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LITERATURE RESEARCH ON THE MECHANICAL PROPERTIES OF FIBRE COMPOSITE MATERIALS - ANALYSIS OF THE STATE OF THE ART -

SCHRIFTTUMSRECHERCHE ZUM FESTIGKEITSVERHALTEN VON FASERVERBUNDWERKSTOFFEN
- ANALYSE DES STANDES DER TECHNIK -

VOLUME 1

Published by

J. J. Gerharz

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LITERATURE RESEARCH ON THE MECHANICAL PROPERTIES OF FIBRE COMPOSITE MATERIALS - ANALYSIS OF THE STATE OF THE ART -

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Volume I

Published by

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(Fraunhofer-Institut für Betriebsfestigkeit LBF)

Report No.TB-145 (1979), Vol I, July 1979, Darmstadt

Translator	Translation editor
Barbara Garland	J.R. Heath-Smith

EDITOR'S SUMMARY

This is a comprehensive review of mechanical considerations in the structural use of fibre composite materials, based on 794 references to the literature, with the emphasis on the aerospace application of resin matrix materials with fibres of carbon, glass, boron and kevlar.

The report first discusses data on mechanical properties: static strength of plain, notched and jointed material and the influence of impact damage, environment and creep; fatigue behaviour under constant amplitude and flight by flight loading, the influence of environment and the effect of load cycling on deformation; residual strength following load cycling and impact. It then

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presents physical observations of damage development in different structural situations, and available methods of damage detection and measurement.

Then follows discussion of the application of fracture mechanics to damage growth and residual strength; methods of stress analysis for loading and environment and prediction of failure; structural design procedures for static and fatigue strength; and inspection methods for ensuring quality in production and safety in service. A section is included on test techniques and methods for determining mechanical behaviour of coupons, components and structures, and a section on the absorption of moisture and its effect on mechanical properties.

Finally the report surveys the field of structural applications which exploit composite properties, discusses economic factors and construction methodology, and makes recommendations on the further development of fibres and matrices and on future research.

LIST OF CONTENTS

PREFACE

Fibre composite materials, for some time used only in secondary components, are being used to an increasing extent in primary structures. It therefore became increasingly necessary to create design data inter alia in the field of the strength properties of these modern materials, to derive computation methods, examine inspection procedures and try out new types of test methods.

For this reason an increasing amount of research has been carried out in the above fields, especially in the US aerospace industry. In order to counteract redundancies in this area of research and to provide users of fibre composites with a survey of useable data for the design of components, a mass of published research reports and other literature has been evaluated, commissioned by the Research Sub-department of the Federal Defence Ministry. The information thus gained on the mechanical behaviour of fibre composites and examples on completed fibre composite structures to assist implementation of projected applications are presented in sections 1 to 12. In conclusion future worthwhile research projects are suggested. The prominence of the US in the field of fibre composites means that most of the results are taken from the American literature, the main emphasis being in the area of aircraft construction. A complete list of the references is given in section 13.

The following reports contain additional information on the state of technology of fibre composites:

Title (subject of report)	Report
- Glass fibre reinforced plastics	BMVg-FBWT 73-17
- Fibre reinforced composites	BWDoK Special Issue No.49 October 1977
- Composite materials for engines	AGARD-CP-112, Ref 2
- Joints and cut-outs in fibre composite construction	See Ref 414 (Appendix VII) in section 13
- Design criteria for the use of fibre reinforced materials	Handbook: Fibre composite light construction

These take account of numerous research reports by German companies and institutes. *Deadline for conclusion of literature search was 31 December 1978.*

NOTE: In order to facilitate the location of test results yielded by the references in section 13 on the mechanical properties of specific fibre composites in special parameter combinations, the related source data are catalogued accordingly in an Appendix*. Evaluation of the literature for this Appendix on data of mechanical properties was carried out up to Ref 650. The remaining references to 794 were only used as additional information for discussion of the present state of knowledge in sections 1 to 12.

* The Appendix is not included in the present translation but is available (in German) as LBF Report TB-145, Vol 2.

1 MECHANICAL PROPERTIES UNDER STATIC LOADING

J.J. Gerharz

Contents

1.1 Preface

The following discussion is based on evaluation of extensive relevant literature which is contained in section 13. The mechanical properties of fibre composites under static load are described and some of the essential differences compared with the metallic materials customarily used in aircraft construction are pointed out. The multiplicity of parameter combinations does not permit entering into details of the more complicated physical processes; the reference contained in the text should be consulted in this respect.

1.2 Fundamental investigations

1.2.1 Stress-strain behaviour

In fibre composite materials a distinction must be made - in a similar manner to metals - between brittle and ductile fracture behaviour, even though the causes are different.

- Ductile behaviour

  The area of deformation (along Hooke's line $\sigma = E\varepsilon$) in which strains are proportional to the imprinted external loads is traceable only to a limited extent or not at all.

- Brittle behaviour

  The behaviour of the material is almost purely elastic up to immediately before fracture, breaking elongation is generally slight.

These two modes of behaviour must be regarded as border-line cases, in general smooth transitions are observed. A material known as ductile can also become brittle under certain conditions (effect of extreme temperatures, chemical agents, etc). In metallic materials this is chiefly determined by the structure of the atomic grid and the matrix structure. In fibre composite materials there are additional limiting quantities which result from their stratified structure (composite, laminate) and the interaction of basic material (matrix) and the fibres embedded therein. In this respect a distinction must be made between two typical modes of behaviour[79,105,191,414].

- Fibre-dominant behaviour*

  This term is used when, as a consequence of structure and orientation of the embedded fibres and type of stressing, the load is borne

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* Hereafter briefly 'fibre-dominant composite' and 'matrix-dominant composite' by which the behaviour of the composite is to be understood under the conditions stated.

predominantly by the fibres in the composition and the fibre material thus determines the mechanical behaviour of the composite.

- Matrix-dominant behaviour*

  This term is used when, as a consequence of structure and orientation of the embedded fibres and type of stressing, the load is borne predominantly by the matrix in the composition and the matrix material thus determines the mechanical behaviour of the composite.

A relevant example: A composite symmetrical to the median plane (Index S) in whose individual layers the embedded fibres are arranged in sequence at an angle of 0^{o} / $+45^{o}$ / -45^{o}/ / -45^{o} / $+45^{o}$ / 0^{o}, in brief a $[0/\pm45]_s$ composite, proves to be fibre-dominant under loading in the direction of the 0^{o} layers, Fig 1.1, but matrix-dominant under transverse loading (90^{o}), Fig 1.2. The differences in these two modes of behaviour become even clearer on comparing 0^{o} and 90^{o} composites with $\pm45^{o}$ composites which are stressed in the 0^{o} direction, Fig 1.3 to 1.5. The non-linear relationship between stress and strain in Figs 1.6 and 1.7 points to a matrix-dominant composite.

In most applications the structure of a fibre composite will be selected in such a way that it can be classified as fibre-dominant in the principal loading direction; it will therefore exhibit largely linear stress-strain behaviour and low breaking strain. Fig 1.8 shows the difference in stress-strain behaviour between a $[0/\pm45/90]_s$ composite and metals. It can be seen that for instance breaking strain of Al alloy can be 5-10 times greater than a carbon fibre reinforced composite (CFC), see Ref 414.

At the edge of the hole in notched specimens of $[0/\pm45]_s$ composite approximately 1.5 times greater fracture elongations were observed on average in Refs 117, 252, 414, than in plain specimens. This is contradicted by the practically linear relationship between stress and strain in Fig 1.9 which would really lead one to expect brittle behaviour with little fracture elongation. The reasons must presumably be sought in a statistical scale effect and/or the fact that measurements were made on the matrix material and not the fibre strands.

As a rule the stress-strain behaviour of fibre composites is the same in the tension and compression area (with signs reversed). Differing from this Refs 23, 191, 486 and 719, observe that in fibre-dominant composites the tangent

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* Hereafter briefly 'fibre-dominant composite' and 'matrix-dominant composite' by which the behaviour of the composite is to be understood under the conditions stated.

modulus (slope of the $\sigma\varepsilon$ line) is load-dependent: it rises with increasing tensile load (the stress-strain curve deviates increasingly in the tensile area from the elastic straight line in an *upward direction*); in the compression area the stress-strain curve runs its usual course.

Refs 23 and 719 demonstrated that this behaviour is caused by corrugation and non-alignment of the embedded fibres. With worsening alignment of the fibres strength also decreases while transverse contraction increases[719]. Static trials with unidirectional Kevlar composites[313] produced linear stress-strain behaviour in the tensile area but non-linear behaviour in the compressive area. Compressive strength here is only approximately 1/5 of tensile strength. In contrast, in composites with embedded carbon fibres (CFC), boron fibres (BFC) or glass fibres (GFC) compressive strengths in the direction of the fibres virtually do not differ in volume from tensile strengths. However, across the fibre grain tensile strength is less than compressive strength[25,218].

1.2.2 Special features of notched bars

Fibre-bearing composites react differently from metals to stress concentrations caused by notches (holes, recesses, etc). This is due to the fact that tough metals (*eg* Al alloys 2024 - T3 and 7075 - T6) can compensate for inhomogeneous stress distribution by local plastic deformation and reduce stress peaks whereas most fibre composites are, as already mentioned, fibre-dominant and because of their accompanying brittle behaviour are not capable of doing so. It follows that higher notch sensitivity must be expected in a fibre composite than, for example, in Al alloys (this does not necessarily apply to repeated loading). Consequently the reduction in static strength (related to the nett cross section) due to the notch effect will be noticeably greater in a fibre composite than in light alloys. In addition the strength of fibre composites is more likely to depend on the hole diameter than on the ratio hole diameter/specimen width[44,96,253,414,622,634], Fig 1.10. The effect of the hole diameter on the strength of fibre composites is satisfactorily described by stress-fracture criteria of Whitney and Nuismer[591,620,622,634,763,766], and Waddoups *et al*[622,770].

