FACTORS AFFECTING THE DESIGN OF MILITARY AIRCRAFT STRUCTURES IN CARBON FIBRE REINFORCED COMPOSITES

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ABSTRACT

The advent of fibre reinforced non-metallics has had a profound impact on the design of aircraft in general and on the modern military airframe in particular.

An overview of some of the major factors and philosophies relating to the introduction of Carbon Fibre reinforced epoxy matrices is presented.

Some of the influences and limitations of the new materials as they affect the structural designer are examined.

KEY WORDS

Carbon fibre composites; military airframes; Design philosophy.

INTRODUCTION

The principal purpose of this paper is to indicate the basis upon which CFRC components are designed within a modern military airframe design establishment.

It is intended to indicate the variety of factors which have to be recognised and accounted for in the real world of the aircraft structure and it should be noted that the bulk of the discussion is relevant to the CF/epoxy systems currently supplied in pre-impregnated, uni-directional tape form.

It is not proposed to justify the structural benefits and economic advantages made available by the advent of carbon fibres which are topics worthy of discussion in their own right.

DESIGN PHILOSOPHY

In order to use effectively any material for some specific purpose it is necessary to have a proper understanding of its behaviour in whatever variety of forms it is available and under all or any of the situations in which it has to work.

The design aim is to meet the static and fatigue requirements without prejudice to the standards of safety established for the present day metallic structures. In general this means static strength design values consistent with a 95% confidence that only 1 in 100 specimens would fail below the design strength; commonly referred to as the 'A' basis for variability, this is illustrated in Fig. 1. In the case of fatigue the requirement is to have a probability of failure of less than 1 in 1000 at the end of the required life.

BASIC TAPE PROPERTIES

When the specific properties of CFRC tape, as shown in Fig. 2, are compared with those of the other common engineering materials there would seem to be an overwhelming case for its use. However, as is common in this life it is not quite that simple, for the values shown here relate to the uni-directional tape at 60% volume of fibre which is the basic supplied form of the material and in the real world there are few, if any, applications available for its use.

It is necessary therefore to introduce some of the ground rules which enable the designer to plan his 'effective' structure.

USEABLE LAMINATE RULES

Seldom in the aircraft structure can it be truly said that a single, unvarying load system is applied to any structural element and for this reason it becomes necessary to align the fibres in directions adequate to provide for at least the major load axes. The tendency is thus away from true anisotropy and towards a pseudo-isotropic element.

The maximum in-plane shear strength and shear modulus is provided by a $\pm 45^{\circ}$ lay-up which in order to prevent warping, due to the difference in orthogonal expansion coefficients, is invariably layed in stacks of four + - - +.

In order to ensure structural elements conforming to the desired moulded shape every effort is made to achieve symmetry about the mid thickness of the laminate at any point in the plane of the structure. Obviously, there have to be changes of thickness tailored to the load intensities and experience has shown that ply drop-off rates need to be carefully controlled.

In the axes of high stress a maximum gradient of 1:20 is permitted. In directions where the stress is low however slopes of up to 1:10 are allowed.

Blocking of more than 4 plies of any orientation is avoided since it is then possible to obviate the "splitting back" phenomenon which can take place along the fibre direction. Dispersal of the plies also tends to maximise the total number of inter-ply shear surfaces available for load transfer within the laminate.

Every effort is made to ensure that all the laminate 'tailoring' is carried out within a continuous outer envelope 'ply set' of shear carrying material which has the effect of preventing the miriad of small peel actions which occur at the ply drop-off ends.

It is worthy of note however, that extra care must be taken to cover both positive and negative shear applications if shear buckling is a possibility. This requirement results from the possibly dramatic differences in out-of-plane rigidities which can exist in mutually perpendicular directions in the same laminate.

Where the loading action is reversible but of differing magnitude this feature can be used to advantage in attaining a minimum weight structure.

Test evidence leads us to believe that the effects of low energy impact damage can be minimised if the plies at the surfaces are perpendicular to each other.

