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WEIGHT SAVING BY COMPOSITE PRIMARY STRUCTURES

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WEIGHT SAVING BY COMPOSITE PRIMARY STRUCTURES

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The structural mass of all means of transportation evidently appears as a component which has an unfavourable influence on the economy of the whole system, as the structural mass together with payload, fuel, engines etcetera has to be accelerated and retarded, as well, and causes induced drag due to the necessary additional lift, or when moved on the ground, friction losses.

This aspect results in the tendency to develop the lightest possible structures for any given environment conditions under the circumstances of operation.

Aiming for optimum structures of light weight, that means low mass, the following combinations of physical constants of the structural materials considered are of great and general influence:

1.) the quotients σ/ρ of ultimate stress under any possible load and/or failure condition to the density. These quotients have the dimension of energy divided by mass or the square of a velocity. In fact $\sqrt{\sigma/\rho} = C_T$ is the velocity of progress of a transversal wave along a stressed element with low bending stiffness.

2.) the quotients E/ρ^n, G/ρ^n of YOUNG's and shear moduli to the density power n, where n is of interest within the limits $0 \leqq n \leqq 2$.

Analogous to $\sqrt{\sigma/\rho}$ the root $\sqrt{E/\rho} = C_L$ is the velocity of propagation of a longitudinal wave in a material.

Since the so-called "advanced fiber-matrix composite materials" with their particular material constants of their components have become available, great hope arose for the possibility of a significant step in the direction of mass reduction of aircraft and space vehicles primary structures.(1)(2)

By pragmatic decisions, not decisions based on principles, just a few out of the at first large number of fiber matrix combinations, proved to be favourable enough from the engineering standpoint as to be taken - after a careful investigation in the field of research - into engineering application.

Since the end of the sixties, boron/tungsten fibers in epoxi, polyimid, aluminum or titanium-alloy matrix have even reached the stage of application as primary structural elements, in particular as components for high speed military aircraft.
(3) (8) (9) (5) (6) (7) et alteri

The gain which has been achieved has been mass-saving as well as stiffness-improving, whereby the stiffness advances have in some cases been decisive.

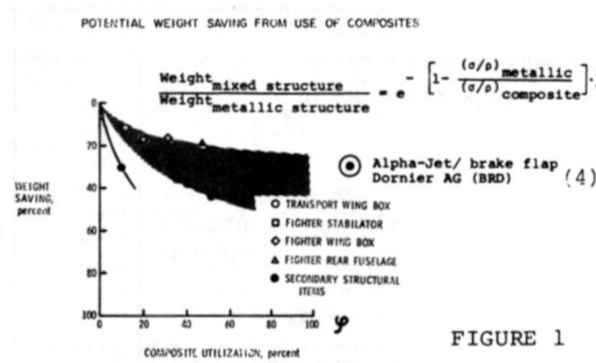
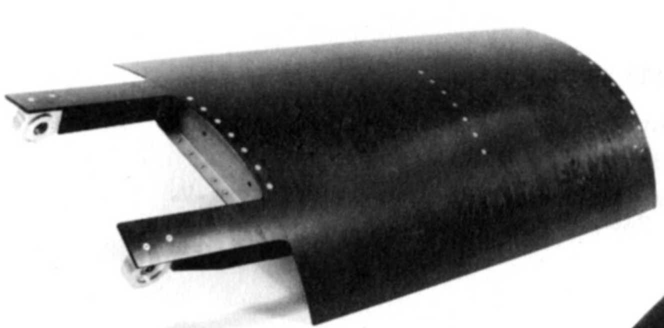


FIGURE 1

POTENTIAL AND ACTUAL WEIGHT SAVING FROM USE OF COMPOSITE MATERIALS.
SYNOPTIC PRESENTATION BY R.A. PRIDE(3)



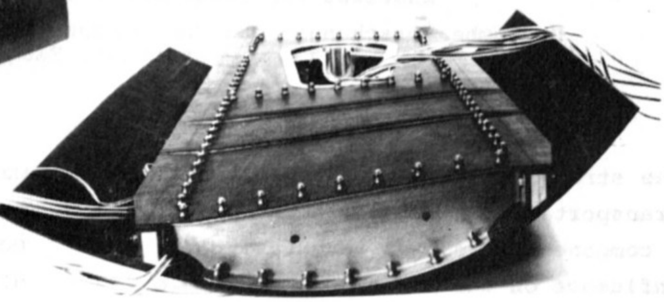


FIGURE 2. EXPERIMENTAL α-JET BRAKE-FLAP. BUILT IN CARBON/EPOXI COMPOUND. CFEP SHELLS, SANDWICH, AND ALU ALLOY SUBSTRUCTURE. DEVELOPED 1971 BY DORNIER A.G. (18) ON A CONTRACT OF THE MINISTRY OF DEFENCE F.R.GERMANY. ZTL PROGRAMME. WEIGHT SAVING AGAINST ALL-ALUMINUM ALLOY STRUCTURE 37,7%.

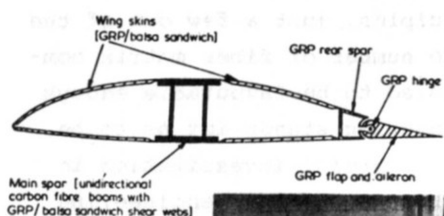


FIGURE 3. THIS WING OF THE 19 m SPAN T59C GLASFLÜGEL-HAENLE/SLINGSBY KESTREL HAS BEEN FLOWN AT THE ENGLISH NATIONAL GLIDING CHAMPIONSHIPS, MAY 1971, COMING SECOND. THE CFEP MAIN SPAR WITH COURTAULD's GRAFIL HT-S CONTINOUS CARBON FIBER IN EPOXI RESIN IS 48% LIGHTER THAN THE EQUIVALENT GLASSFIBER REINFORCED SPARS, RESULTING IN A 8% WEIGHT SAVING OVER THE TOTAL EMPTY WEIGHT OF THE AIRCRAFT. (17) (18)

Influenced not only by the quality of the basic composite elements and the quality of optimisation, but also by the order of magnitude of the fraction of composite material in the structural element, the order of magnitude of structural mass reduction has reached 26 to 39% in relation to the original metallic structure. (See Fig 1,2,3 and Table 1).

If we compare this magnitude of mass reduction with the relation between specific strength (σ/ρ) and specific stiffness (E/ρ) values of titanium- or aluminum-alloy to these of carbon- or boron-fibers alone, the great discrepancy between hope and reality, presumable component possibilities and limitations in hardware are obvious. (10) (11) . (See Fig 4).

	Aircraft type	Structural group	initial metallic	converted into	Weight saving	Remarks	References
Lockheed	C 130	Centerwingbox	Al	Al+BFEP	13 %	-	(3)
Grumman	F 14 A	Horiz.Stabilizer	Ti	BFEP	20	exper.flying	(8) (9) (40)
Lockheed	C 5 A Galaxy	L.E.Slatsegment	Al	BFEP	22	-	(8) (39) (40)
General Dynamics	F 106 A	Entire aircraft	Al	BFAL	23 %	design study	(5)
General Dynamics	F 111	Horiz.Stabilizer	Al	BFEP	25	exper.flying	(8) (40)
Northrop	F 5 A	Landg.gear cover	Al	BFEP	29	-	(8) (40)
Sikorsky	CH 47	Fuselage tail sect.	Al	Al+BFEP	33 %	-	(3) (8) (40)
McDonnel Douglas	F 4 C Phantom	Rudder	Al	BFEP	35	exper.flying	(8) (40) (42)
General Dynamics	F 111	Wingtip box	Al+GFEP	BFEP	35	-	(8) (40)
VFW-Fokker	VFW 614	Spoiler	Al	BFEP	36 %	experimental	(8)
Dornier AG	α-Jet	Airbrake	Al	Al+CFEP	38	experimental	(4)
Bac Weybridge Div.	VC 10	Aileron spar	Al	Al+CFEP	43	-	(8) (40)
Slingsby/Glasflügel	T 59 C Kestrel	Wing spar system	GFEP	CFEP	48 %	flying	(17)

TABLE 1

EXCERPT OF COMPOSITE PRIMARY STRUCTURES REALISATIONS.

Al Aluminum alloy, Ti Titanium alloy, GFEP Glassfiber/Epoxi, CFEP Carbonfiber/Epoxi
BFEP Boronfiber/Epoxi, BFAL Boronfiber Alu matrix.