The stress gradient in the neighbourhood of notches can be arithmetically determined only by the use of extensive calculations (see section 6). These produce greater stress concentration factors for fibre composites than for isotropic metals. On the other hand the ratio of unnotched/notched strength established experimentally on fibre composites is lower than the computed stress concentration factors.

The parameters fibre material and laminate structure have a noticeable effect on the drop in strength of notched specimens according to Ref 252. This shows that the drop in strength in $[0/\pm45/0/90]_s$ composites (BFC and CFC) is approximately 12-25% greater than in GFC for specimens of the same dimensions. Strain measurements at the edge of the hole resulted in:

- CFC : Largely elastic (linear) stress-strain behaviour up to almost 100% breaking load, see Fig 1.11a,
- BFC : Largely elastic stress-strain behaviour up to approximately 40% breaking load, see Fig 1.11b,
- GFC : Plastic strains can be detected even on slight loading, see Fig 1.11c.

Stress-strain behaviour is dependent on the structure and anisotropy of the composite, see Figs 1.1 and 1.2, and provides a reference point for notch sensitivity. Reduction in strength as a consequence of notch effect is greater in fibre-dominant composites with linear stress-strain behaviour than in matrix-dominant composites with non-linear behaviour; the amount of reduction grows with increasing anisotropy of the composite and reaches its maximum in the extreme anisotropic unidirectional composite[252], Fig 1.12.

The behaviour of notched specimens of fibre composite can be improved by laminating on additional layers[343,377,420]. Tests on CFC[63] have shown that the breaking load of notched bars could be increased by reinforcement in the area of the hole to 93% of the breaking load in the unnotched condition. It is also important in this connection whether the embedded fibre strands are severed in drilling or pushed aside by reaming before impregnation and curing of the composite, as shown by an investigation of GFC[420].

The voids in the matrix arising from defects in the manufacture of composites affect the tensile strength at right angles to the fibre direction as well as torsional strength of uni-directional composites[166,174,176]. Transverse tensile strength drops with increasing proportion of voids[176], Fig 1.13. It was found in an investigation of prismatic specimen bars with fibre direction parallel to the bar axis[166] that torsional strength of void free specimens can be 15% greater than those affected by voids.

1.2.3 Special features of joints

The development of component joints with optimum load transfer is of great importance for the quality of practical structures. Various types of joints with load transfer for low weight and their simplified reproduction in the form of specimen bars are shown in Fig 1.14. It is more difficult to achieve optimum design in joints with fasteners such as bolts and rivets in fibre composite materials than in isotropic materials because of their relatively low shear and tensile strength at right angles to the plate plane. Bonded joints are preferred due to their more uniform load transmission for joining fibre composite components, but even here concentration of shear stresses at the overlapping ends must be kept low because of the relatively low shear strength of fibre composite materials. Adhesives with high flexibility are suitable. The most important parameters which (apart from environmental influences) affect static strength according to investigations so far available[23,29,46,152,161,521,549,652,653,657] are:

- Rigidity of adhesive[23,29,152,521] (shear and tension modulus)
- Length of overlap[29,521,652]
- Fibre alignment of layers at the joint face[23,521,549], and
- Joint configuration[23].

In mechanical joints by means of bolts or rivets the tensile strength of the nett cross-section in the rivet line lies below the tensile strength of the undisturbed cross-section in the case of fibre composites in contrast to metals, and must therefore be taken into account in addition to bearing and shear strength which are also critical in metals. Tests on joints in which the bolt transmits the entire load (100% load transfer)[545,547] show that the tensile strength of the nett cross-section depends greatly on the orientation of the layers, the hole diameter and the width of flange per bolt. Composites of 0° and $\pm 45^\circ$ layers (abbreviation 0°, $\pm 45^\circ$ composites) generally produce the best results.

The percentage combination of 0° and $\pm 45^\circ$ layers can be varied over a relatively wide range without the tensile strength changing noticeably, see Fig 1.15. The effect of flange width and bolt diameter is greatest in 0°, 90° composites and smallest in $\pm 45^\circ$ composites. In 0°, $\pm 45^\circ$ composites there is for all bolt diameters examined a largely linear relationship between specimen width and the effective stress concentration factor $\alpha_{k,\ eff}$ which describes the ratio of tensile strength of the unnotched bar and the tensile strength of the joint[545,546], Fig 1.16. As shown by investigations in Refs 44, 222, 545 and 546, the bearing strength of a fibre composite depends in the main on the following parameters:

- lateral resistance to deformation through the clamping force of the bolt,
- orientation of fibres,
- proportionate composition of 0^o layers and angular layers,
- laminate stacking sequence,
- thickness of laminate (number of layers).

In GFC and CFC bearing strength is increased by inhibiting deformation through the thickness at the edge of the hole[548]. This effect is further increased if pressure is exerted by the clamping force of the bolt on the layers of the composite at the edge of the hole. Bearing strength rises with increasing clamping force, see Fig 1.17. At the same time the dependence of bearing strength on the bolt diameter which is observed in unclamped specimens disappears; the orientation of the layers too loses effect under the influence of clamping force.

According to Ref 545, 0^o, $\pm45^o$ and 0^o, $\pm60^o$ composites exhibit higher bearing strength than $\pm45^o$ and 0^o, 90^o composites. If 0^o layers are added to a $\pm45^o$ composite so that their proportion is around 60%, bearing strength rises, *eg* in a CFC from 830-930 N/mm^2. If the proportion of 0^o layers is raised to more than 60% bearing strength drops again. A 60% proportion of 0^o layers also represents a critical value in regard to failure of the specimen: with a lower proportion of 0^o layers the fibres severed during drilling give way at the edge of the hole and failure takes place due to increasing expansion of the hole; with a higher proportion of 0^o layers failure occurs through the specimen splitting in a longitudinal direction, the bolt acting as a wedge[545]. In laminate structure a tendency can be recognised for composites with a more homogeneous structure to exhibit higher bearing strength. For joints without short transverse support the thickness of the laminate has a considerable effect. Tests carried out in Ref 545 show that bearing strength drops sharply if the ratio of hole diameter/thickness of laminate increases, see Fig 1.18.

Resistance to shearing, as expected, depends primarily on the orientation of the layers of the composite. This is shown by results from relevant tests in Ref 545 on 0^o, 90^o, on $\pm45^o$, 0^o, $\pm60^o$ and on 0^o, $\pm45^o$ CFC, see Figs 1.15 and 1.19. Fig 1.20 contains a comparison of tensile strengths, bearing strengths and shearing strengths for 0^o, $\pm45^o$ CFC (HT)* with 1/3 proportion of 0^o layers as against an Al alloy and a steel. In these tests load transfer was 100%.

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* HT - high tensile strength fibre.

1.2.4 Behaviour under impact loading

Despite their brittle behaviour modern fibre composite materials must in many applications be able to withstand the impact of foreign bodies and absorb deformation energy. Since 1972, therefore, the behaviour of CFC and BFC under impact loading has been examined to an increasing extent[644]. Charpy and Izod tests were those most frequently carried out with to a lesser extent falling weight, bending or tensile tests[59,60,100,146,556,642,644,721]. It was found that energy absorption capacity of fibre composites is not affected by notches (*eg* Ref 644).

In the Charpy and Izod tests (pendulum impact tester) the maximum percussive power and energy absorption before and after the occurrence of the maximum percussive power are measured with suitable instrumentation[100] (in fibre composites - in contrast to metals - percussive power reaches its maximum value before specimen fracture). Charpy and Izod specimens experience a three-point bending stress on impact of the hammer which produces both tensile and compressive normal stresses in the specimen as well as shearing stresses. In standard specimens (because they are relatively thick at 10 mm) shearing stresses lead to fracture, as recognised by delamination of the specimens in Ref 721. In thinner specimens ($\leqslant$2.5 mm) failure is predominantly from tensile stresses or else failure occurs through delamination and subsequent buckling of the outer layers on the compression side of the bending specimen.

Results of Charpy tests on unnotched fibre composite specimens[556], Fig 1.21, show that the notch impact strength depends greatly on specimen thickness. In order to be able to arrive at a realistic judgment of the energy absorption capacity of fibre composites, therefore, notched bar specimens should be of thicknesses similar to those used in practice.

In Ref 644, the total energy absorption (before and after occurrence of maximum percussive power) from Charpy tests on uni- and multi-directional composites is compared with metals, see Fig 1.22. Here the energy absorption values for GFC and Kevlar reach the best values of the metals. The values for BFC and CFC are however far lower. In Refs 100 and 644, the energy absorption values up to occurrence of maximum percussive power (*ie* without energy absorption afterwards) were compared, Fig 1.23. This reveals that the tested uni-directional CFC had absorbed as much energy up to occurrence of maximum percussive power as the Kevlar. In judging the notch impact strength of fibre composites, therefore, a distinction must be made according to type of required rgy absorption (before or after occurrence of maximum percussive power), bearing in mind the thickness of the component[642].

The impact energy which the specimen can withstand without damage is established in tests with falling weights[643,721]. Both results of Charpy and Izod tests and results of falling weight tests show that the energy absorption of fibre composites is largely determined by the mechanical properties of the fibres, and that in BFC and CFC energy absorption values can be improved by exchanging a few layers for GFC or Kevlar/epoxide layers[642].

Energy absorption values were established on 20-layer uni-directional composites made of Kevlar layers and CFC (HT) layers in different proportions. The stacking sequence of the layers in the composite was shown to have a greater influence on notch impact strength than the proportional composition of the two types of layer[642].