To re-iterate then, the rules we observe in creating a useable laminate:

- avoid warping
- control tapers
- minimise blocking
- maximise shear faces
- envelope tailoring
- reversible load buckling
- low energy damage

FUNDAMENTAL LAMINATE PROPERTIES

Any structural medium which is built-up from elemental parts having well defined mechanical properties can obviously have an infinite variety of compound properties. Many of these properties will be unusable and some, no doubt, undesirable. It is therefore necessary for the designer to rationalise the range of basic combinations he can reasonably expect to be able to work with, taking into account the production capabilities of his manufacturing colleagues.

The general requirements and behaviour of the airframe structure has led to the choice of the $0^{\circ}/90^{\circ}/\pm 45^{\circ}$ family of laminates and it is to these that the following information applies.

In Fig. 3 we can see the nest of strength curves covering the three basic limits, 100% 0°, 100% 90° and 100% $\pm 45^{\circ}$ and all the combinations available.

These values apply to the particular combination of Hysol-Grafil's XAS fibre made from Courtaulds PAN pre-cursor and impregnated with Ciba-Geigy's BSL 914C resin, but all the currently available high performance fibre/resin combinations will behave in a similar manner.

The compressive strengths in a similar format and the shear strength, which is virtually independant of anything other than the amount of $\pm 45^{o}$ material, appear in Figures 4 and 5.

The tensile, compressive and shear modulii can be plotted in a similar way.

The graphs which have been used here relate to laminates carrying uni-axial applied loads and shear independantly and obviously for any complex loading system it is necessary to apply the rules of an acceptable failure criterion.

Despite world-wide efforts over many years it has not been possible to demonstrate the adequacy of any simple failure law for laminate technology. At British Aerospace the trend is towards checking against a series of zonal criteria which depend on the interactions of the vairous loading modes active within the local laminate. These include:

- fibre tension
- cross fibre shear
- matrix tension ∿ shear interaction
- layer compression ∿ shear interaction
- inter-layer shear

The strain boundaries which result are illustrated in Figs. 6 and 7.

FACTORS WHICH REDUCE THE FUNDAMENTAL PROPERTIES

There are a number of factors which separately or collectively prevent the designer from utilising the full apparent properties of the material and significant among these are:

o Variability	pre-processing post-processing
o Defects	manufacturing impact
o Notches	loaded holes unloaded holes
o Environmental	temperature moisture

It has already been explained how variability is accounted for in the use of the statistical 'A' basis rules for test evidence, we can therefore consider the influences of the other factors.

Defects

Defects can be categorised into two basic types, [i] those occuring during the manufacturing process and which cover inclusions, delaminations and porosity, [ii] low energy impact effects which may be caused by mis-handling, tool slippage, environment hazard [e.g. runway debris] and so on. This form of damage is covered by the generic term Barely Visible Impact Damage [BVID], typically caused by an impact of energy 4-5 joules. The

effects of this damage can be seen in Fig. 8.

The effects of the manufacturing defects are those which determine the inspection standards to which all components are examined by ultrasonics or radiography or both. The standards are set such that no component can start life with a defect likely to cause a loss of strength greater than that attributable to any other damage, notch or environmental cause.

Notches

'Notches' in this context is intended to cover 'engineering holes', i.e. those which are deliberately placed in the structure and are present to fulfill some specific function, i.e. bolted joints, fuel/air flow, access, etc. This means therefore that there has to be consideration given to both filled and unfilled holes in the general laminate.

Since we are dealing with a material which is basically elastic to failure in its general behaviour it is essential that the stress concentration behaviour is carefully considered. With no alleviating plasticity capability the material of the structure will respond to load up to the point of failure.

Figure 9 shows the effects of the filled and unfilled notch on the general field stress for the compression domain and for the open hole in the tension domain.

