PHILOSOPHICAL TRANSACTIONS OF THE ROYAL SOCIETY A
MATHEMATICAL, PHYSICAL & ENGINEERING SCIENCES

Principles of Design of a Carbon Fibre Composite Aircraft Wing

I. C. Taig

Phil. Trans. R. Soc. Lond. A 1980 **294**, 565-575
doi: 10.1098/rsta.1980.0066

Phil. Trans. R. Soc. Lond. A **294**, 565–575 (1980)
Printed in Great Britain

Principles of design of a carbon fibre composite aircraft wing

By I. C. Taig

British Aerospace, Aircraft Group, Warton Division, Preston, Lancashire, U.K.

The two basic decisions in designing a carbon fibre composite wing are the selection of materials and the form of construction to be employed. The paper outlines the programme objectives, the design requirements and the constraints imposed thereby, and then presents in some detail the principles used to arrive at these decisions.

The materials choice is a compromise between technical, manufacturing, commercial and strategic factors. The form of construction is chosen to obtain substantial weight saving at acceptable manufacturing cost, paying particular attention to four technical factors which are found to dominate the design. These are:

(i) design for integrity in the presence of built-in and accidentally induced stress raisers;
(ii) design for structural stability;
(iii) design for integrity in a service environment including the effects of humidity and elevated temperature exposure;
(iv) design and test margins to give adequate allowance for anticipated variability of structural performance.

Introduction

British Aerospace is developing a combat aircraft wing as the main item in a programme to qualify carbon fibre composites for use in future aircraft structures. Design began when the knowledge of the structural behaviour of the material was incomplete. Likewise, the airworthiness criteria, against which the wing will be assessed, were only defined in qualitative terms. Design and development are therefore proceeding in parallel with large programmes of structural and materials research including statistically significant numbers of tests on structural components simulating a lifetime of operational service.

Programme objectives

These are as follows:

(i) to design, develop, manufacture and demonstrate by ground and flight tests a wing whose basic structure is composed mainly of carbon fibre/epoxy resin composite material;
(ii) to embody design principles and methods of manufacture that are likely to be adopted in the wing structure of a future production combat aircraft;
(iii) to save at least 10 % of structure weight compared with the corresponding structure in the existing metallic wing, which is itself designed to the highest standards of structural efficiency;
(iv) to maintain standards of safety at least equal to the present metal wing and embody design principles to satisfy future damage tolerance requirements;
(v) to embody materials and manufacturing methods suitable for evolution into cost-competitive structures.

General design requirements and their implications

The main structural box of the wing is redesigned in composite as a replacement for a two-spar, multi-rib, integrally stiffened skin structure in aluminium alloy. The c.f.c. box is required to be fully interchangeable with, and to meet all the design requirements of, the metal wing. The design is thus constrained to incorporate the following features:

(i) boundary geometry identical to the metal wing box;
(ii) integral fuel storage;
(iii) wing-fuselage mountings at three main stations on each half-wing;
(iv) two external store pylon mountings on each half-wing;
(v) attachments for leading edge slats, trailing edge flaps and spoilers.

The design requirements for this structure are more severe than those of the metal wing in that it must be capable of operating at higher temperatures and incorporate enhanced damage tolerant design features. It must satisfy airworthiness acceptance procedures which were still undefined at the start of the programme. The strength, stiffness and fatigue loading conditions are basically those for the metal wing. The wing design, development and manufacture were scheduled in a timescale that precluded the use of untried or unfamiliar methods of manufacture. It was also necessary for the wing to be made by using existing major facilities. These requirements resulted in further limitations on the choice of layout as follows:

(i) laminated, moulded construction using pre-impregnated tape;
(ii) cure of major components in an autoclave of limited size, resulting in the need to make the wing in at least two parts with a centreline joint;
(iii) manufacture of each half wing in at least two parts, mechanically joined, to provide access for installing fuel and pylon store systems;
(iv) either localized access to all enclosed bays or the facility to dismantle the wing to provide such access.

The last two limitations led to the selection of separate wing skins bolted to the internal structure. A possible alternative, i.e. building the internal structure integrally with one skin, was rejected on the grounds of cost and the time-scale needed to develop tooling and lay-up techniques.