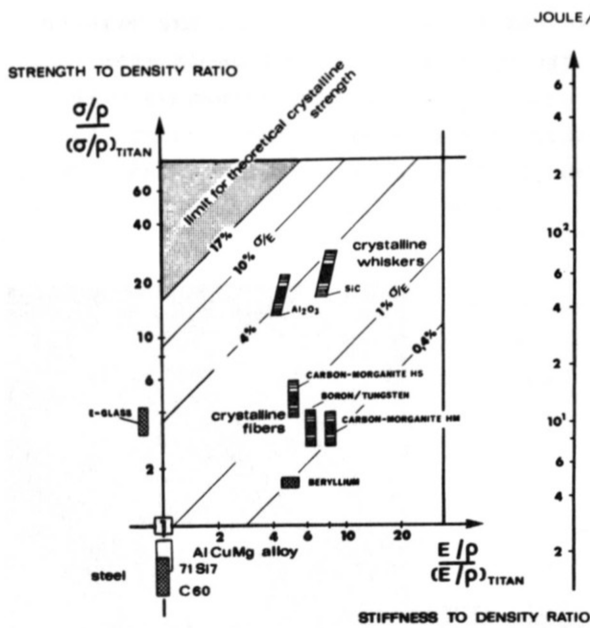


FIGURE 4. ULTIMATE (σ/ρ) COMPOSITE RELATED TO ULT. (σ/ρ) TITANIUM, TiAl6V4 OVER (E/ρ)/(E/ρ) TITANIUM FOR A SELECTED NUMBER OF FIBERS AND WHISKERS.

The apparant discrepancy is considerably reduced if we do not compare the material qualities of fibers as they are, but with those of the optimum orthotropic composite-systems of fiber and matrix, applicable in structures that are: bundles of straight fibers, embedded in the matrix of choice. (12)(13)(14)

The strength- and stiffness to density ratios in the fiber direction of such uni-directional rods is - following a rule of mixture - a function of the fiber volume fraction in the composite.
Under any circumstances, one should aim at a maximum fiber content in the compo-site.

Fiber contents beyond 60% need special devices and procedures, time to squeeze out the surplus of the viscous matrix ma-terial and much care to avoid any damage of the delicate fibers. These difficul-ties increase progresslively when approa-ching the geometric limits of highest-density packing.

Even before these geometric limits are reached, other technological drawbacks are encountered which reveal some very special problems.

The problems appearing here are of the following nature:

1.) The scatter of actual values of strength to density ratios in fiber direction is in the order of magnitude of about $\pm$6% to 12%, even if all production parameters are carefully held constant and if the utmost of care is taken to achieve a product of constant quality. (See Fig 5). (15)(16)

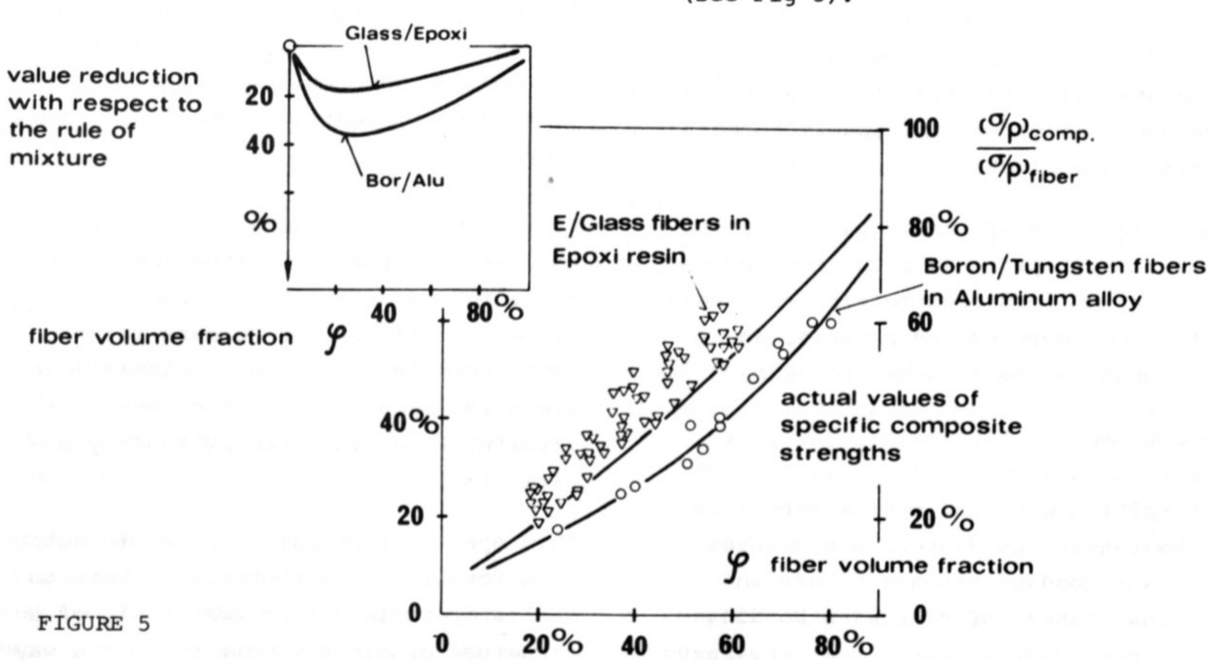


FIGURE 5

It seems that this scatter represents not only a factor remaining from the scatter of values of single fiber tests, but also the irregular fiber distribution in the matrix, the influence of anomalous fiber orientation, fiber wavyness, insertion of matrix voids, matrix impurities, i.a.. Evidently, the influence of irregular fiber distribution and similar effects is most pronounced at small values of fiber volume fraction.

2.) If fibers are embedded in metal matrices, due to the high temperatures in the vicinity of the metal melting point during the injection process, diffusion between fiber and matrix might occur and/or changes in the crystalline structure of the fibers themselves.[14][19]

All this leads to a systematic reduction of the values expected from the rule of mixture. In addition, there remains the unsystematic scatter of the resulting composite strength.

For safety reasons, the lower limits of scatter only can be considered as a reliable design basis. (See Fig 5).

The reduction of the random as well as the systematic deviation is a matter of technological refinement and improvement of production techniques.

Mainly advances of matrix injection and saturation, improvements in fiber orientation by uniform prestressing of fibers during the impregnating process, and similar measures seem to be promising.[16]

The detrimental influences which high temperatures have on fibers in contact with melting metallic matrix materials can be reduced by sintering processes and brace bonding between fibers and matrices instead of diffusion bonding due to the lower temperatures necessary.

More versatile possibilities are offered by the application of extremely thin layers of less reactive materials with higher melting points on the fibers by chemical or vapour deposition, like SiC on boron- or tantalium on carbon fibers.[21]

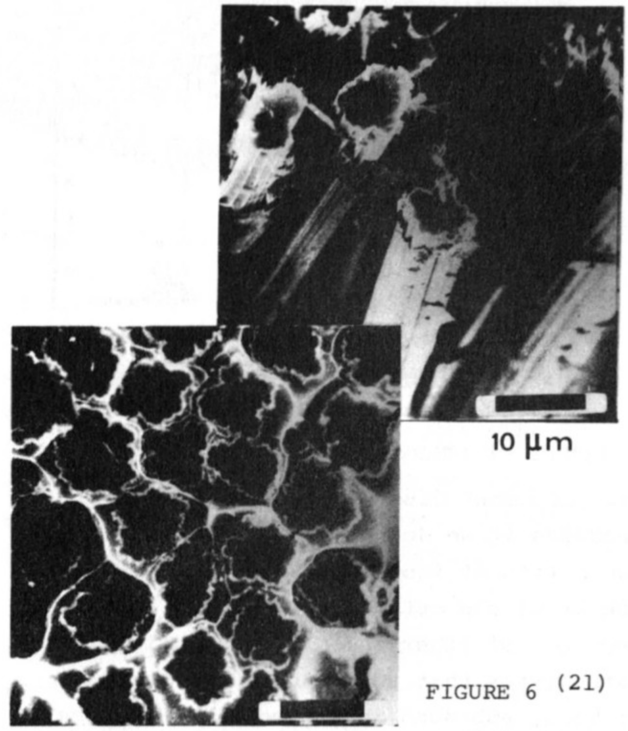


FIGURE 6 [21]

Regardless of how these problems might be solved, they stand at the starting point for any engineering efforts to save mass - that is weight - on composite primary structures. They will be solved by the techniques of continous small but safe steps.

It remains to the hands of designer and analyst to optimally synthesize the elements into an actual structure of the highest possible load to mass relation, from those basic composite elements which are given as an uniform rod, sheet or layer with parallel fibers in only one direction.

