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A SAILPLANE WING CONSTRUCTED OF FOAM CORE AND POLYESTER FIBERGLASS SKIN

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Abstract

The results from a stress analysis of a thin skin, foam core, high aspect ratio wing indicate a possible method of constructing sailplane wings. The analysis includes an approximation of the maximum core and skin shear stress, a computer program to evaluate the stress distribution and displacements of a thin-walled unsymmetrical tapered cylinder and the accountability of creep.

1. System Defined

The wing crossection was defined as a NACA 4412. The root and tip chord are 8.5 and 4.25 inches respectively with no sweep back at the leading edge as shown in Figure 1. The span was 51.1875 inches with a 17.7 aspect ratio. This particular wing had the same dimensions as a wooden wing used on one of the authors radio-controlled sailplanes. The small size made it practical to build and test.

2. Theory of Composite Beams

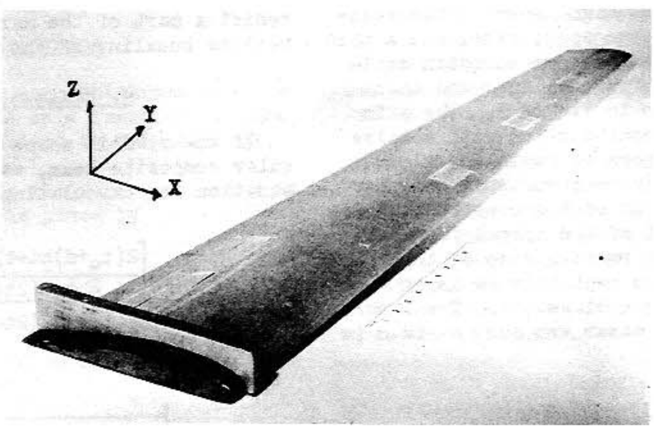
Composite beams are designed so that each material in the beam is most efficiently used with respect to weight, position, and resistance to forces.

Take for example the beam shown in Figure 2. The flange material is positioned so that it resists the maximum normal and shear stresses. Therefore a high modulus material should be used. The core material is positioned so that it spaces the flange material and resists the maximum shear stress. A lighter weight material is choosen for the core material because it occupies the most volume of the beam. But the lighter weight core materials usually have lower allowable maximum normal and shear stresses. This of no consequence because the core's normal stress is small ($E_c<E_s$) and the core's maximum allowable shear stress is within working limits. So the only two limitations

ROOT DIMENSIONS

PNT	X	Z
1	0.000	0.000
2	1.500	0.000
3	2.500	0.000
4	3.500	0.000
5	4.250	0.000
6	5.000	0.000
7	5.500	0.000
8	6.000	0.000
9	6.500	0.000
10	7.250	0.000
11	7.750	0.050
12	8.000	0.070
13	8.250	0.100
14	8.350	0.110
15	8.400	0.140
16	8.450	0.190
17	8.500	0.310
18	8.450	0.450
19	8.400	0.490
20	8.350	0.520
21	8.250	0.570
22	8.000	0.660
23	7.750	0.725
24	7.250	0.825
25	6.500	0.900
26	6.000	0.925
27	5.500	0.925
28	5.000	0.900
29	4.250	0.825
30	3.500	0.725
31	2.500	0.550
32	1.500	0.350

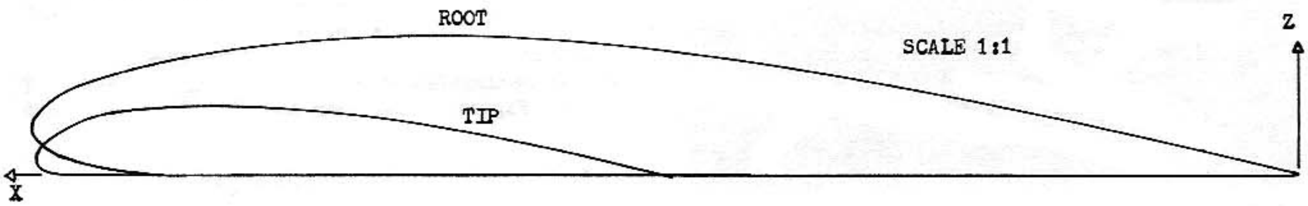
Figure 1. Wing Geometry



NACA 4412
SPAN 51.1875 in.
A.R. 71.7
SKIN THICKNESS 0.030 in.

