SOME ASPECTS FOR LOWERING SAILPLANE COSTS.

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Substantial gain has been made in glider performance through aerodynamic refinement and the development of higher efficiency airfoils while the airframe design and fabrication techniques have remained basically the same for the past 30 years. Accepted aircraft materials and the limitations imposed by them have restricted airframe design to present day standards. We are, however, believed to be at the threshold of creating what may be considered as being a revolutionary airframe structure for sailplanes through the use of plastics. This paper presents the theoretical feasibility of a plastic foam now on the market with regards to its use as a primary structural material in wings. Its successful application in this respect will contribute strongly to lowering the design and production cost of gliders in all performance categories without penalizing their performance.

High cost airframes will continue so long as designers and engineers are compelled to employ conventional materials and methods. There is hope for escape through a new plastic foam, designated by trade name, 'STYROFOAM'. It is an expanded thermoplastic, polystyrene foam very light in weight and possessing strength properties which show it to be suitable as primary structure in glider wings. 'STYROFOAM' is a development and product of the Dow Chemical Company, Midland, Michigan, U.S.A. The material is supplied in the form of planks and logs which are easily cut to any desired shape. The material is also easily bonded to wood and many other materials using cold setting glues.

The aspects of 'STYROFOAM' as a primary structural material is evaluated theoretically by basic stress methods using design criteria from a glider wing now under design. The wing is to be constructed similar to that illustrated in Figure 1 and is to be tested to determine the complete strength behavior of the composite structure under static loads.
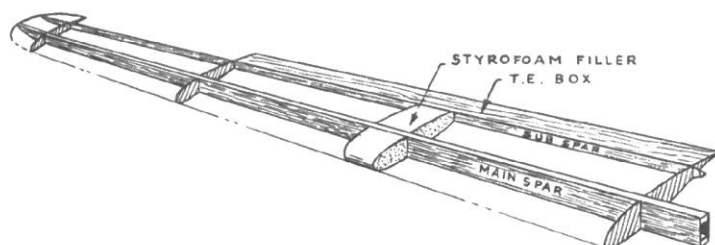

FIG. I SIMPLIFIED WING STRUCTURE

Figure 1 postulates the arrangement of a glider wing considered best for simplification through the use of 'STYROFOAM' as a principal part of the primary structure. The structure is composed of a conventional wood box type main spar for carrying span wise airload shear and bending. A sub spar is employed to assist the main spar in reacting the drag load couple and to serve as a means for attaching the T.E. structure. The two outboard ribs support and distribute the aileron loads to the skin, foam filler, and the main spar. The root rib serves, as in a conventional wing, to redistribute the wing torque to the main and sub spars.

The principal function of the foam filler is to transmit the chord wise airload distributed along the span, to the main and sub-spars. The natural question is, of course, 'Is the material adequate in this respect?' As yet there is no concrete proof but stress analysis do show the material to be very promising with respect to the design air and inertia loads normally experienced in the realm of sailplane operation. To illustrate the potentiality of this foam in this respect, loads and stresses for the aforesaid wing are presented.

The airfoil section, shown in Figure 2 (see next page), and the chord wise pressure shear and bending values are taken from the subject wing design criteria. The design loading is 4.45 $\#$ / ft^2 and the limit airload factor is 5.53 giving a design factor of 8.30. The design area is ninety sq. ft, thus, the ultimate airload is 4.45 x 8.30 x 90 = 3325 $\#$. The airfoil section is an NACA 64_3 - 618 from root to tip with 3° geometric washout from 58% semi-span to tip. The plan form has a 3 . 1 taper with 45" $\updownarrow$ root chord and 16" tip.

Span-wise chord lengths and section data are as listed in the subjoined Table I.

Table I section C_2 values correspond to limit airload wing

$$C_L = \frac{N\,n/s}{q} = \frac{5.53 \times 4.45}{11.82} = 2.08$$

where q = the impact pressure @ 68 M/H as read from the flight envelope for low-speed maneuver-high angle of attack condition.

