded primarily in shear rather than edgewise compression (see Fig. 9).

8) When ribs must be located in semi-sandwich areas, the other half of the truss core is added locally at the rib station to provide a floor to which the rib can be bonded (see Fig. 9).

9) Removable, solid polypropylene mandrels could have been employed instead of the foam mandrels. However, the added cost of the polypropylene mandrels and the fabrication

Table 2 Actual fabricated sandwich dimensions

Truss core, in.	Honeycomb core (face pattern #1), in.
	$t_c = 0.80$
$h = 1.00$	$t_f = 0.036$
$t_f = 0.037$ (incl. liner)	$d = 0.25$
$t_w = 0.030$	$t_w = 0.001$

Table 3 Summary of weights and stresses

Wing construction	σ_{max}, psi		τ_{max} psi	$\delta_{tip}{}^{a}$, in.	$\phi_{tip}{}^{a}$	Total weight, lb
Truss core	−28,200	3950	(Sta 18)	22.0	<3.5°	327
	+29,200	6000	(Sta 16)			
Honeycomb core	−39,880	7870	(Sta 18)	27.5	3.5°	295
	+41,730	8360	(Sta 16)			

[a] Based upon 10% reductions in E and G values.

cost increase brought about by their oftentimes troublesome removal are not justified by the slight weight saving over the use of the foam type.

10) Total sandwich panel thickness in the wing lower surface is 0.75 in., and $t_w = 0.020$ in., based on the minimum weight results.

From the results of the foregoing design summary, the subsequent cost analysis, and the summary in Table 3, it was decided that the truss core type of sandwich wing construction should be used, despite the 32-lb weight penalty of the completed wing structure. This is because the honeycomb core design works the facing laminate closer to its static strength level than does the truss core design, thus leading to lower fatigue endurance properties compared to the truss core design. More than this, however, the cost picture governed the selection. At a typical inboard area of the wing (between Stations 30 and 85), it was determined that the material cost per square foot of the honey-comb construction was 56% higher than the truss core construction. In addition, the fabrication labor cost is significantly lower because there are fewer plies of fiberglass and no adhesive system to lay up, and the careful handling and preparation requirements for the metal honeycomb core are eliminated. For the particular vehicle design, it is felt that the 32-lb weight penalty is a small enough one to pay in view of the cost advantage.

Fuselage and Empennage

Minimum gage requirements size the sandwich skin thickness on the fuselage and vertical stabilizer structures. Full sandwich and semi-sandwich truss core construction are used throughout, except for the secondary structural items such as the engine cowling, fin cap, and tail cone. External skin

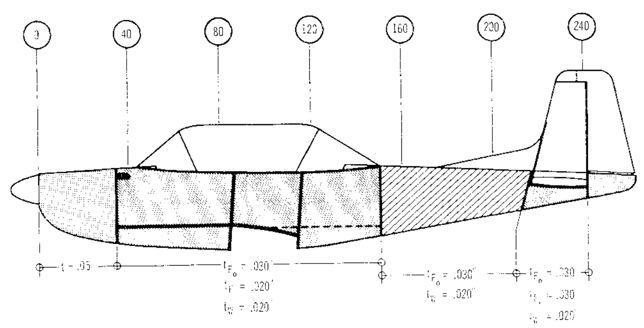

Fig. 11 Fuselage structural framing and skin thicknesses.

thickness on the fuselage is 0.030 in. throughout to provide sufficient impact resistance. Since none of the sandwich paneling in the fuselage is buckling critical, a 45° flute web angle is utilized to keep the core weight down. The height of the truss core is 0.60 in. throughout the fuselage, and the facing and web thicknesses are distributed as shown in Fig. 11. Low fuselage torsional stresses obviate the need for facing plies oriented at 45° as in the wing. Therefore, the fuselage skins are made up of parallel laminated 181 woven fabric material. The foam mandrels of 3 lb/ft³ density which are retained in the core flutes will increase greatly cabin comfort through their noise and vibration dampening properties and their low thermal conductivity.