TIP DIMENSIONS

PNT	X	Z
1	4.250	0.000
2	5.000	0.000
3	5.500	0.000
4	6.000	0.000
5	6.375	0.000
6	6.650	0.000
7	7.000	0.000
8	7.250	0.000
9	7.500	0.000
10	7.875	0.000
11	8.125	0.010
12	8.250	0.020
13	8.375	0.040
14	8.425	0.050
15	8.450	0.060
16	8.475	0.075
17	8.500	0.150
18	8.475	0.225
19	8.450	0.240
20	8.425	0.260
21	8.375	0.280
22	8.250	0.320
23	8.125	0.350
24	7.875	0.409
25	7.500	0.450
26	7.250	0.460
27	7.000	0.460
28	6.750	0.450
29	6.375	0.420
30	6.000	0.375
31	5.500	0.290
32	5.000	0.180



are the flange's maximum allowable normal stress and the core's maximum allowable shear stress. With these two design requirements a beam can be designed with a tolerable reduction in maximum strength and a substantial savings in weight.

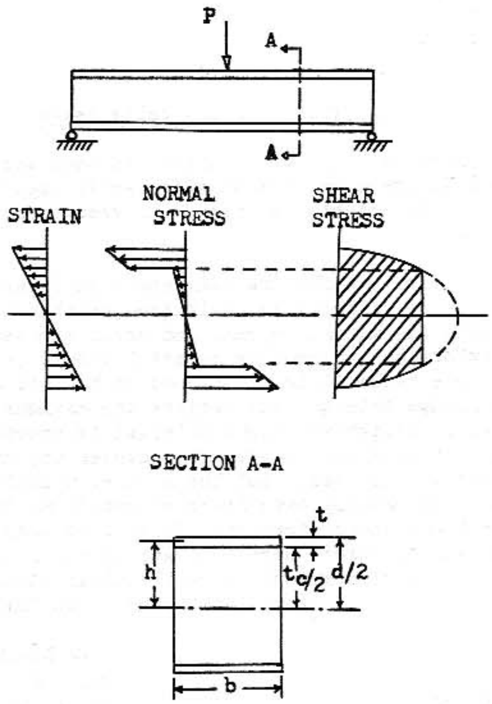


Figure 2. Composite Beam

The construction of most high aspect ratio sailplane wings today include one spar, ribs, and a thin skin. Most of the high performance european sailplanes have wings constructed of a spar and thick skin with no ribs as shown in Figure 3. The skin has a composite sandwich construction with a balsa core laminated between layers of resin and glass cloth. The spar is usually constructed from epoxy resins and glass fabric. In such a construction the skin and spar resists most of the normal forces and the shear webs in the spar resists some of the shear forces. The shear webs are kept thin as in an I-beam to make efficient use of weight. The idea of using a different type of shear web such as foam is possible so long as the same shear stress requirements is satisfied. Buckling of the skin is resisted by the thick laminated skin. The use of a foam core would also resist skin wrinkling. Therefore the use of low density foam core instead of a thin higher density shear web is completely within reason. And such is the objective of this study, to structually evaluate a high aspect ratio wing constructed of foam core and resin-glass skin.

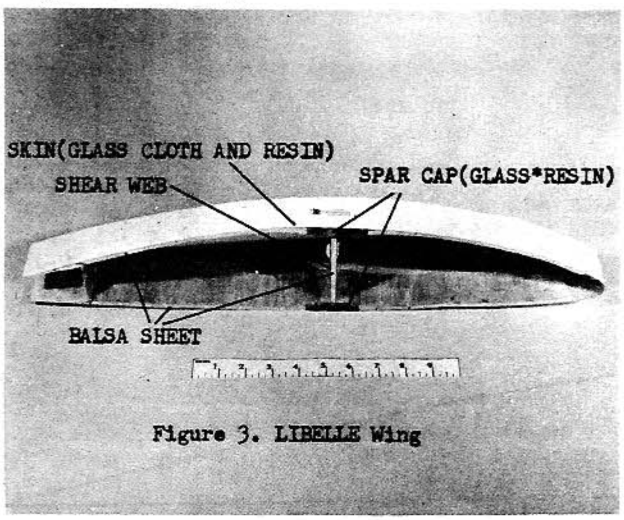


Figure 3. LIBELLE Wing

3. Skin-core Approximations

The flexural rigidity (bending modulus) of a composite beam shown in Figure 2 is give below.[2]

$$(EI)_{composite} = \frac{E_s\, b}{12(1-\mu^2)}\left[d^3 - t_c^3(1-E_c/E_s)\right] \quad (1)$$