In evaluating the stress level in the filler under the most critical load condition, a section along the span is selected where the section combined airload and geometry produce the maximum chord-wise shear and bending. This combination, for maximum, occurs at .60 semi-span station. The section lift in #/inch of span at this point is

$$\Delta L\, \#/'' = q. \Delta b. c. C =$$

$$(11.82 \times 1'' \times 27.6''/144)\ 2.101 =$$

$$4.75\ \#/''$$

for 5.53 limit load factor and 7.15 #/'' at ultimate.

The leading and trailing edge shears and bending moments in # and in .-lbs., respectively, are shown in Figure 2 for the section at semi-span 60% station. The leading edge shear and moment at the spar ₵ are.

Shear - 3.04 #/in. of span
Moment - 14.65'' - #/in. of span

and for the trailing edge
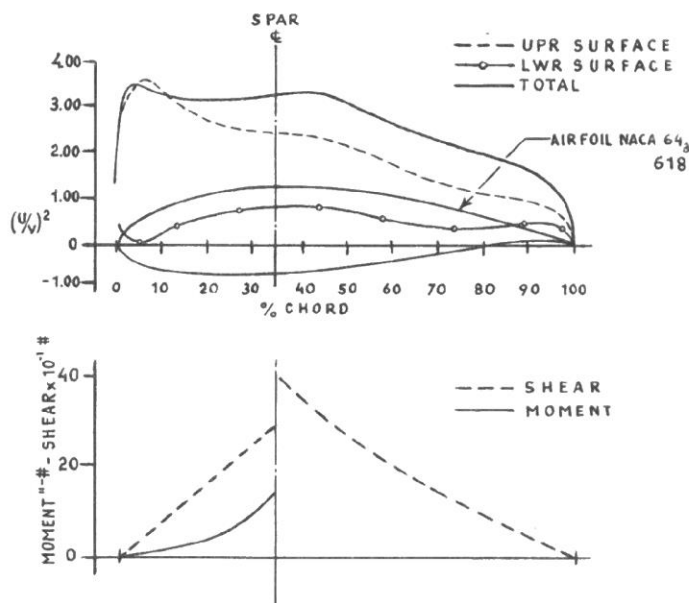
Shear - 4.13 # /in. of span
Moment - 42'' - #/in. of span

FIG. II   WING SECTION PRESSURE, SHEAR ₵ BENDING PER INCH OF SPAN.

TABLE I - WING DATA & SPAN-WISE LIFT DISTRIBUTION

Semi-span Sta	0	.2	.4	.6	.8	.9	.95	1.00
Semi-span Inches	₵	42.70	85.40	128.10	170.8	192.15	202.8	213.5
Semi-chord Inches	45	39.2	33.4	27.6	21.8	18.9	17.45	16
$C_{l\,al}$	.942	.995	1.03	1.04	1.00	.921	.787	0
$C_{l\,b}$	.0918	.071	.047	-.0591	-.110	-.127	-.117	0
$C_L\ C_{l\,al}$	1.960	2.07	2.14	2.16	2.08	1.915	1.637	0
C	2.052	2.141	2.187	2.101	1.97	1.788	1.52	0

The filler material like the air pressure has complete continuity in all directions. Since the stress developed in a loaded structure tends to follow the line of greatest rigidity, almost 100% of the leading and trailing edge loads will travel via the filler direct to the spar. We may think of the filler as performing as though the wing had an infinite number of ribs. Upon this premise, there is present maximum shear and bending stress levels in the foam at the spar $\cent$ based on the section properties of the area and section modulus of a rectangular section per inch of span which are estimated by the expressions for simple beam shear and bending:

$$f_s = \text{shear } \#/\text{spar depth in inches}$$

and

$$f_b = Mc/I$$

Since the wing has an 18% constant depth, the spar depth at the section under analysis is

$$d = .18C = .18 \times 27.6 = 4.96''$$

Thus, the filler maximum shear stress at this section is

$$f_s = 4.13/4.96 = .833 \text{ p.s.i.}$$

and a bending stress

$$f_b = 42 \times 6/4.96^2 = 10.25 \text{ p.s.i.}$$

The Dow Chemical Company curves (see Table II and Figure III) show an allowable compressive stress of approximately 15 psi at 165° F. for the 1.6 #/ft³ density 'STYROFOAM'

It is not likely that this temperature would ever be exceeded even in desert operations.