The operational environment and design criteria

Environmental degradation

Structural integrity is to be demonstrated over the full temperature range and under any environmental conditions which might occur during a normal aircraft service lifetime. In particular, the elevated temperature performance of carbon fibre–epoxy composites is sensitive to the uptake of moisture in service and no practical method is known for inhibiting the uptake of sufficient moisture to cause significant degradation. A survey of American and British data showed that composites in structural components might take up moisture to the extent of 0.8–1.0% of their initial mass and early assessments of property degradation were based on these figures.

Variability and safety margins

From the outset it was necessary to allow for variability of material properties in the critical design conditions, without having access to relevant statistics or knowing what margins would be required. Allowable property values were derived from statistical analysis of coupon tests including notional allowances for environmental degradation. A safety factor of 1.3 (in addition to the customary aircraft ultimate load factor) was applied to all design loads. Subsequently, as environmentally degraded property statistics were obtained, the design data were modified to comply with emerging airworthiness requirements and the supplementary safety factor was reduced to 1.1 to allow for the difference in variability between material test coupons and real components. Before clearing the first wing for flight, a large programme of testing will indicate the strength variability levels actually achieved by structural components manufactured in the same manner as the wing. It is expected that clearance will be based on '*A*' values of component strength (i.e. the values reached or exceeded by 99% of structural specimens, assessed with 95% confidence).

Notch sensitivity and damage tolerance

The wing design must cater for perforation of the main load-bearing members by the assembly bolts, which themselves apply local loads to the structure. An early survey of static and fatigue data on strength of laminates with 'loaded holes' showed a large stress concentration effect on static strength in tension and compression. As illustrated in figure 1, this varies according to lay-up, being most marked in laminates containing roughly equal amounts of material in three or more fibre directions.

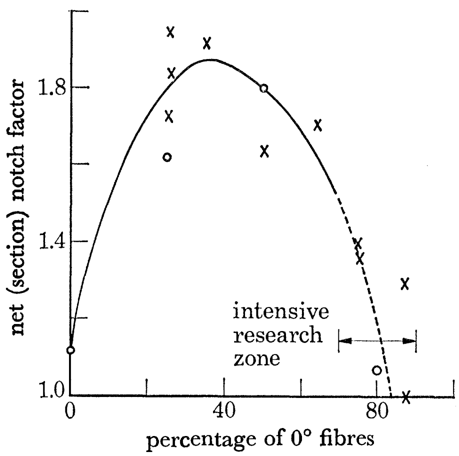

FIGURE 1. Stress concentration of holes; o—o, 0°/±45° lay-up; ×—×, other lay-ups.

The requirement for damage tolerance was interpreted as a need to contain incidental low energy impact or local battle damage under limit load (the highest load likely to be experienced in service). The most likely forms of damage were considered to be penetration of laminates by projectiles and local delamination caused by either low energy impact or local blast pressures. Studies showed that crack propagation under steady load from a damage zone is influenced more by the size of the zone than by its nature or shape. Insufficient data were available to establish precise design criteria, but the principles of limiting initial damage zones and maintaining substantial static and fatigue strength in the presence of damage were of great importance in selecting the structure design.

Choice of materials

A review of existing British, American and part-Japanese materials short-listed six basic fibre–resin combinations. Non-British materials were limited to proven systems for which there is already production and service experience and British materials to those which figure prominently in the national development and data collection programme. With one exception (the fibre type) the list excluded materials which were in the development stage and for which there are, therefore, inadequate data and experience.

The short-list was further reduced by an objective screening procedure in which a wide range of critical technical and manufacturing characteristics were compared by using effective cost-to-achieve-performance as a relative ranking criterion. Two important subjective factors

Table 1. Material ranking chart

criterion	material score† U.K. material 'A' (XAS fibres)	range, all materials	U.S. material 'B' (AS fibres)
basic properties			
interlaminar shear strength	4	2–5	5
flexural strength	4	2–5	4
tensile strength	5	3–5	5
fibre modulus	7	2–10	4
critical properties			
notch sensitivity	8	4–10	4
environmental sensitivity	2	2–6	6
fabrication and handling			
dimensional control	4	2–4	4
laminate quality	4	2–4	2‡(4)
prepreg handling	5	2–5	4
shelf life	6	3–6	4
resin bleed	4	2–4	2
cure cycle	1	1–2	2
total fabrication and handling	24	12–25	18‡(20)
material cost (to customer specification)	2	2–10	10
total (objective factors)	56	30–76	56 (58)
subjective factors			
U.K. national interest	5	0–5	0
supplier confidence		1–5	