This optimum synthesis aims at an adaption of a few of such orthotropic layers to shells or stringers as components of a structure of minimum mass in such a way that given collectives of load can be supported without failure. The collectives of loads may consist of bending and/or torsion moments, transverse forces, internal or external pressures and single local concentrated forces, load amplitudes, frequencies, i.a..[17][22]

The adaption of structural conceded anisotropy does not necessarily imply isotropy but a qualified anisotropy to match the most adverse combination of loads within the load collective.

To put into reality a complete isotropy in all three space axies in composites, which is only possible with very short fibers, implies a loss of some 80% of strength and almost 70% of stiffness related to the optimum orthotropic composite rod.

For an optimum wing box section under bending and torsional moments, with shear flow due to a transverse force and with internal pressure, e.g. due to fuel in the box, it is possible to achieve 70 to 80% of the load capacity of a similar body built from a hypothetic isotropic material with the optimum orthotropic strength of the composite considered. In this case the penalty due to anisotropy of the composite material is just 16% to 38%. (See Fig 21).

The in-plane elastic constants of a multi-layer-composite shell are given by the corresponding elastic constants of each of the layers, the related layer thicknesses and the angles ω which the axes of orthotropy of any layer include with the axis of symmetry of the shell.

All in-plane strainstates of any layer of the shell must be the same to satisfy the quasi internal compatibility conditions of the shell.[23][24]

As with plane or just smoothly curved shells these internal compatibilities can almost only be enforced by interlayer shear, i.a.. This shear proves to be an important strength limiting value for the matrix, especially because the matrix strength in general is more than one order of magnitude smaller than the fiber strength.

This at least holds as far as the displacements due to the in-plane POISSON's ratios are concerned, if the axial load on a wing or fuselage shell is dominant.

On the other hand, the interlayer shear as consequence of a shear loading of a composite shell is mostly not decisive as far as interlayer failure is concerned.
The same is true concerning the interlayer shear due to axial loads in such a shell. These loads vary strongly with the axial coordinate, attaining extremely small values at the ends of the structure.

Conversely, the problem becomes critical at joints of sub-structures or of structural elements where edge loads are to be transmitted.

In shell structures containing closed cross sections, the tangential compatibility conditions are satisfied not mainly by interlaminar shear but by the closed loop of all layers together.
We call this phenomenon "Formschluss" because the structural cross section contour decisively affects the satisfaction of layer compatibility by transfer of pressures perpendicular to the surface of the shell.[22]

The resulting symmetry of rotation yields equal stress distribution around the whole circumference especially if axial, tangential and/or torsional loads are given.

This certainly is the reason why thin-walled tube specimen with circular cross section proved to achieve the highest values of ultimate stresses at a given layer configuration and collective of loads

Yet it has been possible to prove that even for structures which for producibility reasons can only be assembled from flat panels, the penalties encountered can be kept within a reasonable limit. This is especially so if the ratio of width to wall thickness of such panels is greater than ≈ 200.

Not directly influenced by the means which effect the strain compatibility between the orthotropic layers of different fiber orientation in a shell are the limitations caused by matrix failure in between the fibers of the layers due to shear distortion. Even if only symmetric axial or tangential loads are applied, this distortion could go beyond failure critical limits, especially with "unsupported" panels with only two directions of the axes of orthotropy in the component layers of a shell.

For application to aircraft structures, symmetric panels are of particular interest, where the angles ω which the two different groups of layers include with the axis of load and/or the axis of symmetry are:

$$\omega_{\text{(odd layer numbers)}} = -\omega_{\text{(even layer numbers)}}$$

In such shells, the in-plane POISSON ratio can reach values beyond any experience with isotropic materials. (See Fig 7). (25)(26)

This at first sight astonishing result is explained by the inhomogeneity of the material. If YOUNG's modulus of the fibers considerably exceeds that of the matrix material, the in-plane transverse contraction is in the limit determined largely by a kinematical mechanism formed by the crossed-over relatively stiff fibers.
The strain and thus stress state in the matrix is to a large degree prescribed by the kinematics of the above-mentioned mechanism. Yet the actual transverse direct strain in the matrix itself thus may be

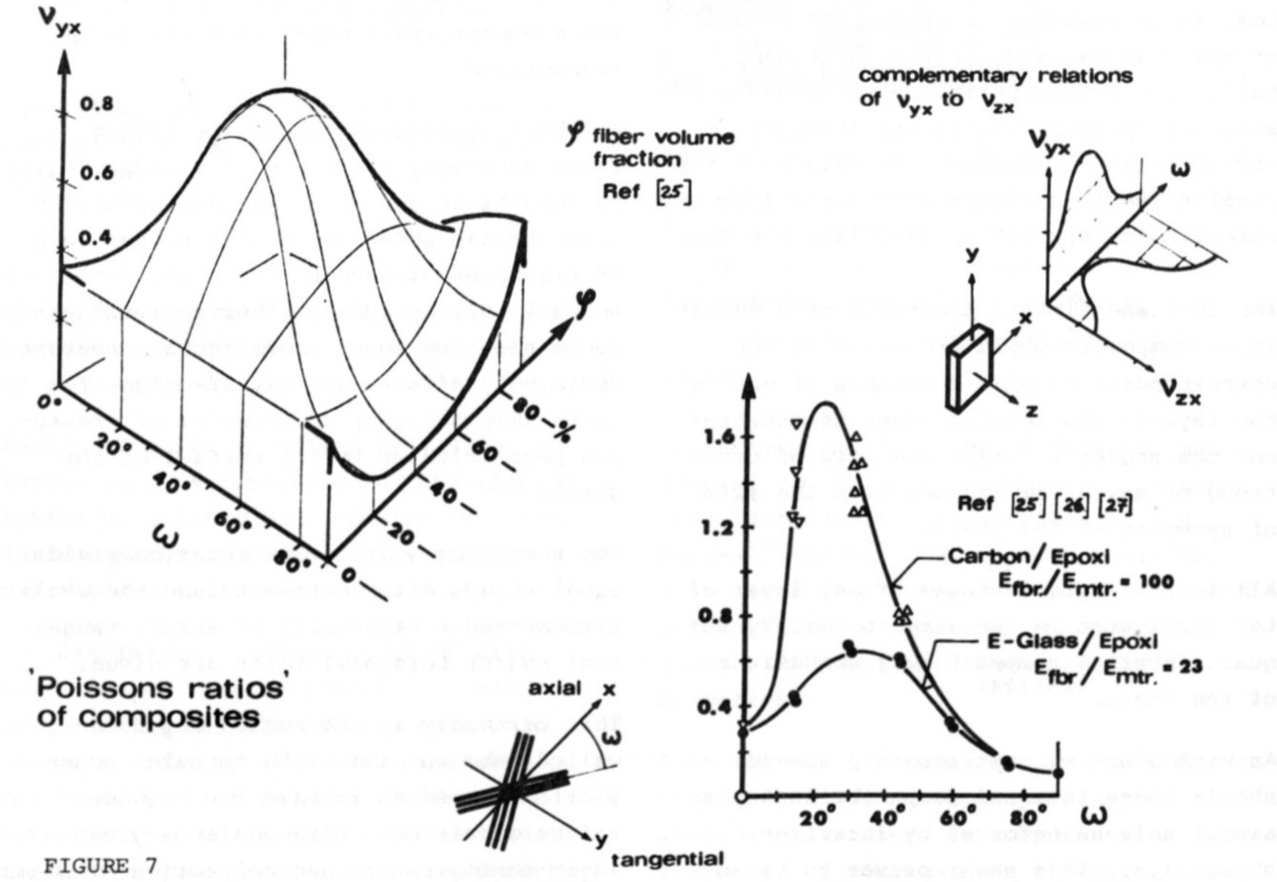


FIGURE 7

less or greater than zero for prescribed overall in-plane extension in the direction of an axis of symmetry.

Such behaviour applies e.g. to carbon fibers in epoxi matrix, where YOUNG's moduli of the fibers and the matrix material differ by a factor of over 100. [27]

As large POISSON's ratios indicate large shear distortion in between the fibers of symmetric layer systems it is to be expected that additional supporting layers should improve the strength of panels of the symmetric two-groups-of-layers type. (See Fig 8, 9, 11, 15 and 17). [2] [22] [37]

The optimum volume ratio of the supporting layers in a shell with a given angle $\pm\omega$ of the mainly load bearing layers has been calculated and proved by experiments to be

$$\sum s_{\text{supporting layers}} / \sum s_{\text{all layers}}$$

$$= 1/(1+\chi_{cpr/ext} / \sin^2\omega \cdot \cos\omega),$$

where

$$\chi_{cpr/ext} = \frac{\sigma_{\text{ultimate compression}}}{\sigma_{\text{ultimate extension}}}$$

This optimum volume fraction is approximately the same for flat as for tubular specimen, but the gain in strength is greater with flat specimen than for thin walled tubes. (See Fig 8).