With increasing intermixture of the two types of layer a significant, if only minor, increase in the total energy absorbed was observed. Further, differences in the time of energy absorption were noted depending on whether Kevlar or CFC (HT) layers were on the outside (surface layers). Composites with three Kevlar layers on each surface took up more energy up to occurrence of the maximum percussive power ('incipient crack phase') than composites with CFC (HT) surface layers. On the other hand, the proportion of the total energy after occurrence of maximum percussive power ('crack propagation phase') was less in composites with Kevlar surface layers than in composites with CFC (HT) surface layers.

The following emerged from falling weight tests and subsequent assessment of damage according to Ref 643:

- Resistance to damage increases with increasing fibre and decreasing matrix strength,
- Resistance to damage drops with increasing fibre and decreasing matrix stiffness.
- Composites with two different fibre directions are more satisfactory than composites with three fibre directions or only one.
- Composites with an intermixed arrangement of layers of different fibre direction are more satisfactory than composites with a blocked arrangement of layers of the same fibre direction.

The damage observed in these tests appeared on the side of the specimen away from the falling weight in the form of cracks parallel to the fibre direction. In thick specimens the first damage may be expected to occur beneath the surface layer(s).

The visible damage alone, however, does not disclose the degree of damage from striking foreign bodies. Thus for example a test of compression-stressed CFC specimens exposed to ball bombardment with increasing energy showed that the strength of the plates had already dropped to 40% of its original value before the appearance of recognisable damage (see section 3).

1.3 Investigations on environmental effect

1.3.1 Parameters investigated

The environmental influences most frequently encountered in operating aerospace equipment are changes in temperature and moisture content of the air. In addition to the media air with humidity and the extreme borderline cases of vacuum and water, fibre composite components of load bearing structures occasionally come into contact with hydraulic and cooling fluids, fuels, etc. The effects of these media are dealt with in Refs 271, 356 and 528. A far greater number of publications deal with the effects of changes in temperature and humidity which have proved problematical for fibre composites with plastic matrix[271]. This is reported below.

As described in section 10, plastic matrix materials absorb water from a humid environment. It is useful to distinguish between environmental stress with and without moisture absorption.

Investigations of environmental influences can be classified as follows:

- Investigations with environmental stress before loading: storage at constand or alternating temperature with or without moisture absorption.
- Investigations with environmental stress during loading: short-term tests at ambient temperatures below or above room temperature, long-term tests at alternating or constant temperature with or without moisture absorption.
- Investigations with environmental stress before and during loading: short-term tests with stored specimens at high or low temperature, long-term tests with stored specimens at alternating or constant temperature with or without moisture absorption.

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Modern fibres of carbon, silicon carbide and boron according to Ref 426, retain their room temperature strength (tension) even at high temperatures (to 800°C). They are also insensitive to the effect of moisture. In glass fibres tensile strength declines with rising temperature, but decrease in strength is only slight within the temperature range permissible for the matrix material; for Kevlar fibres it is greater, see Fig 1.24. However if the fibre detaches itself from the matrix (debonding) glass fibres can be 'leached out' by water penetration[268].

After many years of use and after long-term tests in extreme climatic conditions a deterioration of the properties of fibre materials has never been established according to Ref 528, whether they be glass, boron or carbon. In the organic fibre there are however signs of deterioration from ultra violet rays, *eg* sun light[720]. In contrast, the effect of humidity and temperature changes on the mechanical properties of matrix materials (epoxide resins, polyamides, etc) and the fibre/matrix bond can be great and it is connected with:

- Residual stresses which remain between fibre and matrix and in the layers after the last cooling in the curing process[218,515,569,632,639,647]; the residual stresses can be computed with the aid of the coefficient of thermal expansion of fibre and matrix for each layer, see section 6.
- Residual stresses in the fibre/matrix bond due to inhibition of deformation by the fibre as the matrix swells through moisture absorption[518,566,569]. These residual stresses can be computed, see section 6.
- Falling off of glass equilibrium temperature through moisture absorption by the plastic matrix[92,343,528,670]. Computation of the lower glass equilibrium temperature is possible, see section 10.
- Appearance of fine cracks in the matrix, probably due to spatial expansion on absorbing moisture[528,648].

In every case the behaviour of the matrix material and/or the fibre/matrix bond is crucial to the reaction of a composite to environmental influences.

The matrix materials in common use may react differently to environmental influences. Then temperature and humidity changes in matrix-dominant composites will have different effects depending on the matrix material used. Tests on environmental effects on the mechanical properties of fibre composites with different plastics were carried out in Refs 92, 214, 218, 271, 356, 396, 514, 515, 553 and 667.

The upward range of the operating temperature of fibre composite materials is limited by the glass equilibrium temperature of the plastic. The closer the operating temperature is to the glass equilibrium temperature the more temperature changes affect the behaviour of the composite. Ref 426 contains a survey of the maximum permissible operating temperatures of fibre composite materials, see Fig 1.25. According to this fibre composites with the following matrix materials

Epoxide	up to $150^{\circ}C$	
Polyamide	up to $240^{\circ}C$	$\pm 10^{\circ}C$ in each case
Aluminium	up to $310^{\circ}C$	
Titanium	up to $\approx 500^{\circ}C$	

can be used for components where both strength and stiffness are critical with a saving in weight as compared with metals used by conventional construction methods. As in fibre composites with plastic matrix, the matrix is the decisive component in fibre composites with Al or Ti matrix; in this case the known properties of the Al or Ti alloys can be used to assess the behaviour of the metal matrix composite.

1.3.2 Effect of exposure at high temperature

Storage at constant high temperature without moisture absorption of fibre composites with epoxide resin matrix shows no effect on the static strength and modulus of elasticity, either in fibre-dominant or matrix-dominant composites[436,516,538,667]. An investigation of fibre-dominant 0° and 0°, $\pm 45^{\circ}$ composites in glass/epoxide, carbon/epoxide and carbon/polyamide[667] showed that after 1000 hours only the tensile strength of the uni-directional carbon/polyamide composite had dropped, namely by approximately 13%. The tensile tests were carried out at the storage temperature. In another investigation of HM fibre polyamide composites*[436], a drop in strength of 24% and in stiffness of 18% was observed after 1000 hours/$300^{\circ}C$. In this case the tests were carried out at room temperature.

1.3.3 Effect of variable temperature

Since the coefficients of thermal expansion of the fibre materials (-5 to $+5 \cdot 10^{-6}$ per degree) and matrix materials (20 to $60 \cdot 10^{-6}$ per degree) are distinctly different and the thermal expansions along and across the fibre

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* HM = high modulus fibre.

differ (thermal anisotropy), temperature changes lead to thermal stress changes in the composite which overlie the stresses from applied loading. As explained in section 6,

- normal stresses within the fibre,
- normal stresses within the matrix,
- normal and shearing strains between fibre and matrix can be computed with the aid of micromechanical analyses, and
- normal and shearing strains within the individual layers,
- normal and shearing strains in the entire composite with the aid of macromechanical analyses.

In uni-directional composites thermal stresses occur only between fibre and matrix (micromechanical range[218], whereas in multi-directional composites thermal stresses occur in addition between layers with differing fibre orientation (macromechanical range). For example, in the 0^{o} layers of multi-directional composites tensile stresses can occur transversely to the fibre direction in consequence of thermal expansion; these stresses can exceed the lateral tensile strength of the 0^{o} layers (see section 6) and consequently produce longitudinal cracks there.

The thermal stress-free condition of fibre composites with plastic matrix occurs in the curing cycle at a temperature above the later operating temperature. Thus in the micromechanical range compressive stresses in the fibre and tensile stresses in the matrix arise at operating temperature. Macromechanically compressive stresses arise along the fibre and tensile stresses across the fibre in the 0^{o} layers of an 0^{o}, $\pm 45^{o}$ composite at operating temperature[218]. In computing thermal stresses in the composite changes in the modulus of elasticity (E), modulus of shear (G) and transverse contraction ratio (ν) must also be taken into account.

Some results of relevant investigations are quoted in section 1.3.4, see Fig 1.26. Temperature variations below room temperature (RT) in the area $<0^{o}C$ have, as expected, proved more critical than temperature variations above RT[218]. Specimen bars of an $[0_2/\pm 45]_s$ CFC* which had been heated 100 times from RT to $+138^{o}C$ showed no change in mechanical properties. In a second series of tests specimen bars of the same composite were cooled 100 times from RT to $-73^{o}C$.

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* Abbreviation $0_2 = 0/0$.

Afterwards the modulus of elasticity had dropped by about 25% and tensile strength by about 35%. It should be mentioned that in the laminate structure examined here matrix stress between the 0^{o} layer and the $+45^{o}$ is comparatively high. Stresses would surely be less if the stacking were more homogeneous.

In a different test of $[0/\pm45/0/\overline{90}]_s$ CFC*[516] a drop in compressive strength of 25-50% was observed after 500 and after 1000 temperature changes between $+38^{o}C$ and $+127^{o}C$ or $+177^{o}C$. Tensile strength and stiffness were not affected. In Ref 647, in contrast, a distinct drop occurred in tensile strength and stiffness of a uni-directional CFC after temperature cycling between $-55^{o}C$ and $+150^{o}C$. In this investigation it was conjectured that thermal stresses reduced with increasing number of temperature cycles.

Tests with uni-directional and multi-directional CFC, for which a curing process with maximum 140^{o} instead of the usual $170^{o}C$ was developed, showed no tensile and stiffness loss in longitudinal and transverse directions after 25 temperature changes from $-160^{o}C$ to $+100^{o}C$. Fractographic tests at the end of the experiment showed resin cracks parallel with the fibre which were obviously stopped by embedded carbon fibre tissues in the multi-directional composites[514].

In a test related to the application of fibre composites in space flight[174] the tensile strength in direction of thickness in multi-directional CFC dropped steadily during 50 temperature changes from $-157^{o}C$ to $+66^{o}C$ in vacuum. Here, as also in Ref 514, effects on the coefficient of thermal expansion were observed.