† Each point difference between materials represents unit cost.
‡ Low score through lack of experience. Bracketed figure is probably correct rating.

were also included, namely the British national interest and the Company's confidence in the ability of suppliers to meet quality and quantity demands. The objective assessment reduced the contenders to three, which were then subjected to a specific detail design comparison. This, together with the subjective factors, eliminated one more material, leaving two very different contenders on which to make a final judgement. Table 1 shows the objective and subjective ranking of these two materials and table 2 gives a summary of their principal advantages and disadvantages.

Table 2

	advantages	disadvantages
U.K. material 'A' (XAS fibres)	familiarity to B.A.C. processing characteristics: (i) long shelf life (ii) proven zero-bleed system good strength and stiffness properties established support programme in U.K. all-U.K. material	single source supply of fibre and resin relatively high cost no firm data on XAS fibre relatively high environmental degradation doubts about elevated temperature compression strength
U.S. material 'B' (AS fibres)	lowest cost material good handling in its class large production throughput vast experience and data in U.S.A. similar materials can be specified for back-up good environmental stability	relatively low stiffness, notch sensitivity shorter shelf life and tack life not proven for zero bleed little data, experience or support in U.K. or Europe doubts about strategic/commercial commitment of suppliers non-U.K. material

The decision was made in favour of the British material on the grounds of familiarity in our own workshops, minimum programme disruption and absolute assurance of continuing provision of data.

Neither material is considered wholly satisfactory and the decision for this demonstrator programme does not commit the company to continue with the chosen material through to the production of components for new aircraft. Indeed, the parallel decision was taken to include other fibre and resin types in other flight demonstrator components and to seek improvements, notably in the resin and the fibre surface condition, for future airframe applications.

In the case of the fibre, a calculated gamble was made. The bulk of materials data had been compiled for the HTS and Modmor 2 grades of fibre. The cheaper type 3 (or A) materials were insufficiently stiff and were ruled out as contenders in the initial objective assessment. At the time of the evaluation, Courtaulds announced a new fibre (designated XAS) which was expected to have strength at least as high as HTS, stiffness at least as high as Torayca/Thornel T300 and price little higher than type A. On cost effectiveness grounds the decision was taken to specify this material in advance of its evaluation with the option of reverting to HTS at extra cost if it should fail to meet expectations. This reversion has not proved necessary.

Structural design

Choice of layout

The first design decision, given the constraints already listed, was to choose the layout of internal structure and the methods of ensuring structural stability of the load-bearing skins. Of the concepts considered, conventional sandwich panel skins were ruled out on the grounds of loss of fuel volume and lack of damage tolerance. A form of thin sandwich construction in which longitudinal plies were isolated into local regions near the spars (see figure 2) was considered in some detail, as was the more conventional skin–stiffener construction. Both were rejected on cost and complexity grounds, the sandwich concept with some reluctance as it was clearly the cleanest and potentially the lightest design.

Finally, detailed comparisons were made between a number of designs which all featured

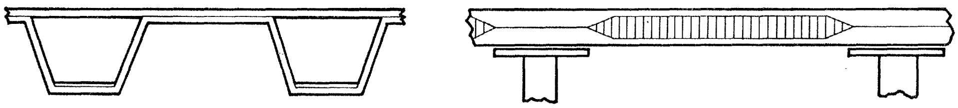

FIGURE 2. Skin panel concepts.

solid laminate skins, multiple spars (longitudinal internal shear members supporting the skins) and ribs (transverse members) only at essential stations.

Internal members

To minimize the numbers of assembly fasteners and the extent of skin perforation, each spar was designed to be attached by a single row of fasteners. The spars themselves must be stable under the action of substantial shear and transverse compression loads. Two concepts, shown in figure 3, were considered for internal composite spars while solid composite laminated channel members were considered for the front and rear spars and for transverse ribs.