The wing suspension frames are shown in Figs. 12 and 13. They close off the fore and aft ends of the wing box cutout in the fuselage and tie the fuselage to the wing by way of the suspension bolts that pick up the front and rear spar webs. The detail of how this would be done is illustrated in Fig. 12, which depicts the front spar pickup. The rear spar pickup is identical in nature, except that the wing suspension web must extend above the top of the rear spar to reach the bolts. This is necessary because the axes of the rear suspension bolts must be in line with the axes of the front suspension bolts for proper wing flexing. The wing suspension frames would be

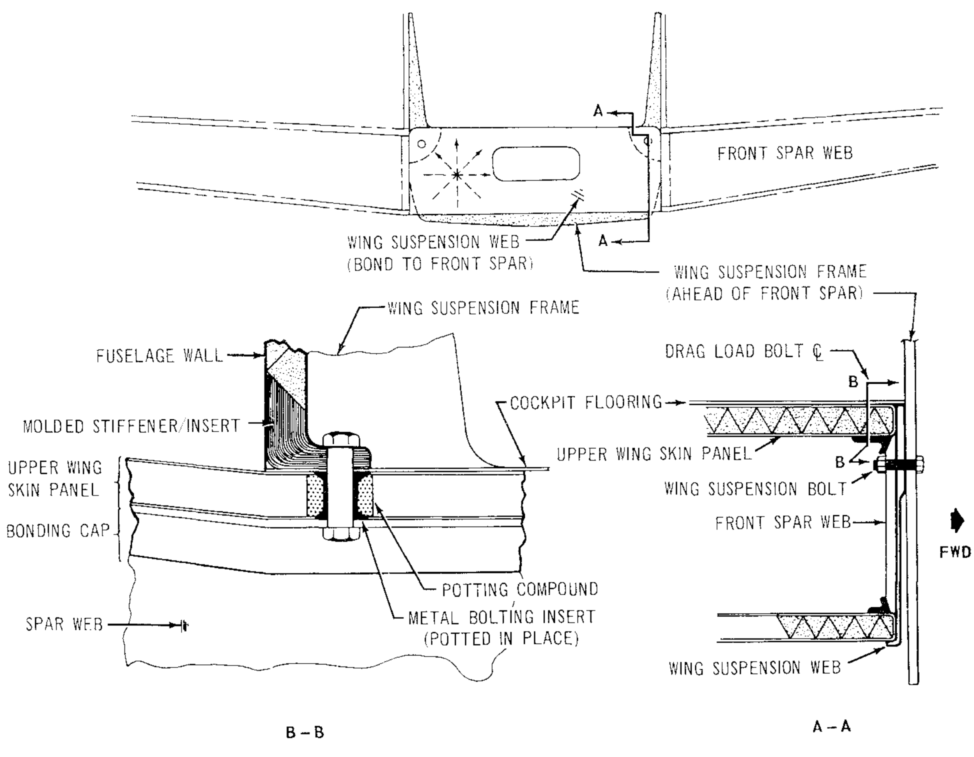

Fig. 12 Details of wing-fuselage attachment.

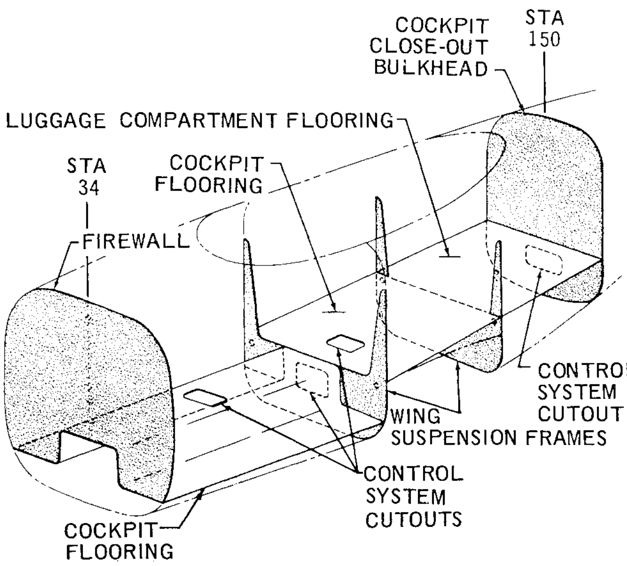

Fig. 13 Location of cockpit flooring and framing.