The effect of a combination of constant tensile loading (70% of tensile strength) with 100 temperature changes from RT to $138^{o}C$ and RT to $-78^{o}C$ was investigated in Ref 218. $[0_2/\pm45]_s$ CFC and GFC did not survive the temperature changes above RT without damage. After the subsequent temperature changes below RT both composites exhibited damage which in the CFC led to 20% drop in tensile strength and stiffness. In Ref 218 four further composites with the same laminate structure but different fibre/matrix combinations were tested, and after temperature changes with and without steady loading no loss of strength and stiffness was detected.

1.3.4 Effect of high and low temperatures

As the environmental temperature approaches the glass equilibrium temperature of the plastics their stiffness drops while transverse expansion increases[396,667]. Although the strength of the resin increased after tests in

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* Abbreviation $\overline{90}$ means that only one 90^{o} layer is set in the middle.

Ref 667, all tests with matrix-dominant composites show higher strength losses than fibre-dominant composites[316,396,516,648,667]. The effect of temperature changes on fibre composites presumably rests mainly on changes in matrix stiffness, strength of the fibre/matrix bond and matrix strength as well as on residual thermal stresses at low temperatures.

A drop in the modulus of elasticity of the matrix at high temperatures reduces the support of the fibre under compressive stress and can lead to premature local mechanical instabilities which initiate the crack[54,316]. These instabilities in conjunction with the shear deformations which are greatly increased at high temperatures cause a relatively severe temperature effect on bending strength[321,361,436,648]. Since fibre materials exhibit no temperature effects up to the glass equilibrium temperatures[426], stiffness and strength losses of the plastic matrix must be responsible for the drop in tensile strength and modulus of elasticity frequently observed, even in the case of fibre-dominant composites (including the uni-directional). Polyamide composites according to Refs 3, 92, 119, 316, 516 and 667 have proved to be particularly vulnerable. From an overall point of view, however, the drop in strength is less in fibre-dominant composites than in matrix-dominant composites. A possible explanation for the temperature effect on uni-directional composites is provided by the following observation:

Load transfers take place via the matrix at weak and broken fibres in the 0° layers so that the fibre carries its full loading again at a certain distance from the weak or broken spot. The temperature behaviour of the matrix which is thus highly stressed can therefore affect the strength of the composite[426].

A review of the influence of temperature on the mechanical properties of the CFC (HT) with T300 fibre and epoxide matrix (Narmco 5208) is contained in Fig 1.26 with mean values from five tests from Ref 516. A comparison of the influence of temperature on fibre composite materials with fibres of differing stiffness and strength is undertaken in Refs 92, 110, 195, 316, 383, 396, 436 and 667. Material data for dimensional changes at different temperatures for GFC and BFC are contained in Ref 600. The matrix material is subject to greater stress in composites with fibres of low stiffness than in composites with fibres of high stiffness. Consequently the more highly stressed matrix material is responsible to a correspondingly greater extent for the temperature behaviour of the composite. Therefore the effect of high temperatures on the static strength and stiffness of GFC is relatively great[2,92,600]. At low temperatures the increase in stiffness of the matrix material eases the load on the glass fibres which works out favourably for the GFC composite[195]; at temperatures in the range of -185 to -269°C the strengths of GFC composites are approximately 50% higher than at room temperature[195,383]. While the strength of BFC, Boron-Al and Kevlar are not affected or only slightly affected by these temperatures[33,195,383,667], CFC shows a significant drop in strength at only -55°C[195,383,667]. Probably the higher residual thermal stresses in CFC are responsible which (as already mentioned) may exceed the transverse tensile strength in the 0° layers and may therefore indirectly lead to a drop in compressive and bending strength by reduced support under compressive stress due to longitudinal cracks. Furthermore longitudinal cracks in the 0° layers prevent load transfer via the matrix so that the strength of the weakest fibre bundle determines the strength of the 0° layer.

In notched specimens the temperature effect is no greater or less than in unnotched specimens[416,667]. In contrast the bearing strength of multidirectional composites (T300/5208) is greatly affected; according to tests in Ref 251 it is at 127°C only 70% of the bearing strength at room temperature. The drop in strength of bonded joints is also very great, as shown by an investigation in Ref 161 on a double shear CFC (T300/5208) to Ti 64 bond. At +150°C its strength dropped to 40% of strength at room temperature, whereas it increased at low temperatures. From a resume of numerous test results the following can be stated according to Ref 718:

- in 0° and 0°, ±45°, 90° composites with high modulus fibres, temperature changes in the range of 0-100°C have a negligible effect. In the range of 100-180°C strength can drop slightly, at most by 20%. The drop in stiffness is negligible over the whole temperature range,
- in 90° composites a rise in temperature from 0-180°C causes a severe drop in strength which may be as much as 60%. Stiffness drops by 50% with this rise in temperature.

1.3.5 Effect of humidity

The spatial expansion of the matrix caused by absorption of moisture (see section 10) produces residual stresses in the layers of multi-directional composites which increase with spatial expansion. The residual stresses can be computed by macromechanical analyses (see section 6.3.3) while residual stresses due to moisture absorption can counteract residual thermal stresses.

Moisture absorption leads to a build-up of compressive stresses in the surface area of a composite, which in turn generate tensile stresses in the layers beneath (this follows from the condition of the equilibrium of forces). Moisture desorption has the reverse effect[528,566]. Residual stresses of a macromechanical type due to moisture absorption surely contribute to the change in the mechanical properties of fibre composites in moist environment.

The negative effects of moisture absorption found in many investigations, *eg* Refs 528, 648, 718 and 722 are, however, primarily due to processes which take place in the non-homogeneous individual layer and therefore even in the unidirectional composite lead inter alia to losses in compressive strength in the fibre direction and in tensile strength transverse to the fibre. In considering the causes for the negative effects a distinction must be made between reversible and irreversible processes[172,271,356,518]. Here are some examples:

If fibre composites are dried after moisture absorption and regain their original mechanical properties in a dry condition then the effect was reversible[528]. Epoxide resins mostly behave reversibly[92,271], see Figs 1.27 and 1.28; their strong molecular bonds make chemical attack by water and irreversible changes connected therewith improbable. Weakening of the fibre/matrix bond and finally debonding of the fibre are essentially irreversible[172,356]. Voids and surface cracks generated by rapid warming (temperature peaks) in the resin of wet fibre composite[528] presumably result in permanently raised diffusion coefficients and increased saturation content[214,322,528,614] and are thus also irreversible.

In practice irreversible and reversible processes occur jointly but to a different degree[172,271,340]. Which processes predominate depends in individual cases on the behaviour of the resin on absorbing moisture and on the quality of the fibre/matrix bond. CFC with non-surface treated carbon fibres as Ref 271 for instance, contains more voids than surface treated fibre composite; it therefore takes up more water (see section 10) and suffers a predominantly irreversible drop in interlaminar shear strength.

Both reversible and irreversible changes in the behaviour of the fibre composite through moisture absorption reduce the mechanical properties of the fibre/matrix composite in a moist condition; thus for instance matrix crazing was discovered in the moist resin[648] and is held responsible for the noted loss of transverse tensile strength, modulus of elasticity and interlaminar shear strength at room temperature. The formation of cracks in the matrix through

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spatial shrinkage on desorption of moisture (on drying and in temperature peaks) is presumably intensified in a relatively thick multi-layer composite[528], since the outer layers can contract more rapidly than the inner during emission of moisture and are therefore under high tensile stresses. This effect cannot occur until moisture has penetrated far into the laminate. The distribution of moisture through the laminate thickness can be computed as a function of the total moisture content of the composite and the humidity level of the environment as Ref 573, as has been done, *eg* in Refs 528 and 718. After moisture variation a deterioration in the fibre/matrix bond (debonding) was found fractographically in Ref 369 which led to a decline in strength in relatively thin axially loaded $\pm45^{o}$ CFC.

In becomes clear from the processes of moisture absorption that only those mechanical properties of a fibre composite are subject to changes which are determined predominantly by the matrix material (matrix-dominant composites). These mostly negative changes are always related to the measured time-dependent increase in weight of the composite caused by penetrating water. The form of this relationship varies with the resin system[271]. In resins with a lower saturation content the decline in the mechanical properties will be less than in resins with a higher saturation content[172,214,553]. Speed of diffusion and saturation content also depend on the curing process[214,528], especially when it entails differences in the proportion of voids.

Above are summarised the reasons for the negative effect of moisture absorption on the mechanical properties of fibre composites at room temperature; the essential parameters were also stated. The observed losses of mechanical properties of fibre composites through the effects of moisture and temperature will be shown below. Under these operating conditions the effects of moisture absorption at room temperature are superimposed on those at raised temperatures. The previously described effect of temperature is reinforced in a moist environment by the fact that the so-called glass equilibrium temperature falls with increasing moisture content in fibre-reinforced plastics (see section 10); *ie* moist resins soften at lower temperatures compared to dry ones, where the transition from hard to soft condition takes place over a larger temperature range. This is shown in Fig 1.27 from Ref 92 by means of the bending of resin specimens with rising temperature, for different moisture contents.

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The following paragraphs report on the effect of moisture and temperature on

- Tensile, compressive and shear strength,
- Stiffness
- Creep strength depending on time and creep behaviour
- Notch impact strength.

The results of investigation on the effect of temperature and moisture content on the tensile strength and modulus of elasticity of fibre composites can be summarised as follows according to Shen and Springer[718,722]:

- on tensile strength:

  according to Refs 92, 214, 321, 516, 648, 667 and 720 in fibre-dominant 0°, $\pm 45^{\circ}$, 90° composites the moisture content of fibre composites has only minor effects on tensile strength. For moisture contents below 1% the effect of moisture appears to be negligible. Above 1% moisture content tensile strength drops with increasing moisture content; the maximum drop is around 20%. The drop in tensile strength, however, appears to be *independent of the temperature*. In matrix-dominant 90° composites the influence of moisture content on tensile strength is significant. The drop in strength depends both on the moisture content and on the temperature. The drop in tensile strength can be 60-90%.