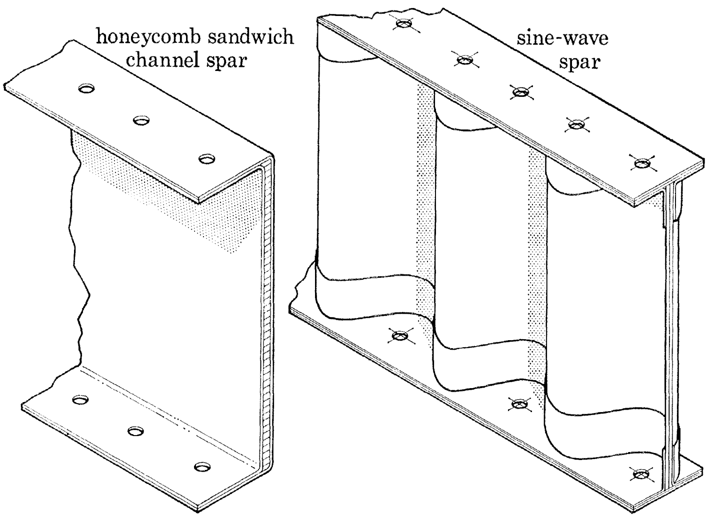

FIGURE 3. Intermediate spars – alternative designs.

The sine-wave spar design was selected, for the internal members, on the grounds of lower mass, higher transverse stiffness (and hence better skin stabilization) and, perhaps surprisingly, lower cost. The selection was influenced by favourable experience with such members in American companies. Manufacturing methods suitable for making high quality components have been developed with surprising speed and success. Figure 4 shows one of the early test specimens made by using this principle.

Skin panel design

The skin panels are the most important load bearing members of the wing and contribute roughly half of the total wing structure mass (nearly $\frac{2}{3}$ in the metal wing). Their design thus

PHILOSOPHICAL TRANSACTIONS OF THE ROYAL SOCIETY A MATHEMATICAL, PHYSICAL & ENGINEERING SCIENCES

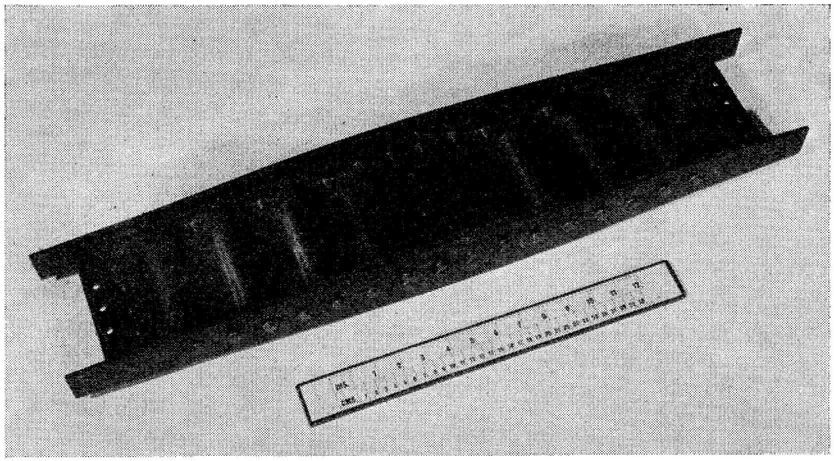

FIGURE 4. Sine-wave spar test specimen.

constitutes the most important single aspect of the programme and is considered here in rather more detail than the others. The first consideration was the effect of perforation and bolt loading and the aim was to improve performance compared with the simple approach of designing down to overall stress levels compatible with laminate notched strength. This meant overcoming the fundamental strain compatibility problem illustrated in figure 5.

Longitudinally oriented fibres must be utilized at higher strain levels than can be tolerated by multi-directional laminates with high notch factors (figure 1). Note that substantial amounts of angle-plied material (in this case chosen in $\pm 45^\circ$ directions) are needed to provide skin stability, to carry torsional and flexural shear loads and to help to carry pressures normal to the skin.

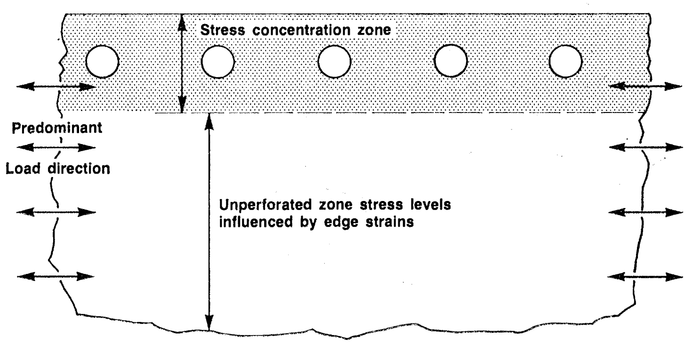

FIGURE 5. Skin-to-spar bolted joint.