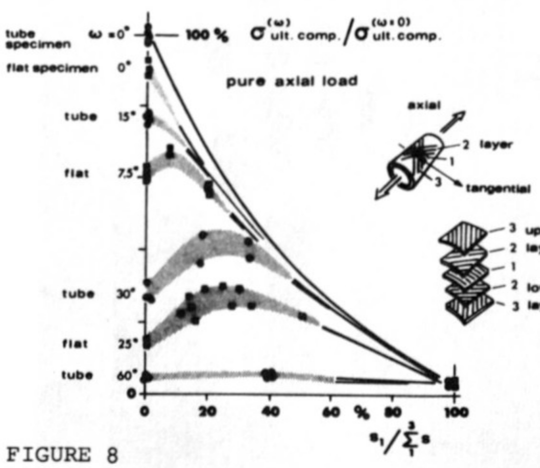


FIGURE 8

To investigate the influence of panel width to wall thickness ratio, the necessary analysis has been found to be already fully developed by civil engineers occupied with prestressed concrete structures, thus just an adaption of existing theory has been necessary. (See Fig 9). [29]

Experiments proved that with panels of high width to wall thickness ratio, the shear stresses reach values beyond negligible magnitudes only in the vicinity of the edges. Thus the failure of supported flat panels starts from those edges causing delamination of supporting and mainly bearing layers.

$$\mu = (\nu_{yx_2} - \nu_{yx_1})\, 2G_v/\delta_v s_2$$

$$\lambda^2 = \frac{2G_v/E_{y_1}}{s_1 \delta_v}\left[(1-\nu_{yx_1}\nu_{xy_1}) - \frac{E_{y_1}}{E_{y_2}}\frac{s_1}{s_2}(1-\nu_{yx_2}\nu_{xy_2})\right]$$

$$\tau_{y_{1/2}} = \bar{\varepsilon}_x \frac{s_2}{2}\frac{\mu}{\lambda}\left[1 - e^{-K\lambda(\frac{1}{2}t_y - y)}\right]\cdot sh\,\lambda y / ch\,\lambda \tfrac{t_y}{2}$$

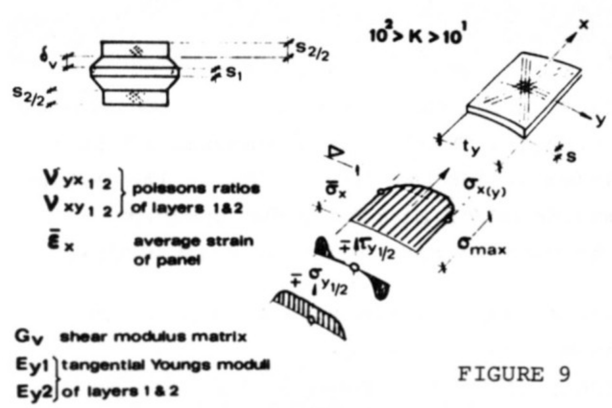


FIGURE 9

$$\sigma_{y_2} = \bar{\varepsilon}_x \frac{\mu}{\lambda^2}\left[1 - ch(\lambda y)/ch(\lambda \tfrac{t_y}{2})\right]$$

$$\sigma_{x_1}(y) = \left[\varepsilon_x - \nu_{xy_1}\cdot\sigma_{y_2}\frac{s_2}{s_1}\right] E_{x_1}/(1-\nu_{xy_1}\nu_{yx_1})$$

The reduction of average strength $\bar{\sigma}_x$ due to unequal longitudinal stress distribution $\bar{\sigma}_x < \sigma_{xmax}$ for large panel width is in general too small to explain the observed discrepancies between filament-wound tubes and flat specimens of the same panel structure. (See fig 8, 10 and 11).

On the other hand, tests with sets of flat panel specimens with same shell dimensions and layer programme show not only clearly the characteristic influence of width to supporting wall thickness or "slenderness" ratio which appears in the argument of the hyperbolic sinus (figure 9), but also the fact that already at slenderness ratios >200 the panel strength reaches its asymptotic maximum.

Thus the panel/tube strength discrepancy very probably appears as a precocial delamination failure, beginning at the rim of a panel. This hypothesis is not only supported by the direct observation of failing flat panels, but also by the observation that panels carefully clamped at the edges by a number of independent small clamps reach some 15% higher failure limits than the same panels not clamped. (See Fig 12 and 13).

This clamping at the edges improves the panel strength almost to the tube specimen strength. This could be understood as a valuable hint of how to design the joints at the edges of panels for optimum strength to mass ratio of a structure.

The supporting elements of a panel not necessarily have to consist of a special group of layers integrated in the shell, but can as well be designed as discrete elements with the same cross section area as the total supporting layers.

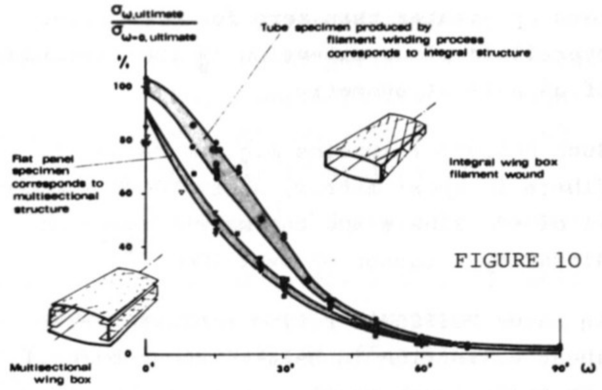
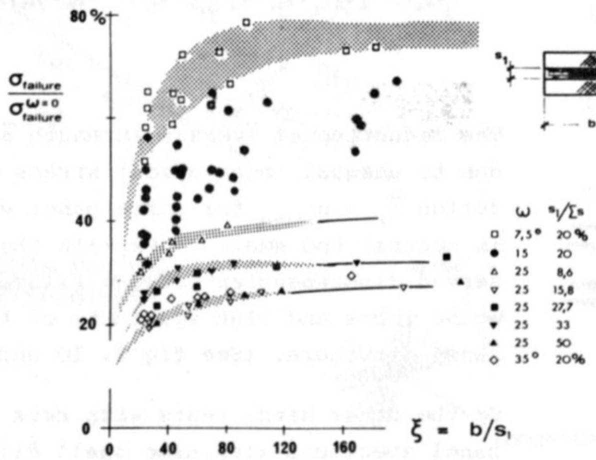


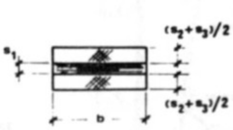
FIGURE 10

Strength-tests with tube specimen with discrete elements showed no systematic dependency between supporting element distance related e.g. to the tube diameter, and the strength of the tube under axial extension. (See Fig 10 and 14).

All this indicates a certain flexibility of freedom for an adequate design of composite structures in a similar manner as it has been so far with classic metallic structures. This holds especially if position of frames or ribs and the distribution of buckling stiffeners are concerned.

Not taken into consideration at present is the fact that structural materials which relative to others have an absolutely lower density offer a special advantage as far as buckling of panels is concerned because the buckling strength to mass ratio of any compressed element of given dimension goes with E/ρ^2, whereas the general stiffness to mass goes just with E/ρ.(7) (30)