- on modulus of elasticity:

  according to Refs 214, 321, 516, 648, 677 and 722 there were only very small changes in the modulus of elasticity in fibre-dominant 0°, $\pm 45^{\circ}$, 90° composites in the ranges examined of moisture content (0-2%) and temperature (-75 to +180°C). In matrix-dominant 90° composites stiffness drops considerably with increase in moisture content. The decline in the modulus of elasticity depends both on moisture content and temperature and reaches values between 50 and 90%.

In most tests moisture was not evenly distributed within the specimens. In fibre-dominant composites differences in moisture distribution do not appear to affect results. In matrix-dominant composites the moisture distribution can affect the amounts of strength and rigidity but it will not alter the tendencies in the data. Figs 1.29 and 1.30 show strength and modulus of elasticity respectively as a function of test temperature and moisture content for a

CFC[718,722]. The effect of moisture content on compressive strength of fibre composites has been examined less frequently; the results described in Refs 92, 214, 516, 528 and 614 are based on different experimental methods. In Refs 214 and 516 it was found in fibre-dominant and matrix-dominant CFC after storage in an environment with constant humidity that compressive strength had dropped by up to 25% both at room and higher temperature. In environmental stressing with realistic environment simulation (moisture absorption and temperature change) the drop in compressive strength at high temperatures can be greater[92,528,614]; it is then up to 40%.

A pronounced moisture and temperature effect is observed for interlaminar and intralaminar shear; however it has different causes depending on the type of loading. For intralaminar shear (within a layer) moisture and temperature influence on the fibre/matrix bond is decisive, for interlaminar shear (between the layers) it depends primarily on the behaviour of the matrix material[172,516].

Tests with interlaminar shear stress in Refs 92, 172, 271 and 528 show drops in shear strength through moisture and temperature effects of 40-60%. Results in Refs 172 and 271 show a distinct increase in the drop of interlaminar shear strength with increasing moisture content in CFC. The effect on breaking elongation is noteworthy as it can increase threefold through moisture content at high temperature[516,648]. The effect of temperature and moisture on bending strength[92,309,321,528,648] is governed by the effects on shear strength and compressive strength of the material concerned.

It was observed in Ref 648 that the effect of moisture on bending strength of a composite is caused indirectly by a drop in interlaminar shear strength due to moisture absorbed. Confirmation of this can be extracted from Fig 1.31. This shows the effect of cyclic moisture absorption on the bending strength of uni-directional epoxide resin composites with fibres of differing stiffness[92]. According to this the glass fibre composite (lowest stiffness fibre) shows the greatest drop in bending strength and the carbon-HM composite the least, since shear stress decreases with increasing rigidity.

The increase in the drop in strength of dry and moist composites, however, takes a different course with rising test temperature for shear stress (short bending test) and bending stress as shown by a comparison in Ref 321 and a comparative table in Ref 528 for realistically aged $\pm 45^{\circ}$, 90° CFC with 18 layers, see Fig 1.28. In changing environmental conditions almost constant moisture

content is established in the interior of fibre composite plates after long operating periods according to Ref 565, while the moisture content in the surface layers varies continually with the environment conditions (see section 10). In changing environmental conditions therefore the outer layers suffer corresponding changes in expansion or residual stress which can contribute to damage.
In order to reveal this proportion of damage comparative bending tests with and without the effect of moisture are recommended. The table, Fig 1.32, taken from Ref 528 gives an insight into the drop in static strengths and moisture absorbed (percentage by weight) after real and simulated changes in environmental conditions during long periods of service in aircraft (1-20 years). Static tests on realistic components of GFC, CFC, BFC and CFC + BFC were carried out at RT and +127°C. Only at 127°C did static strengths drop noticeably.

The effect of natural climatic change on 0 ± 45° composites with different epoxide resins was examined in Ref 515. Static tests were carried out during 18 months' storage in sub-tropical and temperate marine climates. Shear strength and breaking elongation fluctuated most in all resin systems, mechanical properties dropped noticeably in only one resin system. Surface treatment with polyamide primer and polyurethane enamel proved to be advantageous (see also section 2).

The following is all that can be said at present on the effect of temperature peaks:

After repeated rapid heating to high temperatures (150°C) considerably increased diffusion speeds and saturation amounts were observed in carbon fibre epoxide resin composites[214,322,518,528,614]. It is generally suspected that faults and surface cracks in the resin develop under the effect of temperature peaks. At present systematic investigations of the effect of temperature peaks on the mechanical properties of fibre composite materials are not available.

1.3.6 Creep strength and creep behaviour

The plastics used as matrix material in fibre composites (epoxide resin, polyester resin, polyamides and other thermoplastics) tend to creep because of their visco-elastic behaviour (to a considerably greater degree than metals) so that their strength and stiffness depend on time and temperature. By reinforcing the plastics with 50-60% volume fraction of carbon, boron, glass or Kevlar fibres, however, creep behaviour approximating to that of metals is achieved. The effectiveness of reinforcement depends on the fibre material used. For instance, the creep resistance in the fibre direction of a uni-directional composite with embedded Kevlar fibres proved to be three times greater than S-glass composites. CFC too is judged to be particularly resistant to creep, which is confirmed in investigations by the RAE[535,658,666]. Results show that creep strains in 0°; 0°, $+90^{\circ}$ and 0°, $\pm45^{\circ}$ CFC after 1000 hours at $+80^{\circ}$C stressed at 80% of tensile strength are very small in comparison with aluminium alloys (2024, 7075). This also applies to uni-directional boron-epoxide and 0°, $\pm45^{\circ}$ boron-aluminium composites[657].

In matrix-dominant 90°, $\pm45^{\circ}$ and $\pm45^{\circ}$ CFC composites creep strains were considerably greater, see Fig 1.33, and the $\pm45^{\circ}$ composite behaved like non-reinforced plastics[666]. Similar results were obtained in investigations in Ref 173 on the effect of fibre orientation on time creep strength of GFC in air and benzene. Load and temperature act as in metals: their increase enlarges creep strains[21]. In creep tests[87,716,720] of fibre strands in epoxide with 50-100 specimens per series, relatively wide scatter was observed, as also in Ref 173. For the fibre composites tested, GFC, CFC and Kevlar/epoxide as well as beryllium wire in epoxide the scatter in creep strength was 20-100 times greater than that in tensile strength[720]. An evaluation of the fibre/epoxide composites tested for creep strength produced the following order:

- Carbon (maximum creep strength)
- Beryllium
- Kevlar
- S-glass (minimum creep strength).

The creep behaviour of GFC/polyester composites is described in a different investigation[398]. Several publications by the Institut für Kunststoffverarbeitung an der RWTH Aachen, *eg* Ref 246, also deal with the creep strength of GFC. The effect of temperature and moisture is also treated there.

Creep strength and creep behaviour of CFC and BFC with plastic matrix have also been investigated repeatedly[21,181,320,425,455,516,666,667]. Although results available so far lead to the assumption that there are moisture and temperature effects, it is not yet possible to quote quantitative data because of the small number of tests.

1.4 Scatter of test results

Since fibre composites in general do not plasticize and therefore incline to spontaneous failure, it is necessary to take account of the degree of scatter of strength chracteristics, see Refs 50, 157, 158, 159, 161, 221, 343, 399, 401, 473, 484, 518 and 668 when laying down permissible loads (compare the 'A' and 'B' values in MIL Handbook 5A). As in metals, the scatter of strength values of fibre composites is affected to a major extent by the manufacturing process, quality control, predominant type of fracture, size of specimen and test method and to a lesser extent by the orientation of the layers, the environment and notches.

It has been observed frequently in fibre composites that the variation coefficient (percentage ratio of standard deviation and mean value) as a measure of the scatter of the strength characteristics is largely independent of the type of loading, layer orientation, temperature and shape of notch[113,119,191,414], but greater than in metals. For instance, the variation coefficients of tensile strength of BFC, CFC and GFC according to Refs 113, 119, 179, 191, 254, 414, 634, 778 and 784 amount to between 5 and 13%*. Further data on the scatter observed are to be found in Ref 161 for bonded joints, in Ref 119 for breaking stresses in BFC, in Ref 634 for notched specimens (with hole or crack), in Ref 276 for short-fibre GFC and in Ref 720 for time creep strength of CFC, Kevlar and GFC.

For the future it is to be expected that with further development of quality control in the manufacturing process from material constituents (fibre and matrix) to structural components a progressive reduction in the scatter ranges of strength characteristics will be achieved, as is already shown in the development of BFC technology[414].

1.5 Summary

To summarise, the following can be stated on the subjects mentioned:

Stress-strain behaviour

- if approximately 25% or more of the layers of a CFC or BFC have fibres in the direction of load then the behaviour and the mechanical properties of the composite are determined by the properties of the fibres and the composite is termed 'fibre-dominant',
- if less than approximately 25% of the layers of a CFC or BFC have fibres in the direction of load then the behaviour and the mechanical properties of the composites are primarily determined by the properties of the matrix and the composite is termed 'matrix-dominant',

* For comparison, the variation coefficients for metals are less than 5%.

- in tensile and compressive stress fibre-dominant CFC and BFC exhibit linear stress-strain behaviour and matrix-dominant non-linear stress-strain behaviour,
- with inter- and intralaminar shear stresses the stress-strain behaviour of all fibre composites with plastic matrix is non-linear,
- since in CFC the fibres do not run exactly parallel the following deviations occur from the linear stress-strain behaviour: the tangent modulus rises with increasing tensile stress and drops with increasing compressive stress.
- faulty alignment and parallelism of the fibres reduces strength and increases the transverse strain coefficient of fibre-dominant composites,
- in fibre-dominant CFC, BFC, GFC and boron/aluminium the absolute value of tensile strength is in general equal to the absolute value of compressive strength,
- fibre-dominant composites with organic fibres (Kevlar) are non-linear on compression; their compressive strength is only 1/5 of tensile strength (ratio of absolute values),
- across the direction of the fibres compressive strengths are higher than tensile strengths (absolute values),
- the position of the weakest layer (90° layer) in the composite can affect the strength and the shape of the stress-strain curve,
- despite local linear behaviour higher strains can occur at the notch than at tensile failure of the unnotched specimen.