For beams in bending the deflection equation shows that the deflections will depend on the combined rigidity of the skin and core materials.

$$\frac{d^2z}{dy^2} = \frac{M}{(EI)_{composite}} \quad (2)$$

If $E_c \ll E_s$ then the flexural rigidity is seen to depend only on the skin modulus of elasticity.

$$(EI)_{composite} = \frac{E_s\, b}{12(1-\mu^2)}\left[d^3 - t_c^3\right] \quad (3)$$

The percent of deflection due to shear was assumed to be small and therefore neglected.

Unfortunately the core cannot be totally ignored. Although the core does not resist bending it does resist a part of the maximum shear stress and resists buckling of the thin skin.

4. Maximum Shear Stress

If the airfoil shape is simplified as a rectangular composite beam, as shown in Figure 4, then the equation for calculating maximum shear stress is

$$\tau_{max} = \frac{V_{max}\left[\frac{2(t_c+d)bt + t_c^2(2t+w_cE_c/E_s)}{8bt+4(2t+w_cE_c/E_s)}\right]\left[bt+(2t+w_c\frac{E_c}{E_s})\frac{t_c}{2}\right]}{\left[b(d^3-t_c^3)+t_c^3(2t+w_cE_c/E_s)\right]\left[2t+w_cE_c/E_s\right]/12} \quad (4)$$

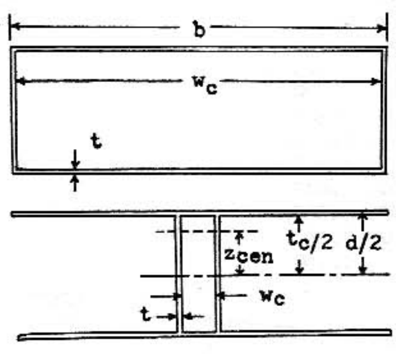


Figure 4. Box Beam Approximation

The maximum shear stress is distributed over the core and skin materials.

$$\tau_{max} = \tau_{s_{max}} + \tau_{c_{max}} \qquad (5)$$

Using the method of equivalent areas a relation for the shear forces in the core and skin was formulated for the airfoil shape at midspan, Figure 1.

$$\tau_{s_{max}} : \tau_{c_{max}} \qquad (6)$$

By approximationg the dimensions of the airfoil at midspan as a rectangular composite beam the equations (5) and (6) were solved simultaneously for core and skin stresses at maximum shear load of 10 lbs. (7G's).

$$\tau_{c_{max}} = 3.33 \text{ lb/in}^2 , \quad \tau_{s_{max}} = 177 \text{ lb/in}^2$$

$$t_{shear} = 0.025 \text{ in}$$

5. Maximum Normal Stress

An approximate maximum normal stress at a 7G wing loading is calculated by using the elastic flexural formula. The root airfoil is approximated as a rectangular crossection so that the moment of inertia can be found as a function of skin thickness.

$$\sigma_{s_{ultimate}} = 6240 \text{ psi}, \; t_{normal} = 0.0091 \text{ in}$$

6. Skin Wrinkling

A method for determining skin thickness required to prevent wrinkling of skin as a function of core and skin properties was outlined by Knight.[1]

The compressive stress in the facing material at which wrinkling will occur is given by

$$\sigma_w = B_1 \, E_s^{1/3} \, E_c^{2/3} \qquad (7)$$

The constant B_1 is plotted as a function ρ where ρ is given by

$$\rho = \frac{t_s}{t_c} \, (E_s/E_c)^{1/3} \qquad (8)$$

For values of $\rho \leq 0.25$, B_1 is a constant, B_1 equals 0.575. Substituting the skin and core modulus and B_1 into equation (7) gives a wrinkling stress.

$$\sigma_w = 5475 \text{ psi}$$

Considering this as a maximum normal stress. The skin thickness can again be determined as was done previously.

$$t_w = 0.011 \text{ in}$$

7. Thin Walled-Unsymmetrical Crossection-Tapered Cylinder

The fact that the core properties could be ignored in the composite beams deflections motivated the stress analysis of a thin walled-unsymmetrical crossection-tapered cylinder.