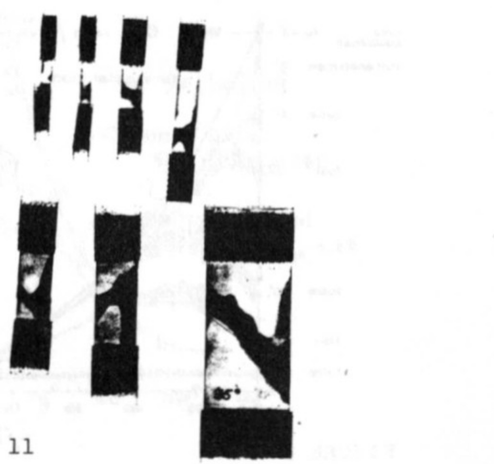


FIGURE 11

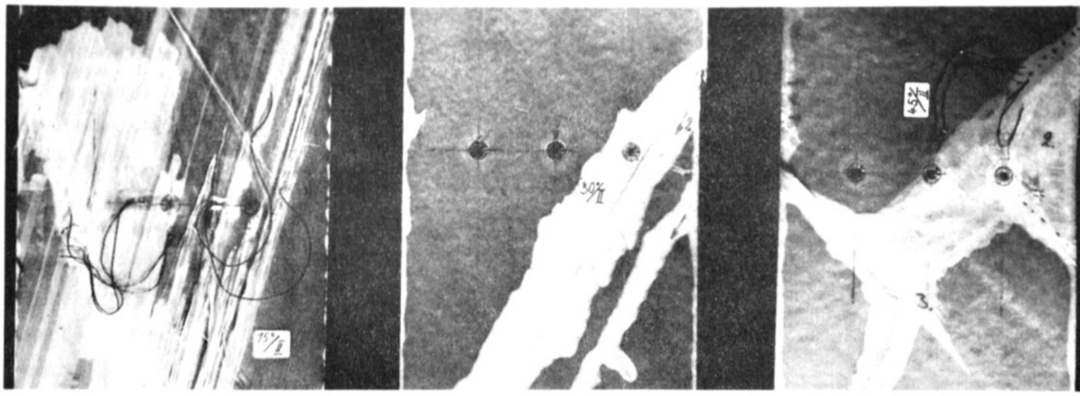


FIGURE 12

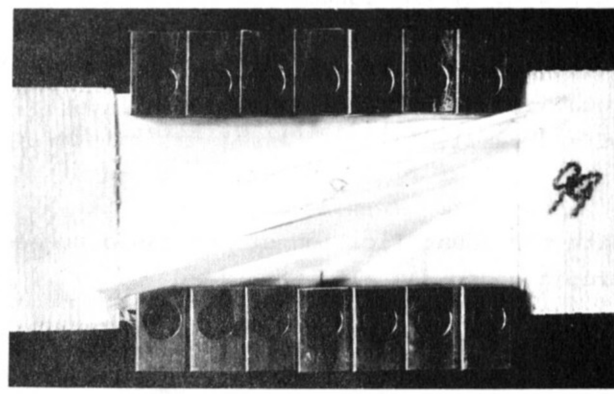
To make the best use of composite shells as basic components of light-weight structures, optimum adaption to the actualy given loading programme is peremptory.

This adaption is enabled by simply choosing the relative layer thicknesses e.g. of longitudinal and transverse individual layer groups of a shell, as well as those orientated at an angle $\pm\omega$. The angle ω here appears as the most important parameter.

Particularly in structural bodies with high aspect ratio like wings and/or fuselages of transport aircrafts, the highest stressed parts are submitted to loads with high axial and low tangential components.

Thus angles ω of $20^{\circ} < \omega < 30^{\circ}$ and layer thickness ratios: $(s_{tang}/\sum s) = 0{,}35$

FIGURE 13



prove to be adequate. In supersonic aircraft structures with $AR \leqq 3$ and a more even pressure distribution across the wing cord, the tangential load components could be relatively larger, and in that case angles $\omega > 30^{\circ}$ and higher tangential layer thickness ratios might be necessary.

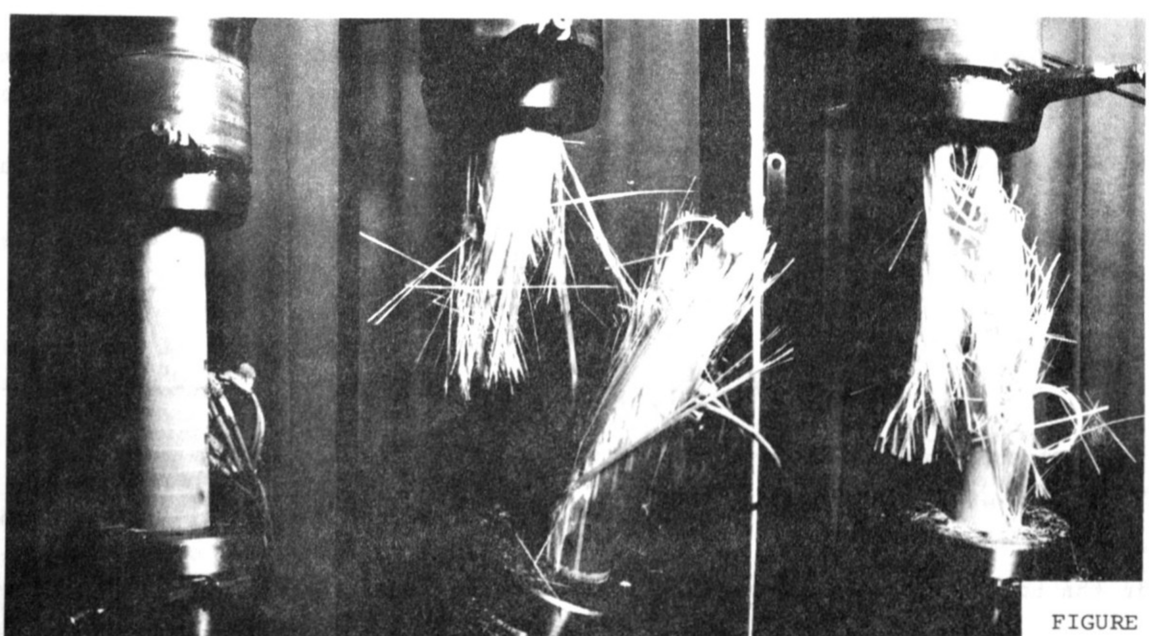


FIGURE 14

Panels of the kind mentioned so far with three groups of layers - a symmetric pair as main longitudinal bearing system and a group of transverse layers as a supporting layer (which in addition can bear the transverse loads) - have proven to be the most versatile as far as the load adaption is concerned.

The in-plane elastic behaviour of such panels can easily be described by linear relations if the assumption holds that the satisfaction of compatibility conditions causes only negligible differences in the total in-plane distortions of all layers of the panel.(23)(24)(31) et alteri.

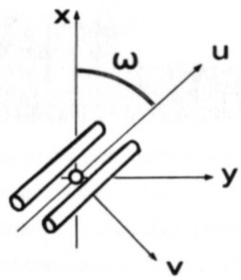
With the identification of the basic orthotropic composite element:

$$i = \{1, 2 \cdots i \cdots n\}$$

$$j = \{uu, vv, uv\}$$

and

$$k = \{xx, yy, xy\}$$



and the definitions of the coefficients for the matrices C_i and T_σ:

$$c_{11} = E_{uu}/(1-\nu_{vu}^2 E_{vv}/E_{uu})$$

$$c_{22} = c_{11} E_{vv}/E_{uu}$$

$$c_{33} = E_{uv}$$

$$c_{21} = c_{12} = c_{22}\,\nu_{vu}$$

$$[T_\sigma]_i = \begin{bmatrix} +\cos^2\omega & +\sin^2\omega & +2\sin\omega\cdot\cos\omega \\ +\sin^2\omega & +\cos^2\omega & -2\sin\omega\cdot\cos\omega \\ -\sin\omega\cdot\cos\omega & +\sin\omega\cdot\cos\omega & +(\cos^2\omega-\sin^2\omega) \end{bmatrix}_i$$

further by a reasonable assumption that all stresses $\sigma_{k,i}$ of all layers summarized to σ_k (23)

$$\sigma_k = {}_i\sum_1^n s_i \sigma_{k,i} / s^*$$

where $s^* = {}_i\sum_1^n s_i$

and the formulation of the compatibility condition

$$\varepsilon_{k,i} = \varepsilon_k$$

we gain the total stiffness matrix D

$$\boxed{C_i = \begin{bmatrix} c_{11} & c_{12} & 0 \\ c_{21} & c_{22} & 0 \\ 0 & 0 & c_{33} \end{bmatrix}}$$

$$\boxed{\sigma_{j,i} = C_i \cdot \varepsilon_{j,i}}$$

$$\overline{\underline{\varepsilon_k}} \qquad \boxed{\begin{aligned} \sigma_{k,i} &= T_{\sigma,i} \cdot \sigma_{j,i} \\ \varepsilon_{k,i} &= T_{\varepsilon,i} \cdot \varepsilon_{j,i} \end{aligned}}$$

$$\boxed{T_{\varepsilon,i}^{-1} = T_{\sigma,i}^{t}}$$

$$\boxed{D = {}_{(i)}\sum_1^n \frac{s_i}{s^*} \cdot T_{\sigma,i} \cdot C_i \cdot T_{\sigma,i}^{t}}$$

and the actual stresses $\sigma_{j,i}$ of each individual layer in the compound of the shell.

$$\boxed{\varepsilon_k = D^{-1} \cdot \sigma_k}$$

$$\boxed{\sigma_{j,i} = C_i \cdot T_{\sigma,i}^{t} \cdot D^{-1} \cdot \sigma_k}$$

As usual, considerable difficulties arise if nonlinearities have to be taken account of.