Static strengths of notched components

- stress concentrations in specimens with open holes and in joints reduce the static tensile and compressive strengths in comparison with the strengths of unnotched specimens,
- static strength drops with increasing hole diameter, this drop can be estimated successfully with the fracture hypotheses of Whitney, Nuismer and Waddoups,
- the drop in strength due to the notch becomes greater with increasing anisotropy. Computed stress concentration factors also increase with increasing anisotropy; they are however always greater than the ratio of the static strength unnotched to the static strength notched,

- in fibre-dominant GFC the drop in strength due to the notch is less than in BFC and CFC,
- the static strength of the notched specimen can be raised to that of the unnotched by laminating on $\pm 45^{o}$ and 0^{o} layers at the hole.

Static strength of joints

- mechanical joining of fibre composite parts by means of load transferring bolts gives the greatest strength in the nett cross-section in composites of 0^{o} and $\pm 45^{o}$ layers,
- bearing strength is considerably improved by the lateral support conferred by clamping of the composite at the edge of the hole,
- bearing strength is greatest in 0^{o}, $\pm 45^{o}$ CFC with 60% 0^{o} layers,
- without lateral support bearing strength drops considerably with increasing ratio of hole diameter to composite thickness (0^{o}, $\pm 45^{o}$ CFC),
- strength against shear-out depends on the orientation of the layers in the composite; 0^{o}, $\pm 45^{o}$ CFC (HT) with 40% 0^{o} layers has high shear-out strength.

Behaviour under impact load

- energy absorption related to the cross-sectional area (nett cross-section for notched specimens) is equal in both unnotched and notched specimens,
- energy absorption of bending specimens is greater in thick specimens (failure due to shear stresses) than in thin specimens (failure due to normal stresses),
- the mechanical properties of the fibres determine the energy absorption capacity of the composites. CFC and BFC absorb less, GFC-Kevlar the same amount of energy as metal materials,
- hybrid composites with well mixed GFC and CFC layers or Kevlar and CFC layers absorb more energy than pure CFC, furthermore it is of advantage for the GFC or Kevlar layers to be on the surface of the hybrid composite.

Environmental effect (General)

- environmental effects on the mechanical properties of fibre composites are related to:
    - the coefficients of thermal expansion which differ for matrix and fibre,

- the thermal stress condition after curing,
- the change in volume of the composite on absorption or desorption of moisture, and
- the fall in the glass equilibrium temperature due to moisture absorption.

- the transverse tensile stresses in an angle ply composite resulting from different thermal expansions of fibre and matrix can exceed the transverse tensile strength of the 0° layer,
- in general the effects of the environment on the matrix and the fibre/matrix bond determine the effects of the environment on the composite.

Storage at high temperatures

- only in composites with polyamide resins was a drop in the mechanical properties of the composite detected after storage at high temperatures (300°C).

Temperature cycles before loading

- temperature cycles in the range below room temperature reduce the strength and stiffness of fibre composites,
- constant tensile stress during alternating temperatures can cause even greater reductions in the strength and stiffness of fibre composites with plastic matrices.

High temperatures during loading

- at temperatures in the range 100-180°C during loading the tensile strength of 0° and 0°, $\pm 45^{\circ}$, 90° composites with epoxide resin matrix and stiff fibres (carbon, boron) drops by 20% maximum,
- in 90° composites with epoxide matrix high temperatures (up to 180°C) during loading cause a drop in tensile strength up to 60% maximum and in stiffness 50% maximum,
- in fibre-dominant composites the adverse effect of high temperatures on static strength and stiffness is greater in GFC and Kevlar than in BFC and CFC.

Effect of absorbed moisture

- in 0^{o} and 0^{o}, $\pm 45^{o}$, 90^{o} composites tensile strength drops by 20% maximum with moisture levels over 1.0%, stiffness is not affected,
- in 90^{o} composites tensile strength and stiffness drop with increasing moisture content; at 2% moisture level the drop is 90%,
- the effect of moisture is greater if the proportion of shear stress on the fibre composite is large.

Creep strength and creep behaviour

- in matrix-dominant composites creep strains are approximately the same as those in the non-reinforced matrix materials,
- in fibre-dominant CFC creep strains are less than in aluminium alloys (2024, 7075).

Scatter of test results

- the scatter of strength characteristics is at present still greater for fibre composites than for metal materials; this difference will disappear with the development of quality controls for fibre composite methods of construction,
- the scatter of creep strengths of fibre composites is 20-100 times greater than the scatter of tensile strengths,
- particular attention should be paid to the scatter of mechanical properties of fibre composites in the present stage of development.

1.6 References to the literature

In this section the references evaluated in the preceding sections are classified according to contents and divided by subsidiary subjects. The references cited in each section are underlined.

Section 1.2.1

Stress-strain behaviour:

<u>23</u>, <u>25</u>, 77, <u>79</u>, <u>105</u>, 114, <u>117</u>, 120, <u>191</u>, <u>218</u>, <u>252</u>, <u>313</u>, <u>414</u>, <u>486</u>, 634, <u>719</u>, 792.

Section 1.2.2

Notched bars:

44, 63, 96, 166, 174, 176, 252, 253, 343, 377, 414, 420, 591, 620, 622, 634, 763, 766, 770.

Section 1.2.3

Joints:

547.

Bondings:

23, 29, 46, 152, 161, 521, 549, 652, 653, 657.

Mechanical bond:

44, 222, 545, 546, 548.

Section 1.2.4

Percussive stress:

59, 60, 100, 146, 556, 642, 643, 644, 721.

Section 1.3.1

Environmental effect:

632, 648.

Parameters:

268, 271, 356, 426, 528, 718, 720.

Residual stresses:

218, 515, 518, 566, 569, 638, 639, 647.

Glass equilibrium temperature:

92, 343, 528, 670.

Matrix material:

91, 92, 214, 218, 271, 356, 396, 426, 514, 515, 553, 667.

Section 1.3.2

Storage:

436, 516, 538, 667.

Section 1.3.3

Temperature cycles:

48, 174, 218, 514, 516, 647.

Section 1.3.4

Temperature during loading:

2, 3, 33, 54, 92, 110, 119, 161, 251, 316, 321, 361, 383, 396, 416, 426, 436, 516, 600, 648, 667, 718, 195.

Section 1.3.5

Moisture absorbed:

92, 172, 214, 271, 309, 321, 322, 327, 340, 356, 369, 515, 516, 518, 528, 553, 565, 573, 614, 648, 667, 718, 720, 722, 566.

Section 1.3.6

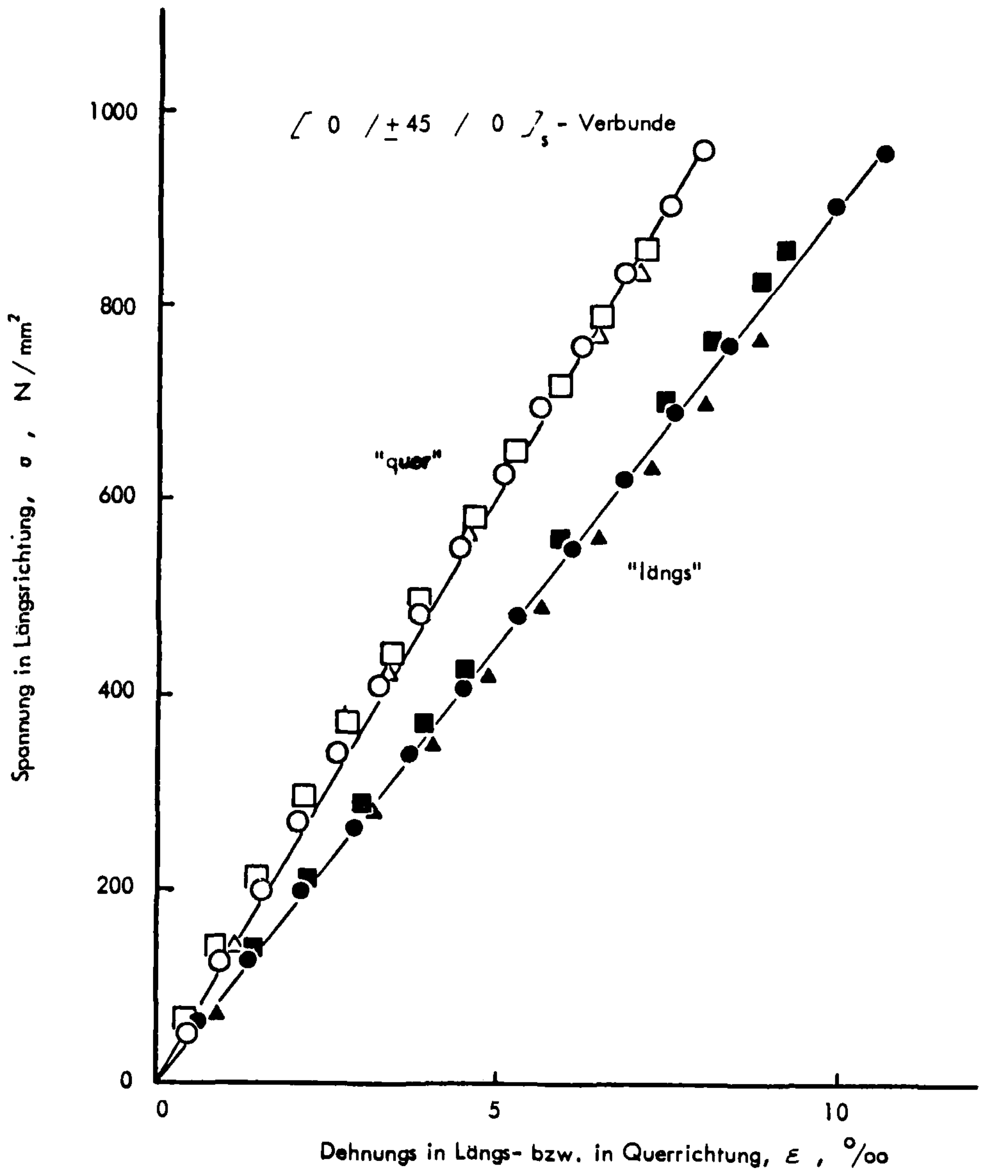
Creep strength and creep behaviour:

21, 87, 100, 173, 181, 246, 320, 398, 425, 455, 516, 535, 585, 657, 658, 666, 667, 716, 720, 776.