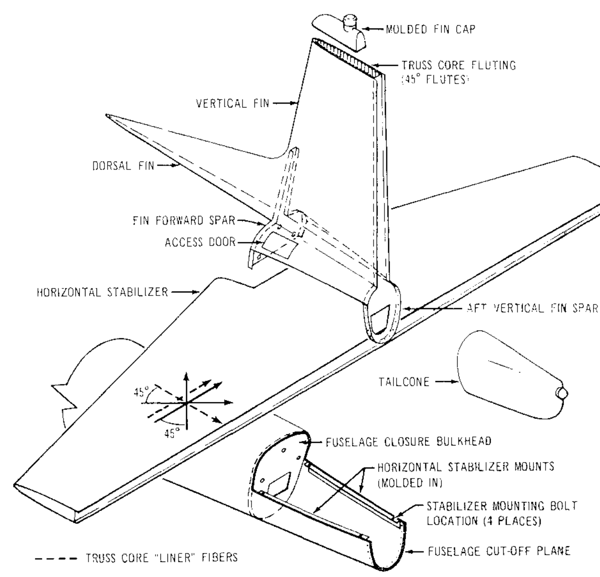

Fig. 14 Details of empennage assembly.

of truss core sandwich construction similar in cross section to the sandwich wing rib shown in Fig. 9. The truss flutes in the frame would run vertically, and would be reinforced rigidly in the region of the bolt hole by shaped laminate inserts sewn into the webbing in place of the foam mandrels. These inserts, being the thickness of the sandwich frame (0.60 in.), will carry the bolt loading at a bearing stress of approximately 15,000 psi in the front spar frame. The suspension webs would be molded plain laminate members locally thickened at the bolt hole locations and bonded to the spar webs as shown in Fig. 9. The use of this type of wing-to-fuselage attachment system will minimize greatly the input of severe aerobatic wing flexing loads into the fuselage structure, thus enhancing the fatigue performance of this critical joint area. Total weight of the fuselage structure excluding engine and nose gear mounts, and the canopy structure, will be approximately 160 lb.

Assembly details of the empennage structure and its attachment to the aft fuselage section are presented in Fig. 14. Torsional and spanwise stiffness requirements dictate the use of the NWO fiber pattern shown in Fig. 14 for the horizontal stabilizer skin. The bending stresses in the root area sandwich panels are sufficiently low that 45° web angle core can be used without local buckling being critical. This minimizes the core weight and counteracts the use of the nonoptimum stiffness-sized facing thickness. Semi-sandwich construction can be used outboard of the root area in order to further minimize weight. The upper and lower sandwich panels are continuous from tip to tip, except for the notched leading edge portion to allow room for the forward vertical fin spar attachment. As a result, these panels are quite effective in carrying virtually all of the spanwise bending loads. Therefore, the stabilizer spar serves mainly to close off the rear of the box section and carry the elevator hinges. Because of this function, it is built in a similar manner to the wing bonding caps, i.e., it is molded from 143 style woven fabric that is oriented with the warp direction perpendicular to the direction of the span. This produces a spar member with high flexural stiffness in a vertical chordwise plane to carry efficiently the concentrated load inputs from the elevator hinges.

Low torsional loads on the vertical stabilizer obviate the need for any 45° fiber orientation in the skin laminates. Consequently, the skins can be made up of parallel-laminated 181 style woven fabric of 2-ply thickness added to the truss core liner. Semi-sandwich construction can be used throughout the structure, with the 45° web angle flutes providing effective stiffening against surface oil-canning. The vertical stabilizer is molded in two skin halves with the dorsal fin skins formed integrally with each half. The stabilizer skins are bonded together on the front and rear spars to form the completed fin unit. This then can be bolted in place at the aft fuselage bulkhead and the fuselage cut-off plane after the horizontal stabilizer has been mounted in position.

Utilizing the type of fiberglass composite construction discussed in the preceding sections, an aircraft gross weight of 1940 lb is obtained. The empty weight of the design is 1290 lb, based upon the use of a fixed landing gear structure utilizing solid fiberglass laminate flexure struts.

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