The identification and listing of the many possible phenomena which initiate and/or cause failure of composite panels is a task long begun and permanently continued. There exist types of initial interface micro failures which are to a certain extent stable as far as failure progress is concerned and therefore non-desastrous even after a great number of load cycles - but unavoidable even at low rates of load. (19)

In general, however, neither the total failure behaviour at short time single load cycle, nor the long time or multi cycle load are basically different compared with that of classic metallic materials.(19)(20)

By sophisticated analysis and somewhat tricky, expensive tests, the shell failure limits can be ascertained within the scope of the chosen shells parameters. (See Fig 15 and 16). (24) (32)

As stated already in the mid sixties the failure limits can not be described by a single equation (Voigt[33], Hill[34] et alteri) representing the "failure surface" as an ellipsoid, but needs systems of quadratic equations interconnected by interaction factors γ_i as Puppo and Evensen[35] have proposed, of the type:

$$\boxed{\{\sigma_i\}_K^t \, [R_i^p]_K \{\sigma_i\}_K = N_i}$$

with

$$K = \{xx, yy, xy\}$$
$$p = a, b$$
$$i = \{1, 2, 3 \ldots\ldots n\}$$

and:

$$[R_i^a]_K = \begin{bmatrix} +1/T_{x_i}^2 & -\gamma_i/2T_{y_i}^2 & 0 \\ & +\gamma_i/T_{y_i}^2 & 0 \\ \text{symm.} & & +1/T_{xy_i}^2 \end{bmatrix}$$

where

γ_i is the interaction factor for layer i.

$$[R_i^b]_K = \begin{bmatrix} +\gamma_i/T_{x_i}^2 & -\gamma_i/2T_{y_i}^2 & 0 \\ & +1/T_{y_i}^2 & 0 \\ \text{symm.} & & +1/T_{xy_i}^2 \end{bmatrix}$$

Such systems of equations describe the failure limits by the surface of a hypothetic body consisting of a multitude of ellipsoidic surfaces intersecting each other by curved edgelines. Thus this cornered "failure body" represents the multitude of individual failure modes of each individual shell component as a single and in cooperation.[22]

The Puppo/Evensen criterion not only fits excellently with experimental results but in addition permits to draw from the tests some useful interpretations, e.g. of the degree of interaction between individual layers. (See Fig 16).

The above-mentioned adaption of composite shell parameters to a given state of load or collective of loads is part of an optimisation process aiming at the minimum mass structure for given load collective.
Tools helpful for such procedure are diagrams like those presented in figures 10, 15, 16 and 17. Figure 17 mainly gives an intuitive synopsis of the influence of the

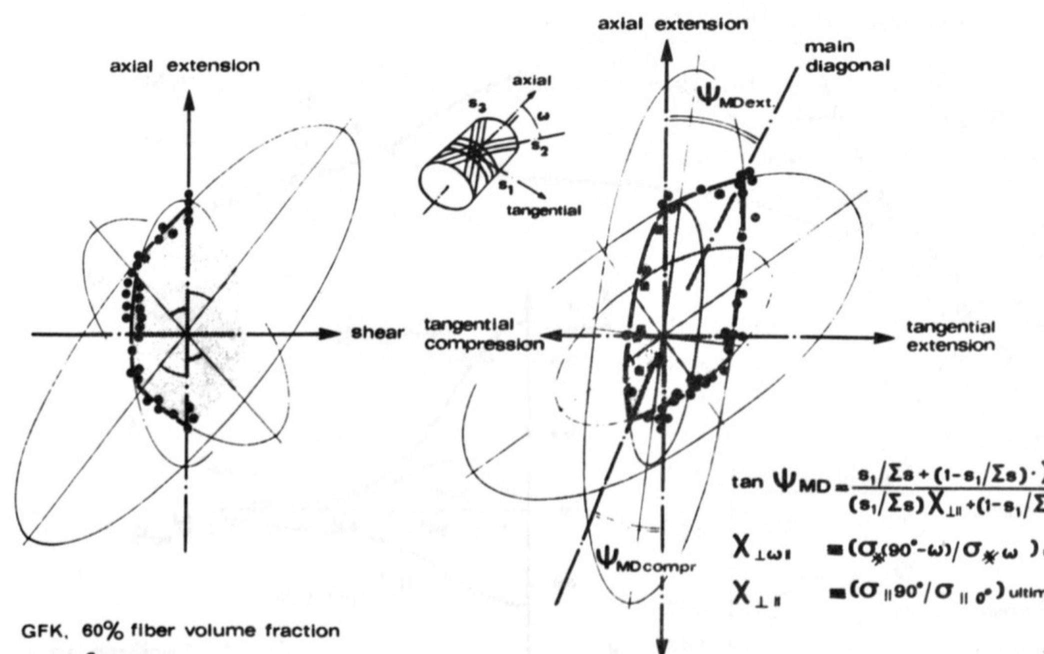


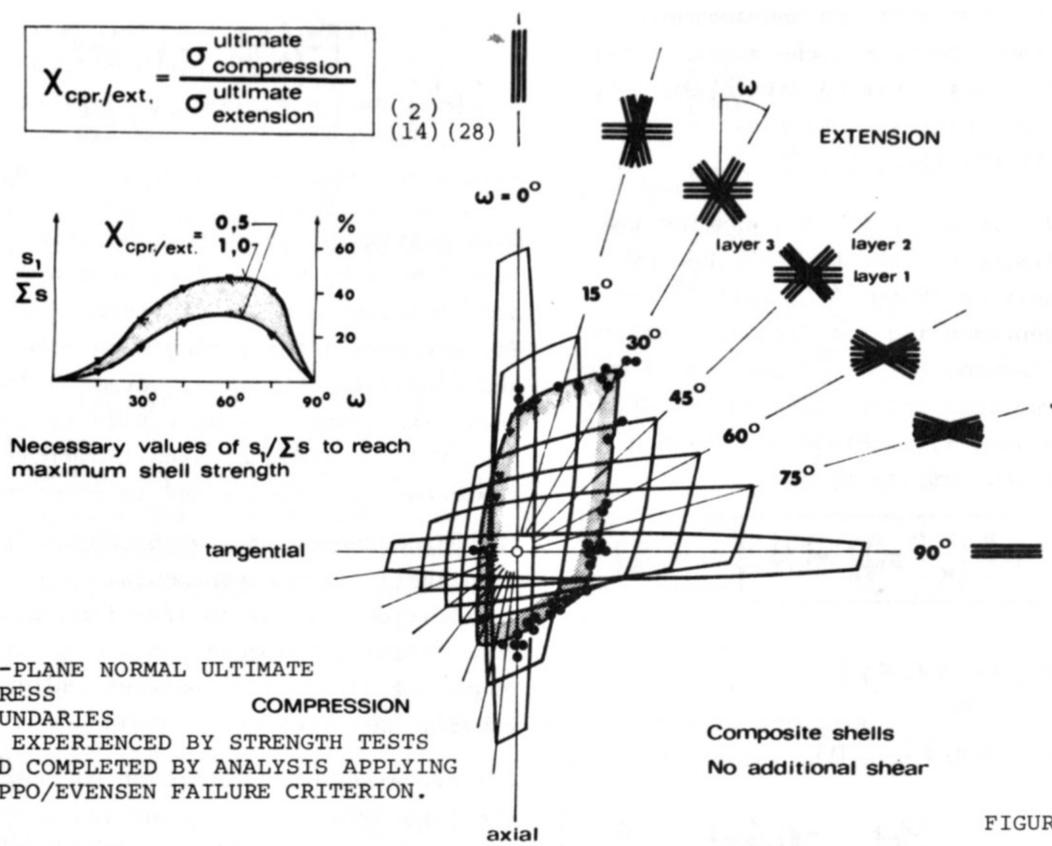
FIGURE 15

$$\tan \Psi_{MD} = \frac{s_1/\Sigma s + (1-s_1/\Sigma s)\cdot X_{\perp\omega\parallel}}{(s_1/\Sigma s)X_{\perp\parallel} + (1-s_1/\Sigma s)\cos\omega}$$

$X_{\perp\omega\parallel} = (\sigma_{*}(90°-\omega)/\sigma_{*}\,\omega)_{\text{ultimate}}$ different for extension and compression

$X_{\perp\parallel} = (\sigma_{\parallel}90°/\sigma_{\parallel}0°)_{\text{ultimate}}$

GFK, 60% fiber volume fraction
$\omega = 30°$
$s_1/\Sigma s = 12\%$



IN-PLANE NORMAL ULTIMATE STRESS BOUNDARIES AS EXPERIENCED BY STRENGTH TESTS AND COMPLETED BY ANALYSIS APPLYING PUPPO/EVENSEN FAILURE CRITERION.