The shear stress at the spar face is too low to be worthy of mention. The bending stress margin of safety is

$$M.S. = \frac{15}{10.25} - 1 = \neq.46 \text{ or } 46\%$$

The forgoing computed stress levels were for a small sailplane with a small wing area. The wing loading is, however, above the average used for gliders. The size of glider does not alter the picture for application of the proposed filler for as the glider size increases so do its geometric proportions. Let us assume a wing for glider with two times the area
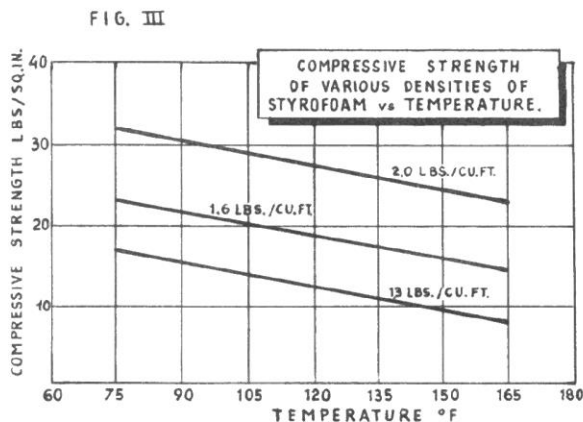

FIG. III

COMPRESSIVE STRENGTH OF VARIOUS DENSITIES OF STYROFOAM vs TEMPERATURE.

TABLE II - 'STYROFOAM' PROPERTIES - 1. MECHANICAL PROPERTIES AT 77° F.

	for density of 1.3 #/ft³	for density of 1.6 #/ft³	for density of 2.0 #/ft³
Compressive Yield Strength (psi)	10 - 20	15 - 25	15 - 35
Tensile Strength (psi)	30 - 45	50 - 70	80 - 100
Shear Strength (psi)	15 - 25	25 - 35	35 - 45
Compressive Modulus (psi)	450 - 1100	750 - 1350	1150 - 1750
Bending Modulus (psi)	200 - 750	650 - 1200	1200 - 1900
Impact Strength (in.-lb.)	0.5 - 1.2	1.1 - 1.8	2.1 - 2.7
(3/8 in. x 1/2 in. section)			

with the same aspect ratio, taper and twist, and having the same wing loading and employing the same airfoil section. Under these assumptions the

$$\text{Wing area} = 2 \times 90 = 180 \text{ sq. ft}$$
$$\text{Span} = \sqrt{AR \times Area} = \sqrt{136 \times 180} = 49.5'$$

The span-wise bending would increase roughly as the ratio of the spans[2] if the wing loadings were equal, but the chord-wise stress level in the foam, which is of principal concern, would decrease with increased span with the same wing loading for as the chord increases its bending moment and shear would increase directly, but due to a larger chord and correspondingly greater depth, the section modulus increases by the ratio of the squares of the relative spar depths.

Efforts directed toward simplified design should not be hamstrung by overcaution on the part of the designer to hold weight to too close limits for in the glider the prime consideration is performance and not pay load as in the case of the airplane. Variation in weight does not seriously influence sailplane performance. The L/D does not change with wing loading and the sinking speed is a function of the forward speed which in turn varies as the square root of the wing loading. Thus, if the weight of given design turned out twice that which was proposed, the forward speed at L/D maximum would be 1 417 times that predicted and the sinking speed would be correspondingly affected. For a design to turn out twice the estimated weight would point to incompetence. But if substantial gain can be shown in design simplification by the addition of a few pounds of weight, it will in most cases be well worth it.