Two basic approaches have been investigated and the first of these has two variants. The first approach is to mix longitudinal fibres of different moduli in such a way that the lower modulus fibres are subjected to the geometric stress concentrations at the lines of holes and stiffer fibres in parallel with these are able to carry higher stresses at the limited strains permissible in the perforated zones. Two ways of achieving this are shown in figure 6, referred to as softening and stiffening respectively.

In the softening approach, as shown in figure 7, all the longitudinal carbon fibre layers are replaced by glass in strips along and adjacent to longitudinal bolt lines. In the stiffening approach longitudinal layers of a higher modulus fibre, boron or h.m. carbon, are introduced, either in addition to, or locally replacing 0° carbon layers in unperforated strips parallel to bolt lines. Both concepts lead to difficulties at transverse bolt lines and the stiffening concept, in particular, imposes limitations on the frequency of attachment to transverse internal structure. Both severely complicate manufacture because it is difficult to insert strips of alternating materials within the lay-up and to control their position accurately during cure.

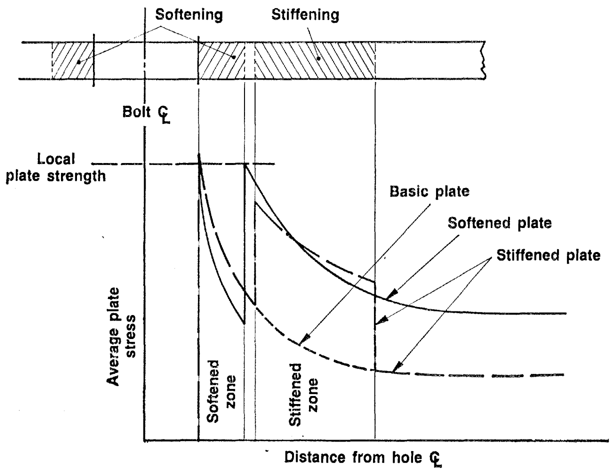

FIGURE 6. Increasing load capacity of skins by softening or stiffening.

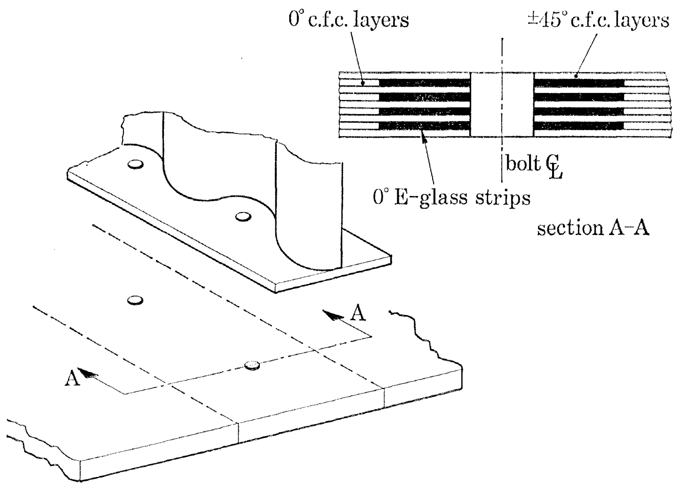

FIGURE 7. Skin softening strip concept.

The efficacy of softening strips in improving static strength and maintaining fatigue strength is shown in figure 8 for strip specimens with very high bolt loads. The strain advantage of the softened strips is considerably greater than shown here because of their lower effective modulus.

The second basic approach, shown in figure 9, is to separate longitudinal load carrying material from shear and pressure carrying material and to exploit the reduced stress concentration factors at both extremes of the $0/\pm 45°$ range shown in figure 1. Basically, $\pm 45°$ material is used for the skin and longitudinal spars with small proportions of 0° or 90° layers to meet local strength and stability requirements. Longitudinal loads are carried primarily by strips of high 0° material with only sufficient $\pm 45°$ (about 20 %) to carry bolt loads and distribute them through the strips. Bolts connect the spars, capping strips and skins and never perforate material which is sensitive to stress raisers. Transverse members are also bolted to the skins without perforating notch-sensitive material. The concept is doubly useful for damage tolerance. First, the $\pm 45°$ skins are insensitive to damage propagation under primarily longitudinal strain as shown by the curves in figure 10 for the effects of barely visible impact damage (b.v.i.d.). Secondly, the individual spar caps, being isolated from each other, provide natural boundaries for containment of damage either to themselves or to the skins.