In dealing with the filled hole tension domain it is necessary to consider a number of features in parallel in order to establish the critical mode. Conventional shear out, tension and bearing failures have to be evaluated. Of further importance to the designer is the magnitude of the load which can be transferred to the bolt or pin in the situation he is considering. Since the laminate in that locality has a finite allowable strength he has to ensure that this is not exceeded by, for instance, the sum of the effects of the concentration due to the presence of the hole in the passing stress field and the concentration due to the pin load itself. Such combinations have to be considered for any of the resultant load systems which can occur at that particular location.

The significance of the tensile stress concentrations due to pin loading can be readily seen for a small selection of laminate configurations in Fig. 10.

So far we have considered the tensile and compressive modes of loading and it is essential that the shear mode is not overlooked. Figure 11 illustrates the shear stress concentration due to pin loading in a plain hole. The influence of 90^{o} material can clearly be seen at the lower levels of $\pm 45^{o}$ content.

Environment

The epoxy resins currently being used in the creation of lightweight structures have been shown to be impervious to the range of fluids commonly met by modern airframes in normal service. These include jet fuel, hydraulic fluid and a wide range of lubricants and detergents. Some paint strippers however have been specifically designed to remove epoxy paints and the operating instructions for composite parts will invariably warn against their use.

There are many composite structures flying worldwide at the present time and the significant number of flying hours accumulating on many of these parts suggests that environmental corrosion is going to be less of a problem than for the current metallic structures.

There are however two fundamental effects which the designer has to allow for; temperature and humidity.

Each of these effects in isolation is not particularly significant but in combination, which is a real occurance, they have an effect on strength which has to be allowed for in the design.

The presence of moisture in the laminate, more specifically in the resin, has the effect of causing a softening of the matrix which therefore reduces its support to the fibres. Since the fibres need this support when in a state of compression it follows that the reduction is most significant in the compression dominated loading situations and to a marginally lesser extent in the shear situation. It is true that transverse properties show a marked reduction in strength also, but since it would not be used as a primary load carrying configuration it is largely irrelevant.

Figures 12, 13, 14 and 15 show the response of the basic strength of a nominally dry laminate tested at ambient temperature, to the absorption of 1% of moisture by weight when tested over a range of temperatures up to what is generally accepted as the useable upper limit for military airframes.

When all these effects are taken into account there emerges a designers "rule of thumb" which enables the first order design to be completed with a high probability that only local areas of particular complexity of loading will need detail refinement. For the typically common range of lay-ups these generalities hold good;

for tension, the unfilled hole, permissible manufacturing defects and BVID allowances tend to coincide, while

for compression, the filled hole and BVID allowances at temperature jointly provide the limit

For reference purposes, the strain limits implied for our particular material combination are:

Tension	4500 $\mu\varepsilon$
Compression	3900 $\mu\varepsilon$

These limits are applicable to any fibre orientation within the laminate for a typical high-temperature environment applicable to high-performance Military Aircraft.

DESIGN OBLIGATIONS

British Aerospace have been actively involved in the progressive introduction of advanced composites technology into the airframe arena since the late 1960's. Indeed the first flying component, a small rudder servo tab on the Jet Provost flew in 1970, followed within a year by an airbrake on the Vulcan bomber.

As one might expect a vast accumulation of data and experience has occurred, as has an understanding of the importance of many aspects of the material which in metallics were of much less significance.

Static Design

There is a basic need to acknowledge the existance of ALL the loading actions which will occur on and within the structure.

Without the advantages bestowed by plasticity even the smallest notch effect, e.g. a ply stop-off, gives rise to a couple, which results in out-of-plane forces in the plate in addition to the stress concentration. This local compound internal stress system has to be accounted for or suppressed in order that the design quality can be maintained. Every possible load or feature which could result in a load has therefore to be scrupulously studied.

The airframe structural analyst has for a quarter of a century made use of computing power available in the industry in order better to understand the effects of the loads applied to his structure and the internal forces which result. Progressively he has taken advantage of the ever increasing speed and power to look more closely at features for which the structure response is not always clear. With composites this philosophy takes a quantum leap forward and through the thickness stress calculations for even the simplest of panels is now routine procedure. To attempt the design of a modern CFRC structure by hand would be folly.