Let us examine the aspects of reducing present fabrication cost in the conventional fuselage structure by taking advantage of the elastic strength of low density materials and a slight weight penalty. Perhaps the most adverse approach toward low cost glider production is extensive tooling. Elaborate tools, jigs, and presses can well become a liability rather than an asset. Unlike the automobile business, the number of gliders to be produced and sold will be limited to a few hundred units per year at the very most, and this is not in the immediate future even in the eyes of the more optimistic. An automobile manufacturer invested $ 90.000.000,-- in tooling required to produce a new model. For the number of units sold during the first year the tooling cost probably represented $ 90,-- per unit. If a sailplane manufacturer spends $ 9.000,-- for tooling, he must sell 100 units to equal this. Thus, it is highly important that tooling requirements be held to an absolute minimum. This can best be accomplished through simplified design using materials that are readily shaped and assembled with simple tools. Now consider a fuselage structure made of metal and observe the reduction in the number of parts and subsequently the tooling requirements realized by taking a slight weight penalty to simplify fabrication and assembly of parts. As mentioned, the use of low density material offers considerable gain by reducing parts and fasteners where stability is the prime factor in composite assemblies To appreciate the advantages of low density materials in this respect, let us examine a very fundamental and simple illustration of stability vs. weight. Imagine three rectangular plates having equal lengths and widths, and of equal weight but composed of different materials, one steel, one aluminium and the third of magnesium. Assume that each is loaded along its width by a uniformly distributed load of $\omega\,\#/''$. Assume that the edges of each plate has the same degree of fixity. The basic formula expressing plate buckling stresses when subjected to this type of loading is

$$\sigma_{cr} = KE(t/b)^2$$

where

b = the plate width in the direction normal to the applied load

E = modulus of elasticity

t = thickness

and K is a function of the plate aspect ratio, Poison's ratio for the material and

the edge fixity. Since the fixity and aspect ratio are taken as being the same for each plate, and Poison's ratio is sufficiently close for the three materials, then K may be assumed equal for each. The deciding variables are then E, t, and the material densities, the values for which are as follows:

Material	Wt. #/in.^{3}	E
Steel	.3	30×10^6
Aluminium	.10	10^7
Magnesium	.065	6.5×10^6

It is interesting to note that each of the above material E values is numerically proportional to its density. The applied stress for each sheet is ω/t. Since the plates are to be equal in weight, then the relative thickness of each is a function of the relative densities, i.e., the aluminium plate will be 3 times as thick as the steel, the magnesium 4.61 times as thick as the steel, and 1.54 times as thick as the aluminium plate. Writing the critical buckling equations for each of the sheets, we have for

Steel $\qquad (W_s/t_s)_{cr} = KE_s(t_s/b)^2$ .... (1)

Aluminium $\quad (W_A/t_A)_{cr} = KE_A(t_A/b)^2$ .... (2)

Magnesium $\quad (W_m/t_m)_{cr} = KE_m(t_m/b)^2$ .... (3)

Dividing Equation (2) by Equation (1)

$$\frac{(W_A/t_A)_{cr}}{(W_s/t_s)_{cr}} = \frac{E_A t_A^2}{E_s t_s^2} \quad \text{............................ (4)}$$

Substituting the values $t_s = 1/3 \, t_A$ and $E_A/E_s = 1/3$ into (4), we have

$$\frac{(W_A/t_A)_{cr}}{(3W_s/t_A)_{cr}} = 1/3 \left(\frac{t_A}{t_A/3}\right)^2$$

from which $\qquad\qquad\qquad (W_A/W_s)_{cr} = 9$

Thus! we see that for the same weight, the aluminium sheet will support 9 times more load before buckling than will the steel sheet. By the same reasoning, magnesium will carry 21.3 times as much as steel and 2.37 as much as aluminium. These ratios are not exact because of slight differences in Poison's ratio and compression yield values, but they do serve to illustrate a substantial gain can be assured through use of lower density alloy in structure where stability is the governing factor controlling weight and contributing to simplicity.

Shown in Figure 3 (see next page) are three sample aft fuselage cone sections for a glider composed of steel, aluminium and magnesium to illustrate the production savings of one over the other. Each unit is assumed to be subjected to the same shear, moment and torque loads, and to have the same over-all dimensions. Each is assumed to have the same structure weight. Structure A is constructed entirely of stainless steel. To meet the specified weight and strength specifications requires that the skin of A be approximately 1/3 the thickness of B, and .217 the thickness of C. The stringers in A being steel will be less stable / pound than in design B. Thus, for stability, A requires more stringers and more transverse skin-stringer stabilizing frames. The magnesium structure will require the