A computer program was written to take any defined shaped thin-walled cylinder (wing shaped) and give deflections and twists along the span for any defined wing loading.[5]

The program was tested for accuracy by defining a cylinderical shape whose deflections, moments of inertia, shear center location, weight, etc. were compared with hand calculations. The program was found to be very accurate for the thin skin approximation.

8. Approximate Deflections

The wing was approximated as a rectangular box with a uniform distributed load simulating different G loads. Skin thicknesses sufficient for resisting $\sigma_{s_{ultimate}}$ and buckling produced intolerably large deflections. Therefore the skin thickness was calculated to give reasonable deflections for a 7G simulated load and resists the maximum shear stress.

$$t_s = 0.025 \text{ in}$$

9. Wing Fabrication

With regard to money and time the wing was not constructed with the best material or fabrication procedures. Instead relatively inexpensive materials and simplified fabrication methods were used to construct a testable wing. The skin thickness was not closely controlled so long as the thickness was known and related deflections measurable.

The foam core was constructed of CPR 9005-2 rigid urethane foam. Originally it was hoped that the foam core shape could be cut by a hot wire quided over root and tip airfoil templates. Unfortunately it was found difficult to cut urethane foam sections longer then a few feet. The urethane core was therefore shaped by sanding spanwise with the root and tip airfoil templates as quides.

The skin was fabricated by wet lay up of 6 ounce glass cloth and polyester resin. The surface was sanded once and a finish coat applied.

A one inch wide hardwood root airfoil shape was glued to the foam and was also covered by resin and glass. This served as a noncrushable rigid support for the condition of cantilever beam as shown in Figure 1.

10. Test Procedure

The wing was mounted in a fixed condition at the root with the flat side of the airfoil facing upward. The flat side of the wing was leveled so that an evenly distributed load could be applied on the flat surface in a predictable manner. Time and presence of creep permitted only a 1G and 2G load for measuring deflections, Table I.

Wing G loading was simulated by a uniform distributed load using a string of metal slugs attached to a line drawn from the root aerodynamic center to the tip aerodynamic center.

The wings airfoil shape at the root and tip were measured. The skin thickness along the span was measured. A piece of the skin was removed from the wing and tensile tested for a modulus of elasticity. The above was substituted into the computer program to obtain the calculated deflections which are compared with the experimental deflections in Table I.

11. Results and Discussion

The deflections were calculated assuming the core was insignificant in resisting bending. The modulus of the core and skin were experimentally measured and the term

$$(1-E_c/E_s) \qquad (9)$$

in the composite flexural rigidity equation was found to be so nearly equal to one that the core had little or no effect on the composite beams deflections. The experimental deflections varified that this was a good approximation, Table I. The percent error in deflections are a combination of, the core materials contribution to resisting bending, accuracy of deflection measurements, creep, and numerical methods used to calculate deflections. The core's contribution in resisting deflection, although shown to be small, account for the smallest of the possible error sources. Creep is the largest contributor, since the percent errors are seen to be higher for the larger G loads. The error in measuring the deflections with a 1 inch travel 0.001 inch dial gage was less then 1%. The error in the numerical methods was found to be less then 1%.

Table II Creep

1G Uniform lift distribution 0.027 lb/in

Time, minutes	0	10	30
Tip Deflections inches	0.550	0.615	0.640

The presence of creep was noted, Table II. For the wing being tested, little should be said about the creep since the fabrication methods and materials were not the optimum. But creep can and should be accounted for in reinforced plastics who's materials and fabrication are more closely controlled. A method for determining time, temperature, and rupture stresses in reinforced plastics shows that creep in reinforced plastics can withstand large stresses for long periods of time at room temperatures.[3] For example from a series of tests Plaskon 920 (a polyester resin-glass laminate) can resist a stress of 28,500 psi for more then five years before rupture. It was found that these long time stress to rupture estimates could be accurately predicted by a relation derived by Larson and Miller.

$$T(20 + \log t) = \text{constant} \qquad T = \text{Temperature},\ t = \text{Time} \qquad (10)$$

So that long-time low temperature creep results could be calculated from data of short-time high temperature creep tests.