It will by now be abundantly clear that the key to the safe and efficient use of advanced composites lays in the use of very complex detail stressing routines dominated by the power of the computer facilities. The need to automate the routine steps of design has never been more strongly felt.

Buckling

The present day structures in advanced composites are designed to work up to the limits imposed by buckling. Although there is a great deal of research work into post buckling behaviour currently in hand in both industry and the academic institutions, the technology is not yet sufficiently understood for any advantages to be identified and utilised.

It is worthwhile just considering the comment made in the useable laminate rules concerning the attention needed when buckling is a possibility. We have already noted that reversible shear forces can cause embarrassment if a panel is thin enough to buckle, due to the different panel rigidities in perpendicular axes.

Naturally there is no problem of load reversal for compressive buckling but any panel which is subjected to bi-axial compression will, in general, have rigidities which are non-equal in directions of 0° and 90° due to the laminate construction and it is important that the correct compromise is achieved with respect to the load spectrum for any panel in order to achieve the maximum efficiency.

Modern airframes tend to have very little constant property material in their construction, even in metallics there are continuous changes to skin thickness, curvature, stiffener configuration, stringer and frame pitches and so on. This will continue to be the case with composite construction and clearly there cannot be a data bank to cover all the variables. It is necessary therefore to use a first order method to establish a base line structure which can then be refined by more exact computer analysis.

We can see representations of the buckling co-efficients for longitudinal, transverse and shear loadings for infinitely long panels with simply supported edges in Figs. 16, 17 and 18. A similar series exists to cover fixed edges.

Figure 19 allows for the interaction of the three basic loadings and permits the quotation of an overall buckling Reserve Factor.

It will have been noted that the basic curves are based on the percentages of 0°, 90° and $\pm 45^{\circ}$ material present in the laminate and it is implicit that the orientations are uniformly distributed throughout the plate thickness.

Once the plate properties have been established it is possible to invoke the basic rules of lay-up and to introduce the special requirements of the particular panel. This having been achieved, revised plate properties can be specified as required by the Composite Buckling program along with the lamina properties and panel edge loads. The true orthogonal rigidities are evaluated and the Reserve Factors for each loading separately and then for the combined actions are quoted. Any alterations can be made at this stage in a logical sequence.

Fatigue

Long term research programmes to investigate the effects of cyclic loading were set up as part of the overall strategy for the development of carbon fibre reinforced epoxy composite as an engineering material suitable for airframe applications.

A range of lay-ups has been used for the production of coupons and structural elements designed to cover the spectrum of structural features common to air-frame construction. These features include the presence of plain and counter-sunk holes, artificial flaws, induced impact damage, fastener joints etc. The conditions include the full range of temperatures between $-50^{\circ}C$ and $+123^{\circ}C$ in the presence of a realistic maximum moisture content of 1% by weight. By this means the normal environmental history is catered for and it includes the effects of the temperature profile caused by supersonic excursions.

To establish fundamental S-N data, see Fig. 20, endurance tests are carried out under constant amplitude conditions while to simulate flying conditions use is made of FALSTAFF sequencing [Fighter Aircraft Loading Standard For Fatigue].

Residual strength tests are carried out on this continuing programme in order to verify the strength retention values after cyclic loading. The structure does not appear to deteriorate with age, as can be seen from Fig. 21.

The endurance data accumulated so far has revealed nothing to indicate that any further allowance needs to be made to ensure a safe life for a military airframe, the limitations catered for in the static design over-rides any fatigue requirements.

The Pressures to Compromise

The structural designer, having optimised the composite airframe almost to perfection as a load carrying entity, finds himself under pressure from several sources.

The very nature of the airframe and its tasks means that all the associated disciplines involved in its existance are used to handling its complexity, but this does not preclude their interests in pressing for simplifications. Indeed it would be remiss of any discipline which did not argue its case.

It is perhaps worth just amplifying some real points which have arisen as debating points between Design and Manufacture by way of illustration. These are listed in Table 1.