FIG. IV   SUGGESTION OF
SIMPLIFIED   STRUCTURE
VIA LOW DENSITY MATERIALS

A - STEEL

B - ALM

C - MAG
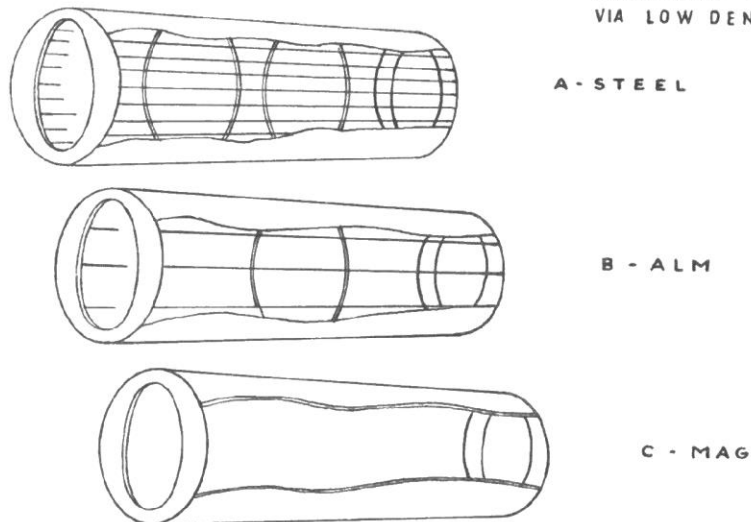
minimum number of stringers and frames than either A or B, and it is possible to construct a satisfactory unit entirely monocoque with magnesium for about the same and possibly less weight. The per pound fabrication cost is obviously much lower due to the reduction in the number of parts and fasteners. Structure A would require a far greater number of form blocks, jigs and tools. Cost does not end with tools and jigs. The greater the number of parts, the greater the need for production, engineering, clerical, and managing personnel. Too often production costs are the outgrowth of personal whims. Most designers are prone to propose and create designs that humor their own vogue, losing sight of the specific objec- tive and the end result is a product that fails to have a market.

As to how cheaply the simplest conventional glider can be developed and manufactured is dependent upon production and labor cost since material is fixed. From a manufacturing viewpoint, consider the prime factor is airframe material requirements. Suppose one is going to produce a conventional, all-metal glider weighing 300 lbs empty. Most of this 300 lbs will be in airframe for which the raw material, assuming it to be aluminium and including a 50% waste factor, will require $ 225,-- per unit at prevailing prices if pur- chased in quantities. This is a blanket poundage estimate disregarding special extrusion, fittings, controls and instruments.

The engineering costs will run $ 15.000,-- or $ 20.000,--. Modest facilities, tooling and jigs will represent another $ 10.000,--. Thus, to engineer and establish a reasonable production set-up will require $ 30.000,--. If 100 units are scheduled, this represents an initial investment of $ 300,--/unit. Thus we have $ 525,--/unit in material and development for 100 units. Assuming that each unit requires 600 man-hours to build it at $ 3,--/man- hour (this is assumed including overhead), then the finished product has cost $ 2.325,-- per unit. If the investor is to realize an equitable return on his investment, the sale price would have to be around $ 2.500,-- to $ 3.000,--. There is a very limited market for this price glider and it does not call for 100 units per year. To promote the sale of as many as 100 units per year will require a sale price of $ 1.000,--. The most likely road to successful business venture in glider manufacture and sale is through a simplified kit suitable for back yard fabrication in the back yards of the many potential customers who have more time than money.

It is concluded:

(1) That glider cost can be substantially reduced through simplified design and corresponding reduction in tooling, jig and production facilities;

(2) That for the present 'STYROFOAM' offers the most promising solution for minimum structure design and production requirements of wings and control surfaces;

(3) That low density alloys offer the best means for simplified design of metal structure since elastic stability is the predominating factor in all airframe structures;

(4) That the number of parts employed in the airframe structure must be held to a minimum to produce a low cost glider.

Discussion following Presentation of

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One question was brought up relative to the strength properties of 'STYROFOAM' above 165° F. Dr RASPET stated that the strength vs. temperature curves varied gradually and that the Dow Chemical Company curves could be extropoloted at least for a few degrees above 165° F. Another point brought up was that with even as low a density material as 'STYRO-FOAM' that cutouts would be needed to reduce the overall weight.

The effect of trapped water or moisture in the 'STYROFOAM' was raised. Dr RASPET stated that there was no possibility of this during the manufacture of the material or during production.

No mention was made of ease of maintenance.