Section 1.4

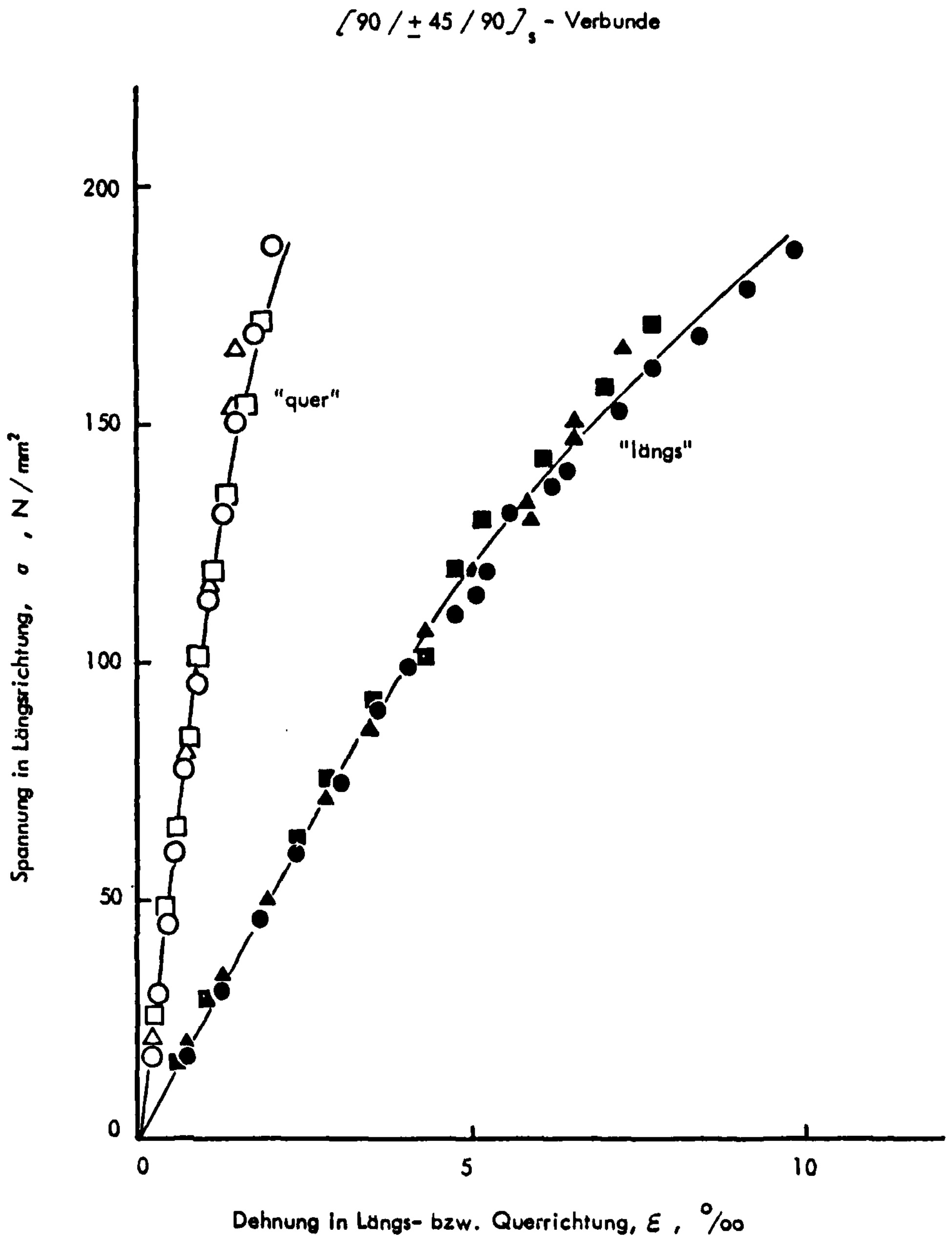
Scatter:

50, 113, 119, 157, 158, 159, 161, 179, 191, 221, 254, 276, 343, 399, 401, 414, 473, 484, 518, 634, 643, 668, 720, 778, 784.



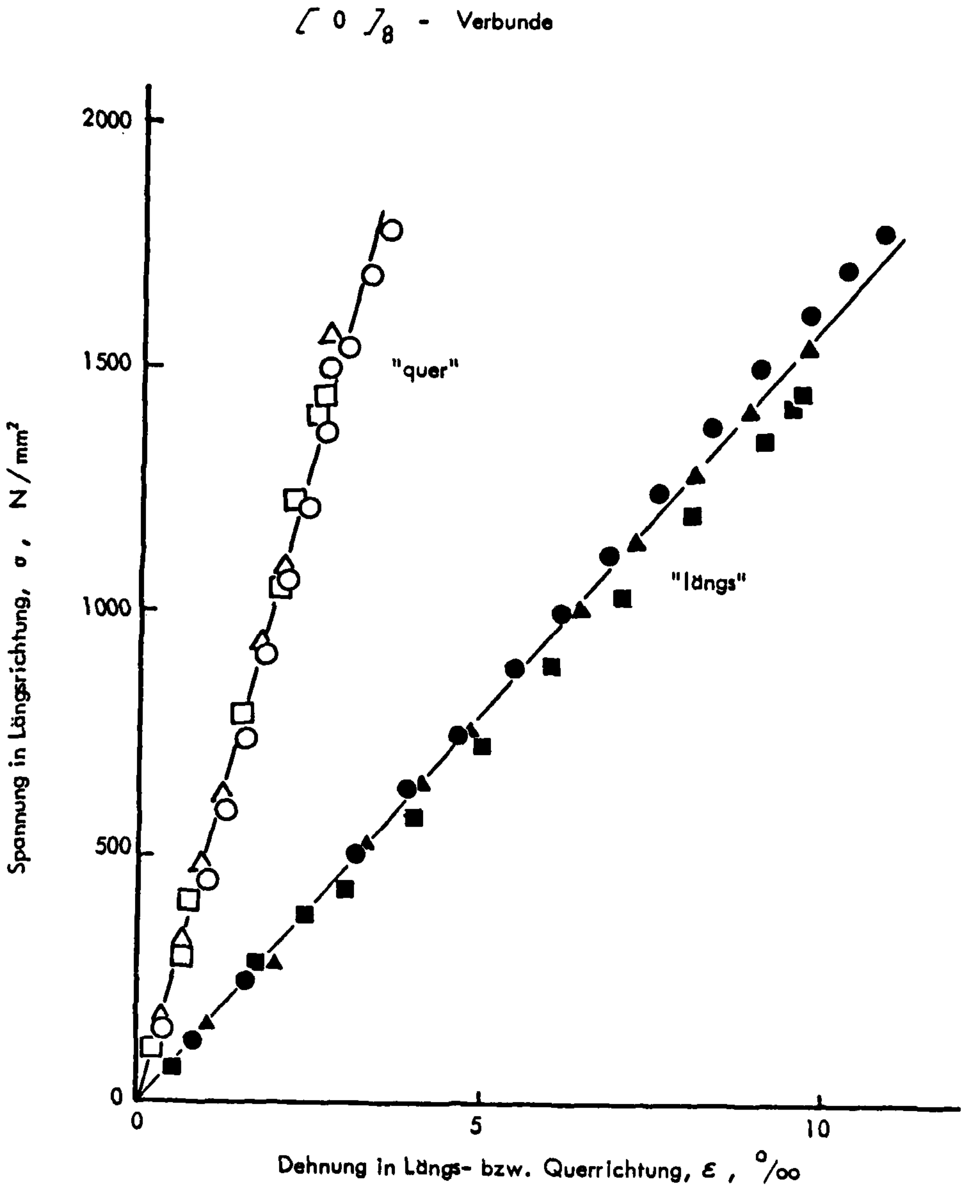
Key:
Spannung in Längsrichtung = stress in longitudinal direction
Dehnungs in Längs-bz·/ in Querrichtung = strain in longitudinal and transverse direction

Fig 1.1 Stress-strain behaviour of a fibre-dominant angle ply CFC (Ref 191)