THE TIME AHEAD

The aircraft industry is rapidly coming to terms with the advent of the modern generation of composite materials suitable for Class 1 structural use. The depths of understanding needed to allow competent use of the advantages of the material are being plumbed and month by month the data bank is being filled. There is little doubt that the next generation of military aircraft will make significant use of advanced composites, perhaps up to 40% of the basic airframe mass being carbon fibre reinforced epoxies.

As is the case with all advanced technology industries however, there is no room for complacency and the challenge from new compound light-alloys has to be met. These new metallic alloys offer enhanced strength - weight ratios and use manufacturing capabilities founded on 40 + years of experience. An equally important challenge is likely to come from the advent of superplastic forming/diffusion bonding of titanium.

A major thorn in the side of the airframe manufacturer is exemplified by his inability to capitalise on the benefits claimed for any new composite system without recourse to extensive and costly assessment programmes designed to obtain for him a competent understanding of:

- o the mechanical properties
- o the cure cycle
- o any manufacturing limitations
- o possible hazardous characteristics
- o its long term behaviour

Unless the claimed advantages are proved to be real and significant there is little possibility that economic reality would allow the acceptance of the new candidate material.

Composites technology means a radical change in the factory process. It is important not to overlook the fact that the fabricator is for the first time responsible for the creation of the material as well as the component and this involves an important change to the role of quality assurance. This will be particularly true as bonded construction starts to play an increasing role in the search for structural efficiency.

TABLE 1

REQUIREMENT	EFFECT ON MANUFACTURE	EFFECT ON DESIGN
Maximise number of plies of similar shape and orientation	Rapidity of cutting and economy of handling	Reduction in tailoring, steeper slopes
Co-incident ply stop-offs	Less complex tooling for matching parts	Step changes in thickness, out of plane load effects
Random stacking of plies with joints	Simpler nesting requirements, quicker ply production, less waste	Lower allowable strengths inferred due to adjacent joints, weight increase
Reinforcing external to basic lay-up	Speed and simplicity of component lay-up	Reinforcing not contained, peel effects, stress concentrations, lack of shear faces
Use of woven cloth at surfaces	Significant aid to drilling and drapability	Lower strength than tape therefore increases weight
More open tolerances	Quicker and cheaper manufacture and assembly	Structure performance less predictable, shimmed joints needed, more difficult to seal

Automation has an increasingly prominent part to play in the entire working environment and the computer is becoming a key device. Now that we have developed the ability to handle the new technology the areas for potential improvement become more obvious and the drive for enhancement more logical. It is likely that there is a finite limit to the improvements which can be extracted from epoxy systems and to achieve a quantum leap in matrix properties may mean radically different chemical systems; as the matrices improve they will be able to support new families of fibre reinforcements. Current research into fibre reinforced metal matrices may provide the answers to the materials of the 21st century but for the next 25 years we are likely to find ourselves fully occupied in extracting the maximum benefits from non-metallic composites.

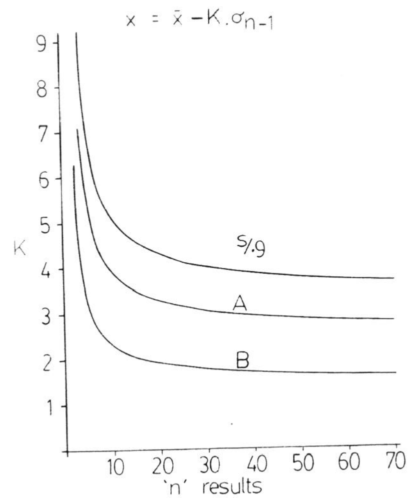

Fig. 1. Variability factors.