FIGURE 16

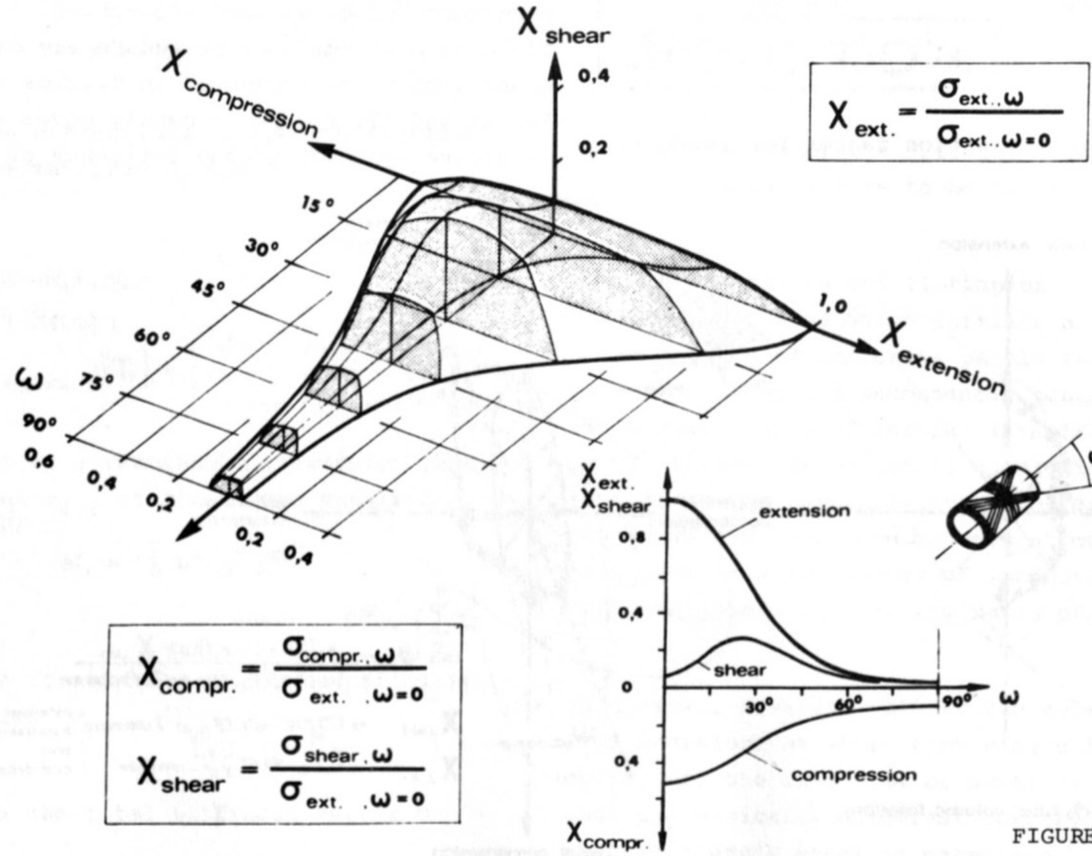


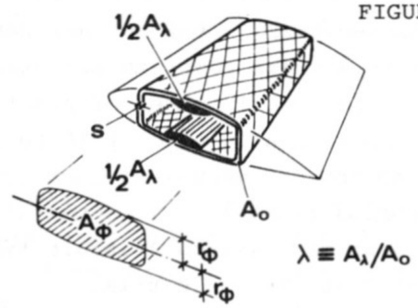
FIGURE 17

parameter $\pm\omega$ on the failure limits if simultaneously axial and shear loads are applied on structural panels. In this figure the volume fraction of supporting layers is assumed to be already optimally chosen in compliance with optimum ranges as presented in figure 8.

If a given wing box or fuselage section is submitted simultaneously to a most adverse combination of a transverse force Q with a torsional M_T and a bending moment M_B the necessary wall thickness s, assumed constant around the whole circumference decides the total cross-sectional area A_o of the wall. (See Fig 18).

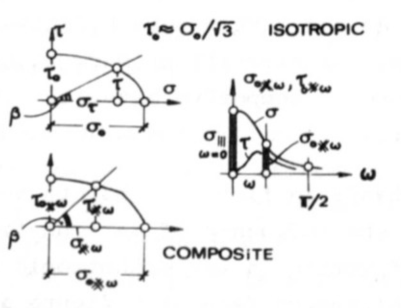
In addition it might be assumed that a group of axial stringers is concentrated close to the region of maximum cross-section height, and the total stringer cross-section may be A_λ.

FIGURE 18



$$M_B = \xi_B A_\Phi s \sigma_{*\omega} + A_\lambda r_\Phi \sigma_{\parallel \omega=0}$$

$$M_T = 2 A_\Phi s \tau_T$$



$$\mu_{QB} = 1 + \tau_Q/\tau_T \qquad \mu_{TB} = M_T/M_B$$

$$\tau_{*\omega} = \tau_T \, \mu_{QB}$$

$$\mu_{QB} \approx 1 + r_\Phi \frac{Q}{M_B} \frac{4(1+\pi\lambda)}{\mu_{TB}(1+4\lambda)}$$

The circumferential wall has to consist of a composite structure of the multi-layer type. The composite stringers loaded only in axial direction are entirely orthotropic.

For the given loading programme this configuration of cross-section may be optimised for a composite with a given volume fraction φ of fibers in the material in the same manner as for an isotropic metallic material. The necessary mass per unit of axial length of that structures shall be compared.

The mass comparison factor

$m_{composite}/m_{isotropic} = C_{cmp/iso}$

proves to be

$C_{cmp/iso} = (\sigma/\rho)_{iso}/(\sigma/\rho)_{cmp} \cdot K_{cmp/iso}$

where $K_{cmp/iso}$ is a factor which represents the influence of the choice of the parameters

$\lambda = A_\lambda / A_o$

for both isotropic material and composites, and ω only for the composite shell.
If $K_{cmp/iso} > 1$, the total area

$\sum A = A_o + A_\lambda$

has to be greater for the composite structure than for the metallic and vice versa.

The comparison of structural masses per unit length of structures with the same cross-section permits an almost general application of the results of this optimisation to any types of sections of primary structures.

In figure 19 the optimum values of $K_{cmp/iso}$, ω and λ are plotted (λ for both materials) against a quotient μ^*_{TB} which represents the relation between shear and normal stress in the structure, where

$\mu^*_{TB} = \mu_{QB} \cdot \mu_{TB}$.

The factor μ_{QB} indicates the influence of the transverse force Q on the maximum shear flow.

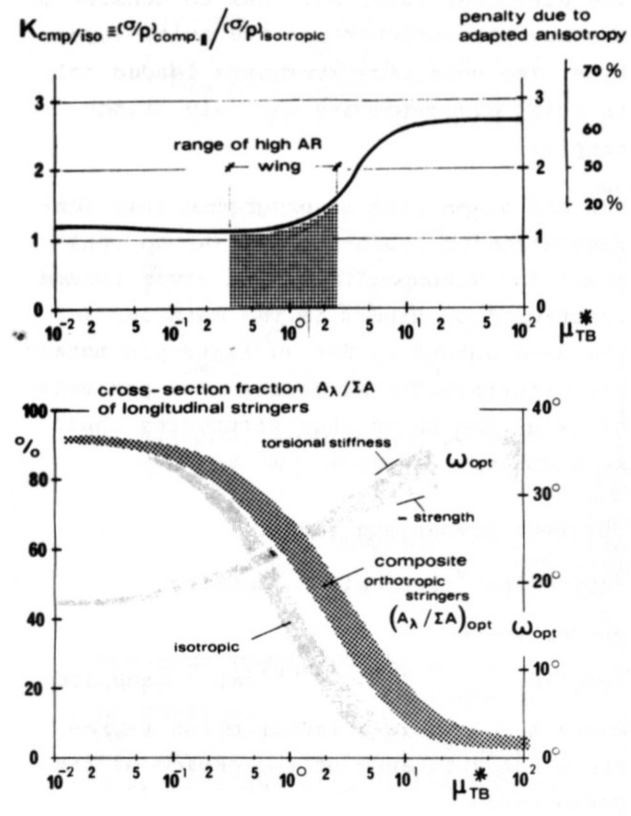


FIGURE 19

μ_{TB} is the abbreviation for M_t/M_b, where M_t and M_b are the actual torsional and bending moment respectively, thus it represents $2\ s\tau/s\sigma$, the strainflow relation if λ would be equal to zero.

For usual aspect ratios, the actual values of μ_{TB} are: $0.3 \lessapprox \mu_{TB} \lessapprox 2.6$ up to some 80% of semispan.