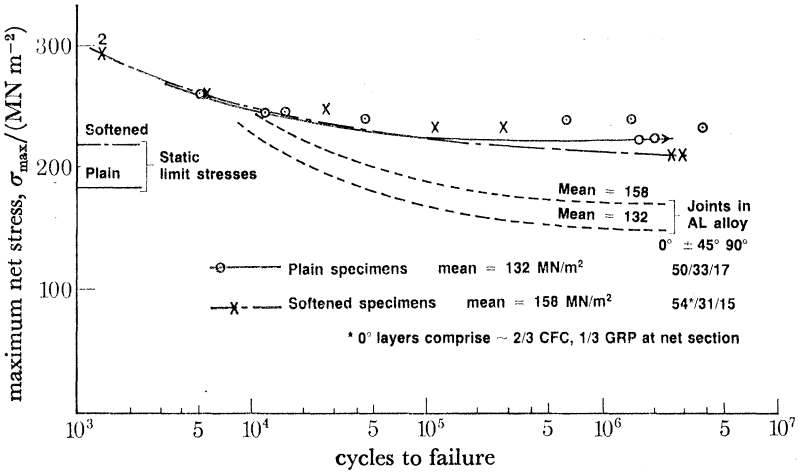

FIGURE 8. Fatigue of single shear bolted joints; material Modmor 2/BSL914.

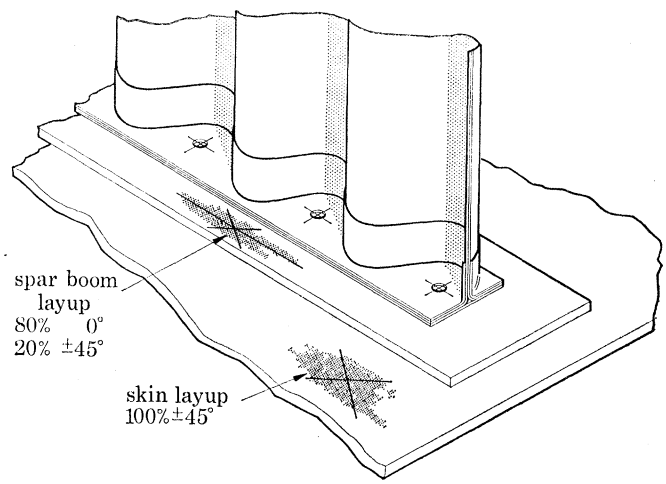

FIGURE 9. Separate flange concept. Both the skins and the spar boom strips use material lay-ups of low notch sensitivity.

The structural disadvantage of the concept is that the skins must either be rather thick, in relation to their primary static strength needs, in order to provide stability and bending strength or the spars must be very closely spaced. Both approaches add weight by comparison with the softened skin concept. From the manufacturing point of view the separate spar cap concept is relatively simple, gives very straightforward skin lay-ups and permits the use of non-parallel spars without any extra cost or complexity. It thus helps to give flexibility to the

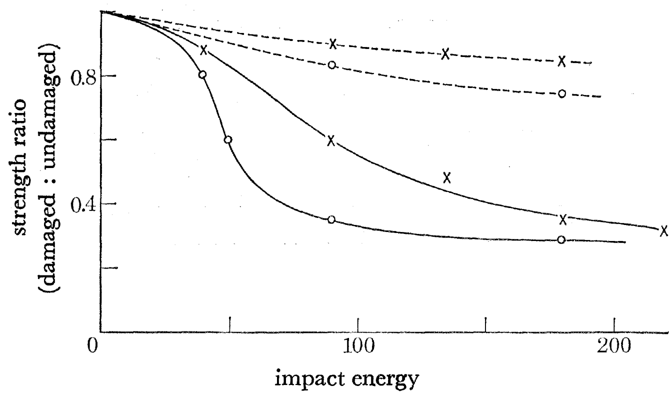

FIGURE 10. Effect of impact damage on tension and compression strength. ×—×, 50 % 0°/±45° tension; ○—○, 50 % 0°/±45° compression; ×—×, 100 % ±45° tension; ○---○, 100 % ±45° compression.

designer and manufacturing planner. To obtain full technical justification of the concept, more work is required on the integrity of the capping strips with high 0° content.