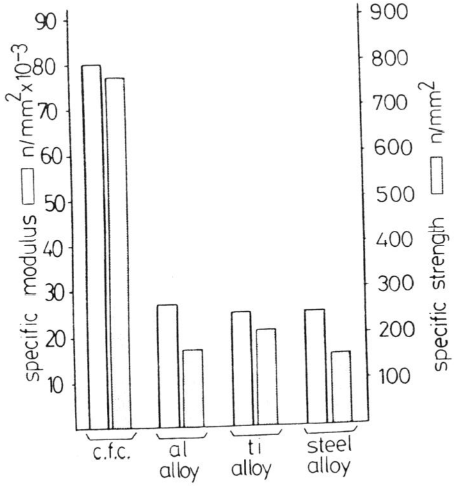

Fig. 2. Properties comparison.

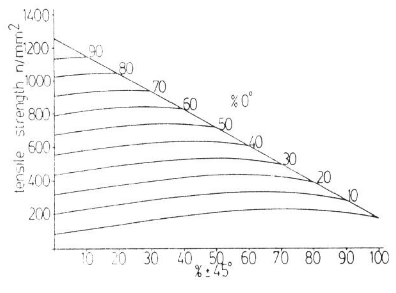

Fig. 3. Tensile strength 0° axis (R.T./dry).

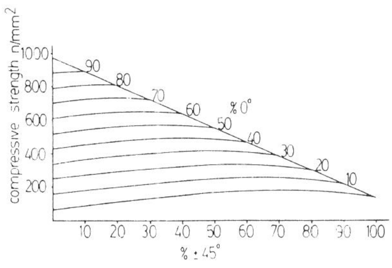

Fig. 4. Compressive strength 0° axis (R.T./dry)

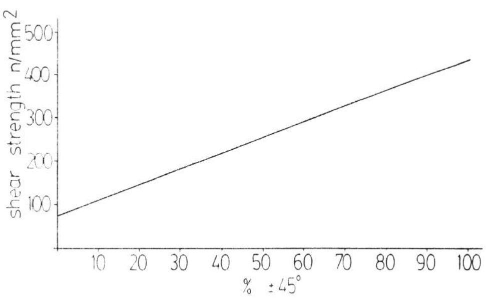

Fig. 5. In plane shear strength (R.T./dry).

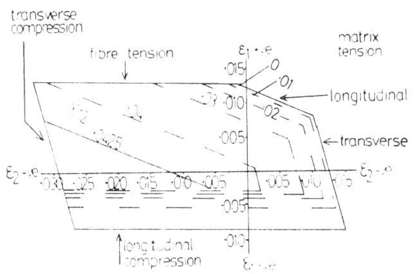

Fig. 6. Layer failure strain envelope (R.T./dry).

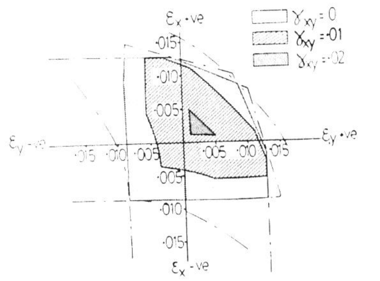

Fig. 7. Failure strain envelope $0^o/90^o/\pm45^o$ laminates.

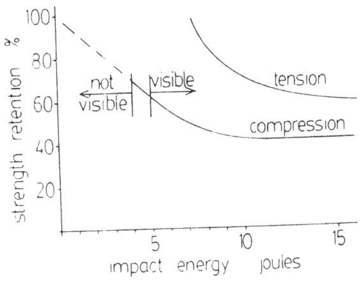

Fig. 8. Barely visible impact damage.

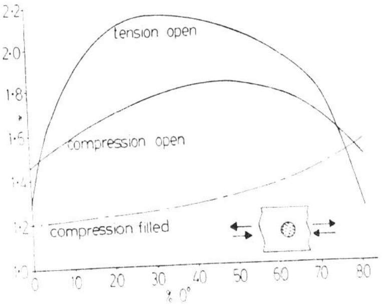

Fig. 9. Notch factors for holes.