Table I

Calculated And Experimental Results

Total wing weight lb	Calculated	Experiment	Error %
	0.966	0.975	0.93

Deflections for 1G Lift

Uniform lift distribution 0.027 lb/in

Span location in inches from root	0	20	30	40	50
Experiment	0	0.100	0.209	0.333	0.486
Calculated	0	0.106	0.224	0.367	0.521
Error %	0	5.7	6.7	9.2	6.7

Deflections for 2G Lift

Uniform lift distribution 0.054 lb/in

Span location in inches from root	0	20	30	40	50
Experiment	0	0.200	0.403	0.635	0.931
Calculated	0	0.212	0.447	0.734	1.042
Error %	0	6.0	11.0	11.9	10.6

Deflections measured by 0.001 inch increment 1 inch travel dial gage

The shear stress distribution used to calculate the shear center location for a thin-walled unsymmetrical tapered cylinder is questionably used for the composite wing since it has already been shown that the core material resists a substantial portion of the total shear load.

The normal skin stresses calculated by the computer program accurately predicts the real stress experienced by the composite wing since it has already been shown that the skin resist almost all the bending in the beam.

The foam core as was previous shown resist only the maximum core shear stress. Unfortunately the lowest density foam core has an ultimate shear stress much greater than the maximum core shear stress. A lower density foam could be used so that a lighter wing will result without lowering the ultimate shear stress past the maximum core shear stress. As of now the wing in comparsion with an identical wing made of wood is about twice as heavy.

It can be shown that the modulus of elasticity for glass-fiber-reinforced polyester resins vary a small amount with respect to the orientation of the glass cloth weave.[4]

12. Conclusions

The computer program used to calculated deflections and normal stresses for the thin skin only, are the same deflections and skin stresses existing for the composite (skin and foam) so long as the term

$$(1-E_c/E_s)$$

in the flexural rigidity of a composite is approximately equal to one.

Creep in glass reinforced plastics at room temperatures can withstand large stresses for long periods of time before rupture.

Skin thicknesses adequate to resist the maximum normal stresses and buckling produced inadequately large deflections. The skin thickness depends only on the desired magnitude of deflections and the maximum skin shear stress.

The foam core resist only the maximum core shear stress and therefore the lowest density necessary to safely resist this shear stress should be used.

13. Recomendations

It is now possible with the established methodology to optimize such a wing with respect to weight and strength. Time did not permit optimization of the composite wing and the lack of information on the construction of present sailplane wings did not justify a comparision. The author feels that both points should be covered before any large wing fabrication is considered.

At best this project demonstrates that such a wing is possible to analize, build, and could possibly be competative with the present high aspect ratio sailplane wings.

As already pointed out the shear center location, although accurately calculated for the thin skin only, was not proven to be accurate for the composite case. Specially shaped composite beams can be constructed so that the emperical shear center location for the composite beam could be compared with the location calculated by the thin skin approximation method.

14. References

1. Knight, R.S., Analysis Of A Sandwich Construction Glider Wing Design Using Different Foam Plastic Core Materials, Senior Project, California Polytechnic State University Pomona, 1971, pg 37-39.

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3. Goldfein, S., Time-Temperature and Rupture Stress in Reinforced Plastics, "Modern Plastics", 32(4) 148(Dec 1954).

4. McClintock F.A. and Argon A.S., Mechanical Behavior of Materials, Addison-Wesley, 1966, pg 87.

5. Bruhn E.F., Analysis And Design Of Flight Vehicle Structures, Tri-state Offset Company, 1965, pg A15.2-A15.3.

Table III

Material Properties

Material	Modulus of elasticity psi	Ultimate normal stress psi	Ultimate shear stress psi	Density lb/in^3
CPR 9005-2 Urethane foam	1415	-	20	0.00116
Polyester glass reinforced skin	* 6.6×10^5	6240	-	0.0363

* Skin was tensile tested.

Specimen - Gage length 6 inches
- Width 1 inch
- Glass fabric orientation 30 degrees from laminate fiber axis