Test programmes are in hand to determine the strength of strips with 'loaded holes' under steady, reversed and cyclic loading over the operating temperature range and taking account of moisture absorption and impact damage. Similar tests are in hand on the skins, although here there is already a substantial body of data to indicate that their performance is likely to be satisfactory. The separate spar cap concept was finally chosen for reasons summarized in table 3.

TABLE 3. COMPARISON OF TWO FINAL DESIGNS

feature	scheme 1	scheme 2	metal
wing box mass/kg			
skins	270.5	295.9	415.6
spars and ribs	108.6	106.8	105.0
joints and mountings	85.9	85.0	59.1
protection and sealant	30.2	29.6	15.0
miscellaneous	47.5	47.5	47.5
total box	542.7	564.8	642.2
peripheral structure and controls	250.1	250.1	250.1
total wing structure mass	792.8	814.9	892.3
damage tolerance	90 kg penalty	yes	limited
production rating	moderate	high	—
suitability for automatic lay-up	difficult	good	—
constraints on layout	parallel spars	straight spars	—
overall assessment	some ongoing tests	chosen scheme	—

Centre line joint and other attachments

The swept-back wing is designed in two halves for many practical reasons and a single major joint is provided at the centre line, as in the production metal component. The wing halves are tapered and have anhedral so that substantial vertical shear and normal-to-surface loads are introduced at the intersection. It was considered prudent to retain an aluminium alloy centre rib to handle these large loads. The wing joint is bolted for assembly reasons and the two alternatives considered were the building-in of metallic inserts during skin lay-up (cf. the F-14 tailplane and other well known composite structures) with bolting through metal only and the reinforcement of the c.f.c. skin to take direct bolt loads into double buttstraps as shown in figure 11.

We already had some experience of both types of design and the latter was chosen as the primary concept on the grounds of cost and simplicity. It involves heavy reinforcement of the skins, first to pick up load from the spar caps and secondly to reduce the stresses at the bolt lines to acceptable levels. Thickness tailoring of skins and buttstraps helps to even out the loads in multiple bolt lines and to minimize offsets and mass penalties.

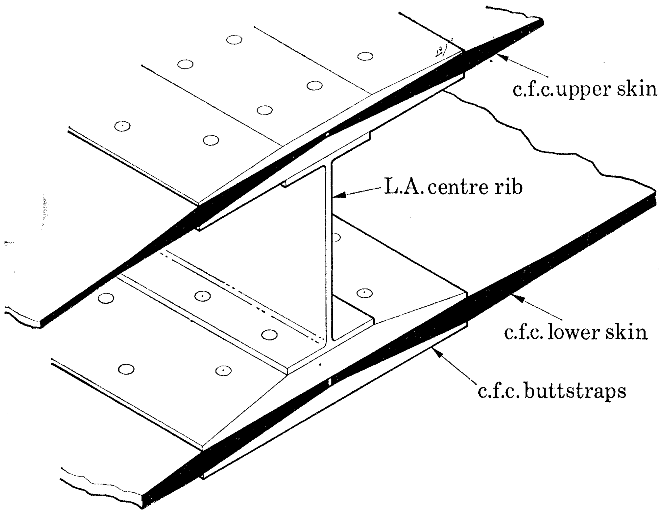

Figure 11. Section through centre wing joint.

An extensive test programme is in hand to prove the integrity of this vital joint and to minimize its mass within the limits of safety.

All other structural attachments are accommodated in a similar way, by building in local c.f.c. reinforcement in skins, spars and ribs and by bolting to metallic or c.f.c. fittings as appropriate. A large and consistent body of design data is being built up to deal with this feature of the structure.

Conclusions

A c.f.c. wing structure is being designed and developed which meets the principal programme objectives without resorting to any exotic or unproved manufacturing techniques. High structural performance is being achieved by exploiting intrinsic properties of the material and by using an 'engineering hybrid' type of construction. The resulting wing is expected to meet existing structural safety standards and to be significantly better in terms of damage tolerance than the metal wing that it replaces.