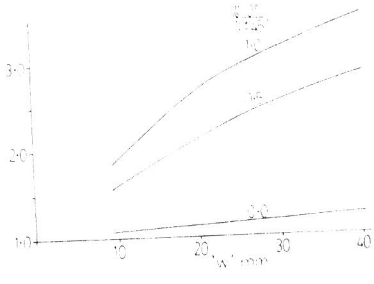

Fig. 10. Tensile notch factor for pin loading.

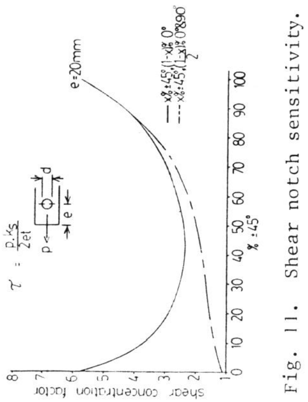

Fig. 11. Shear notch sensitivity.

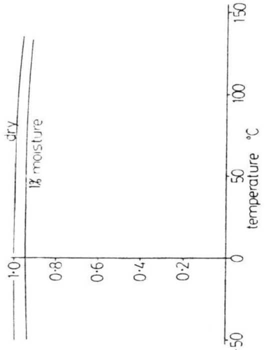

Fig. 12. Longitudinal tension variation with temperature and moisture.

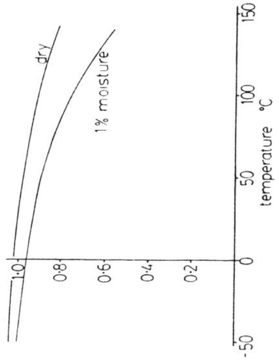

Fig. 13. Longitudinal compression variation with temperature and moisture.

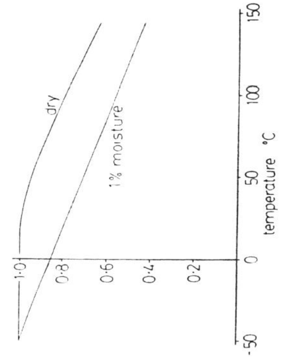

Fig. 14. I.L.S.S. variation with temperature and moisture.

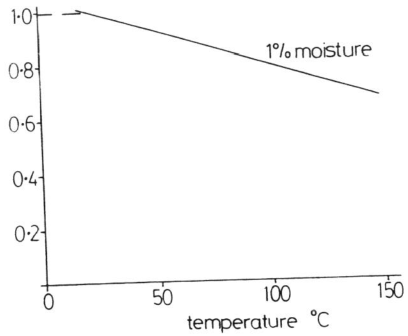

Fig. 15. Bearing strength variation with temperature and moisture.

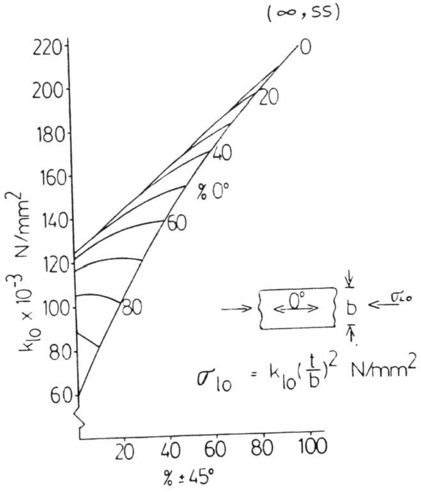

Fig. 16. Longitudinal compression buckling (∞, SS).

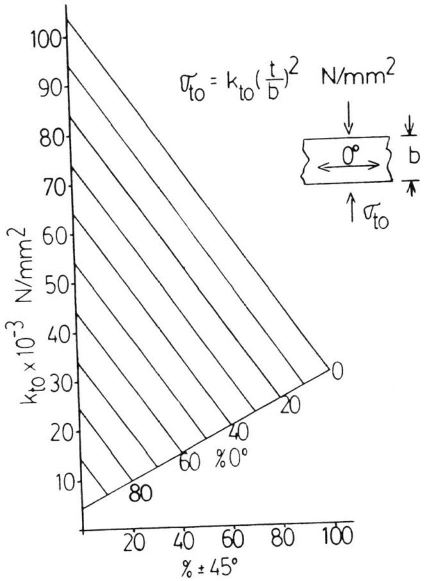

Fig. 17. Transverse compression buckling (∞, SS).