The contribution of the ≈20% of wingtip region to the mass of wings may be neglected in this rough estimate, as μ_{TB} is:

$$\mu_{TB} \approx \frac{\overline{C_m}/\overline{C_L}}{3\,AR} \cdot \frac{1-(Y/S)+(Y/S)^2/5}{[1-(Y/S)/7][1-(Y/S)]}$$

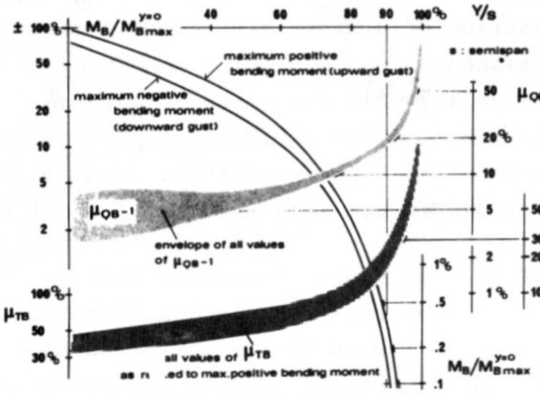
at a moderately tapered trapezoide wing, where $S \geqq Y \geqq 0$, with S for semispan, AR aspect ratio, $\overline{C}_m$ average moment and $\overline{C}_L$ average lift coefficient.

Along some 90% of semispan, μ_{QB} remains close to unity. (See Fig 20).

If we compare the values of μ_{TB} of fig. 20 with the diagrams of fig. 19, it seems obvious that the necessary comparative factor $K_{cmp/iso}$ deciding structural mass related to the $(\sigma/\rho)^{ortho}_{cmp}/(\sigma/\rho)_{iso}$ ratio remains within the boundaries of

$$1{,}10 \lessapprox K_{cmp/iso} \lessapprox 1{,}60.$$

FIGURE 20

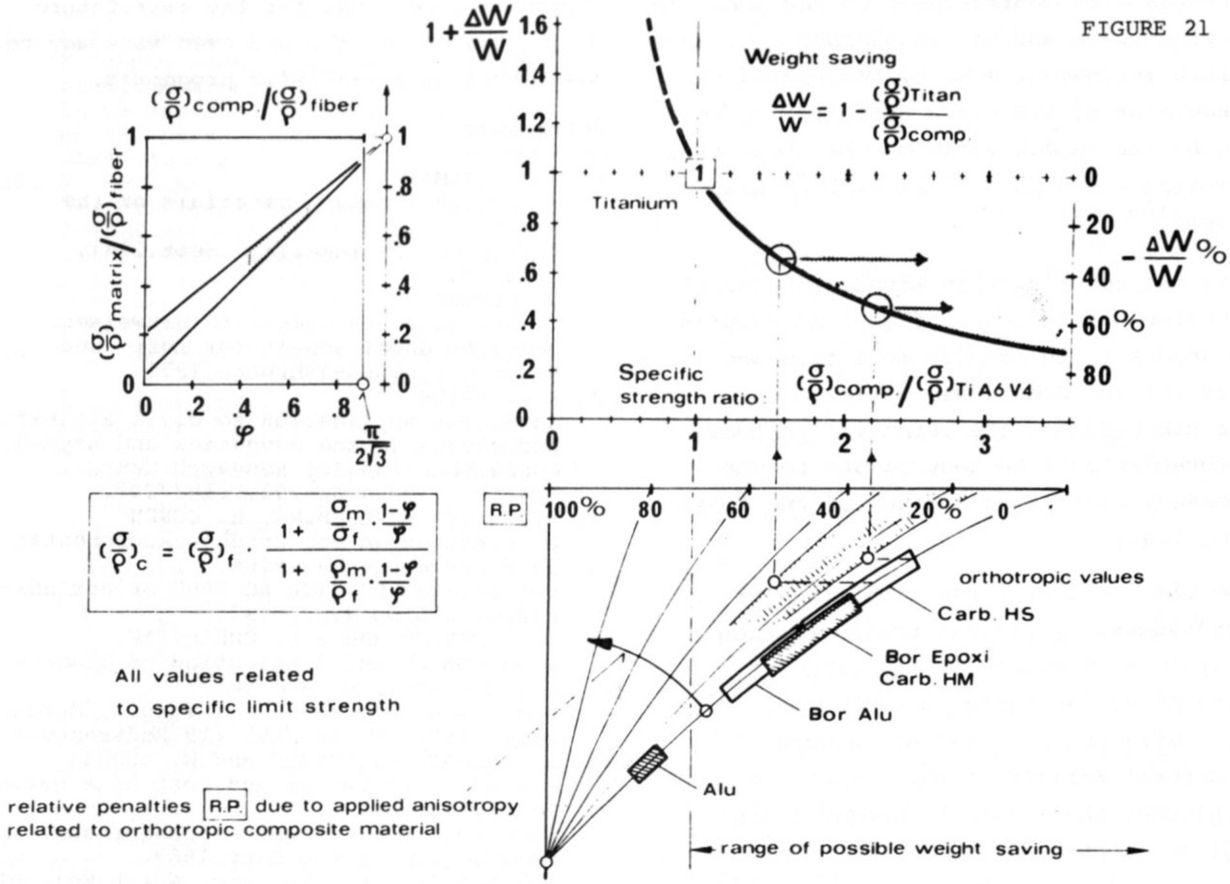


This means that the possible loss due to approaching isotropy from orthotropy only as far as necessary - and not attempting a priori to achieve complete but useless isotropy - implies in cases of practical application a penalty of only 16 to 38% related to the σ/ρ values of best orthotropic composite material, in the direction of fibers at the maximum feasible fiber volume fraction in the matrix.

How much this counts as far as weight saving is concerned, can be read from the diagram figure 21, which explains in synopsis with figure 1 the present situation as well as the future possibilities of mass reduction for aircraft primary structures by the use of composite material. (See Fig 21).

The emphasis in figure 21 points in addition to the influence of maximum fiber volume fraction in the orthotropic composite semiproduct (see also figure 5) as well as on the influence of the rate of penalty $RP = (1 - 1/C_{cmp/iso})100\%$ on the final rate of weight saving $\Delta W/W$ which is one of the possible answers to the problem in question.

FIGURE 21



Applying all the composite data and relations presently on hand - from which the diagrams in this paper present just a tiny extract - the following conclusions seem well justified:

Weight savings in the order of magnitude of 40% are already reality, 50% will be possible soon and 60% and even more can be expected in the future.

These values indicate the weight savings related to the portion of structure which is in fact completely converted into a composite structure.
Figure 1 indicates the degree of reduction if only part of the structure is converted.

At this point, the question arises: "weight savings ... what for?"

This question at this stage of consideration is not an expression of unintelligible scepsis, but indicates that there are other aspects of composite application in primary structures than straight-forward weight saving.

The Bréguet equation states that the range R which an aircraft can reach with a given fuel capacity m_{fuel}, total mass of engines, systems, supplies, fuel reserves and crew m_{syst}, structural mass of airframe m_{struct}, and payload m_{payld} depends on the cruising speed v_{cruis}, the lift to drag ratio L/D and the fuel capacity SFC.

$$R \approx \frac{v_{cruise} \cdot L/D}{SFC} \cdot \ln\left[1 + m_{fuel} / \sum m_{syst.,\,struct.,\,payld.}\right]$$

Now there exist interactions not only between structural mass reduction and possible increase of payload, as is evident, but also between p o t e n t i a l structural mass reduction transformed into higher stiffness with consequences to the possible cruising speed and/or transformed into aerodynamic refinement e.g. by increasing the slenderness of the wings airfoils, or enlarging the aspect ratio of the wing, thus improving speed as well as lift to drag ratios.[36][38]

Using the total gain in structural weight reduction for increase of payload entails the necessity to provide more fuselage space for the additional payload.
As a consequence, the potential payload improvement will be reduced due to the necessary additional surface of the fuselage, i.a..

Thus the trend to a new generation of aerodynamically refined transportation aircraft with supercritical wing[38], advanced active controls technology[36], i.a., will include also an integral structural refinement where advanced composites shall have to render a significant contribution.

Summary

By a partial or total conversion of aluminum alloy and/or titanium alloy primary structures to advanced composite structures, mass reductions of 26 to 39% and more are already reality.

The present state-of-the-art of technology and processing for the manufacture of composite structures on hand would permit to construct entire airframes for any configuration of aircraft.

Well developed knowledge of the anisotropic elastic and failure behaviour of composite light shell structures enables analysts and design engineers to optimize such structures by adapting the shell parameter configuration to any locally given load collective.

Thus, as it has been shown that a large portion of the maximum orthotropic monolayer strength of composites can effectively be used, for the near future weight savings of 50% and even more may be considered as a realistic prognosis.

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