The writer wishes to thank British Aerospace for permission to publish this paper and his colleagues, in particular Mr. A. N. Rhodes, Mr T. Sharples, Mr R. Haresceugh and Mr R. Whitehead, who provided most of the inspiration and information on which it is based. While most of the design work reported herein was carried out under company sponsorship, the writer wishes to acknowledge the generous support of Her Majesty's Government, who are now funding the whole development programme and the extensive testing to support it.

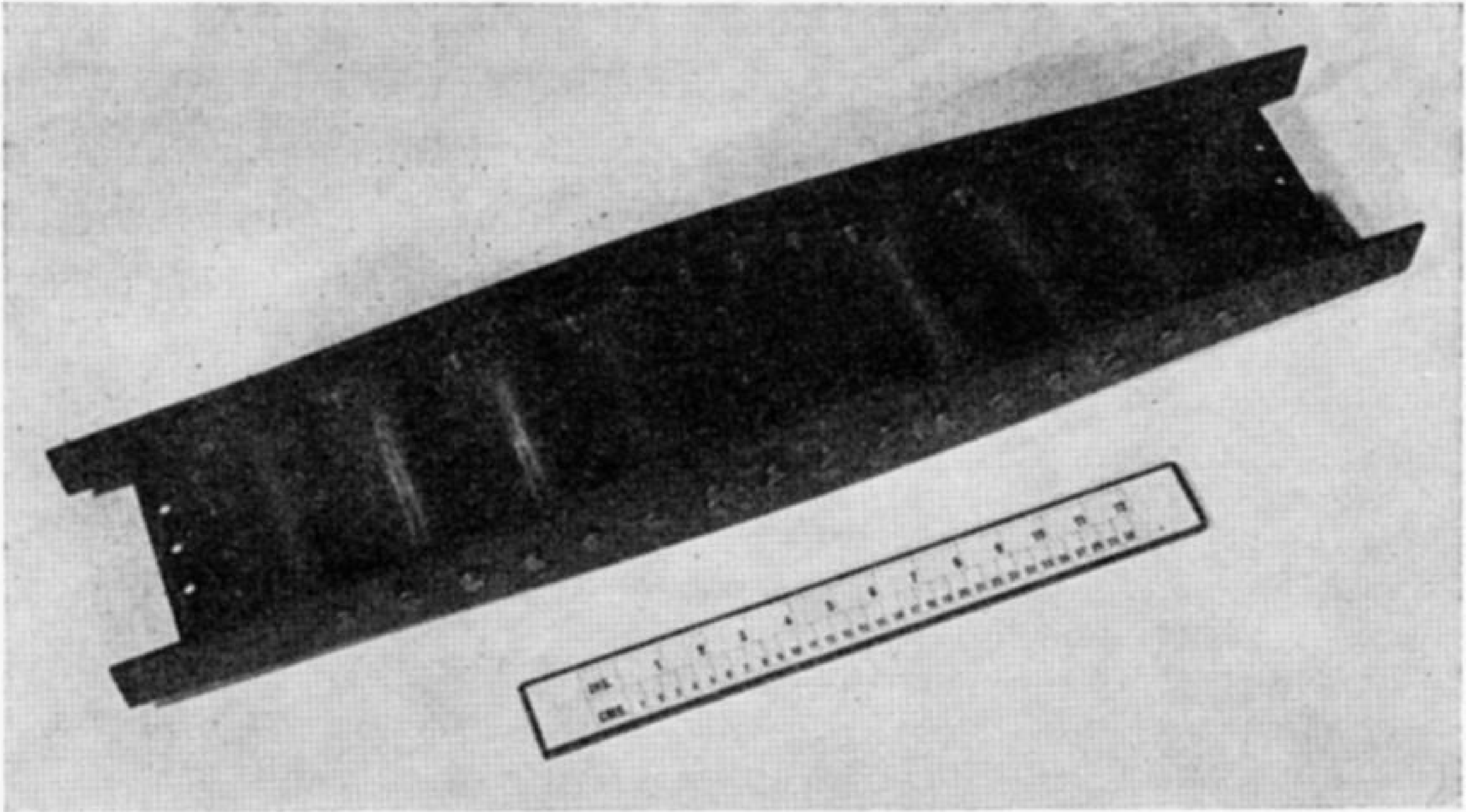

FIGURE 4. Sine-wave spar test specimen.