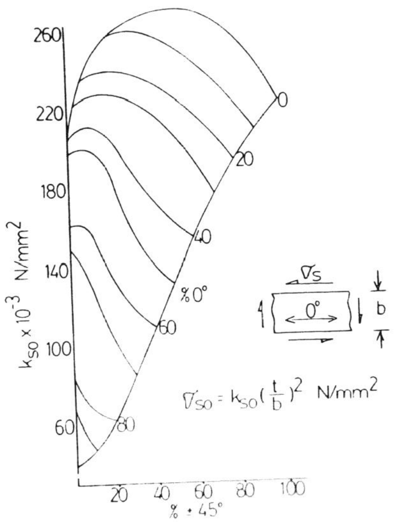

Fig. 18. In-plane shear buckling (∞, SS).

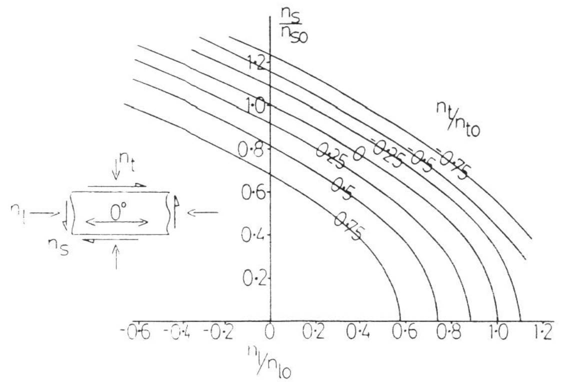

Fig. 19. Buckling interaction curves.

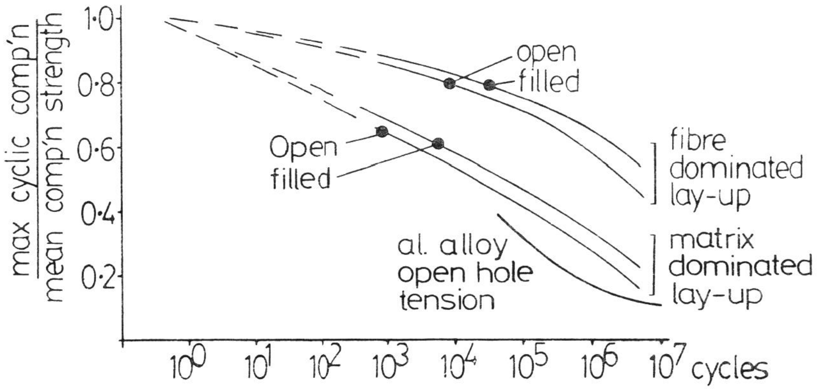

Fig. 20. S-N curves R=-1.

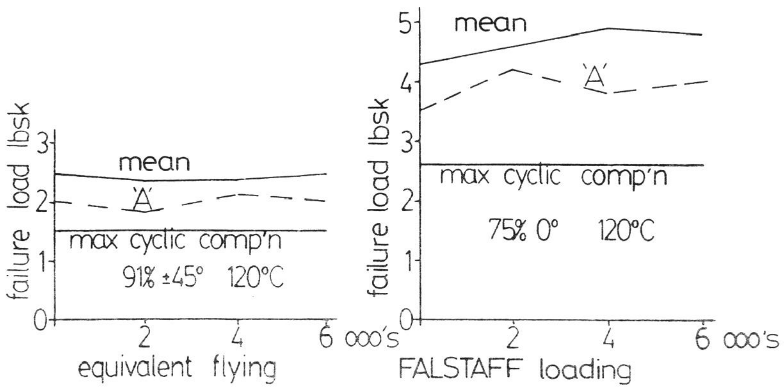

Fig. 21. Residual strengths after fatigue loading.