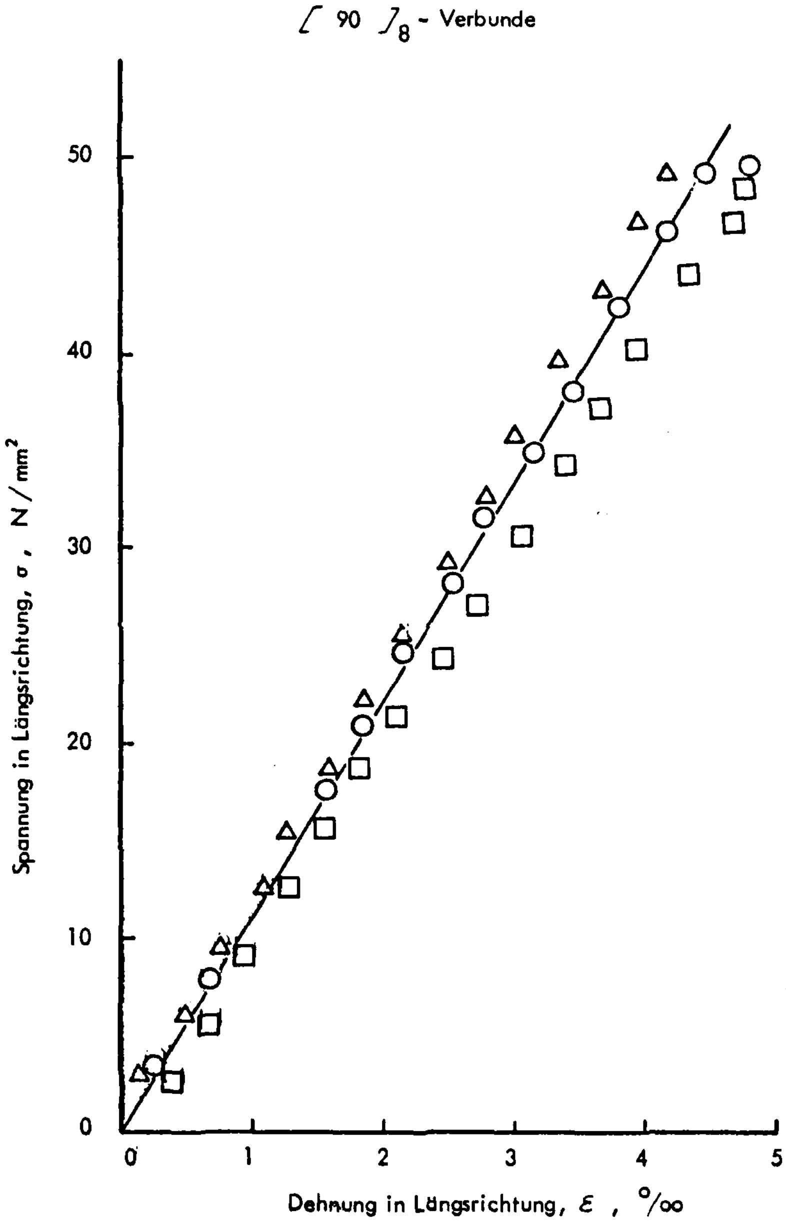
Key:
See Fig 1.1

Fig 1.2 Stress-strain behaviour of a matrix-dominant angle ply CFC (Ref 191)



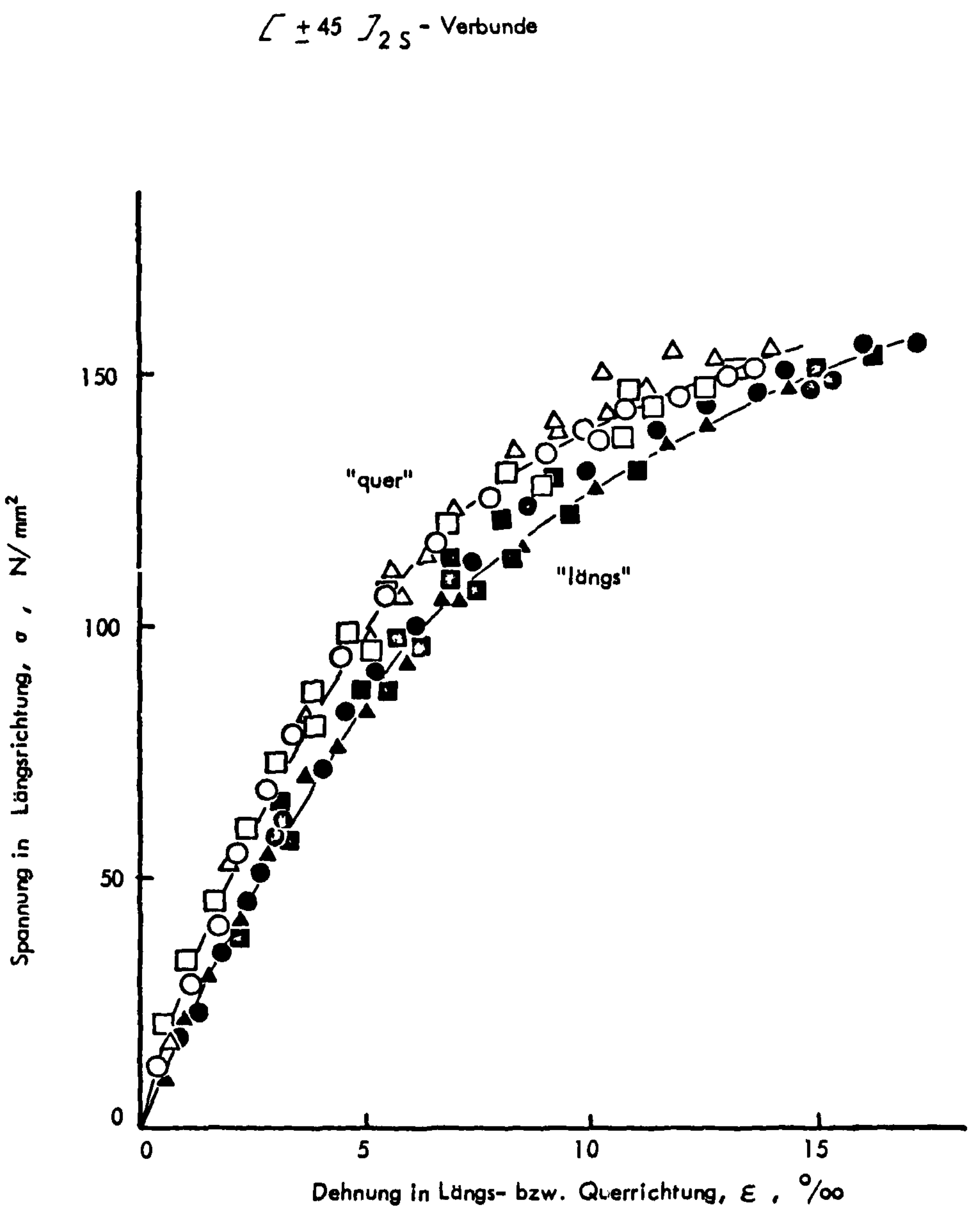
Key:
See Fig 1.1

Fig 1.3 Stress-strain behaviour of a uni-directional CFC loaded parallel to the fibre (Ref 191)



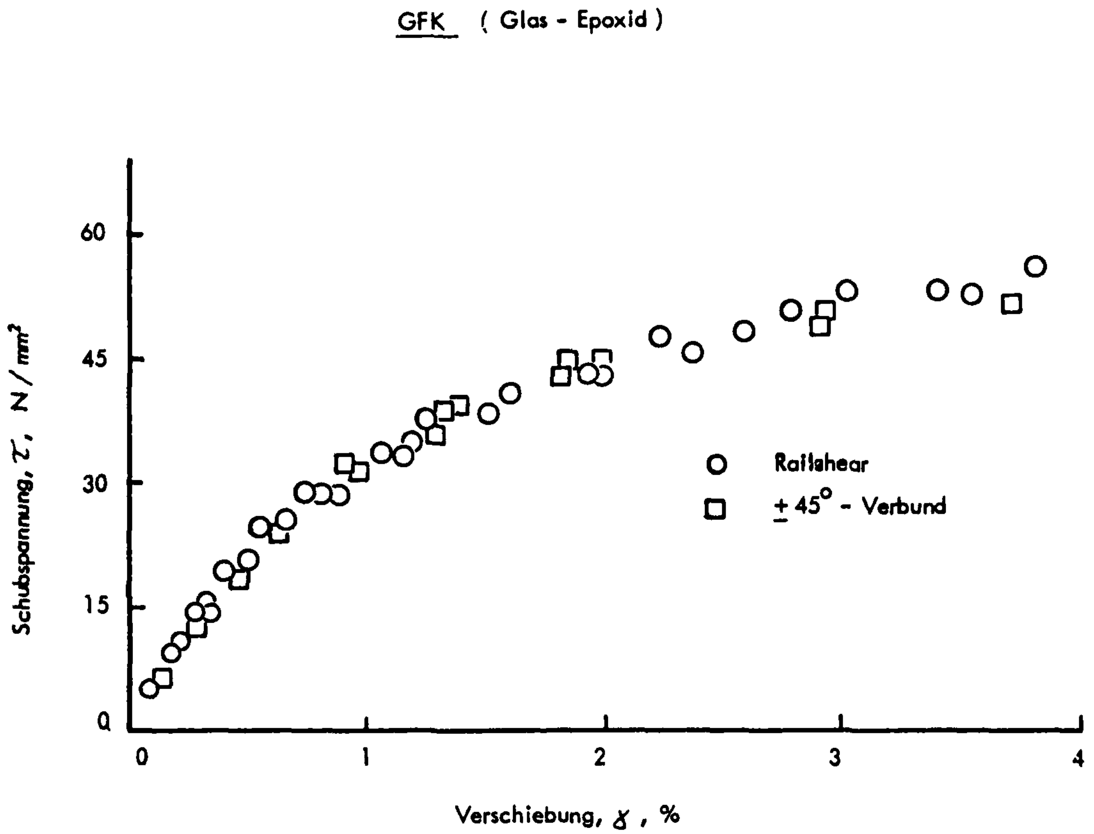
Key:
See Fig 1.1

Fig 1.4 Stress-strain behaviour of a uni-directional composite loaded across the fibre (Ref 191)



Key:
See Fig 1.1

Fig 1.5 Stress-strain behaviour of a ±45° CFC (Ref 191)



Key:
Schubspannung = shear stress
Verschiebung = shift

Fig 1.6 Stress-strain behaviour of GFC under intralaminar shear load (Ref 79)

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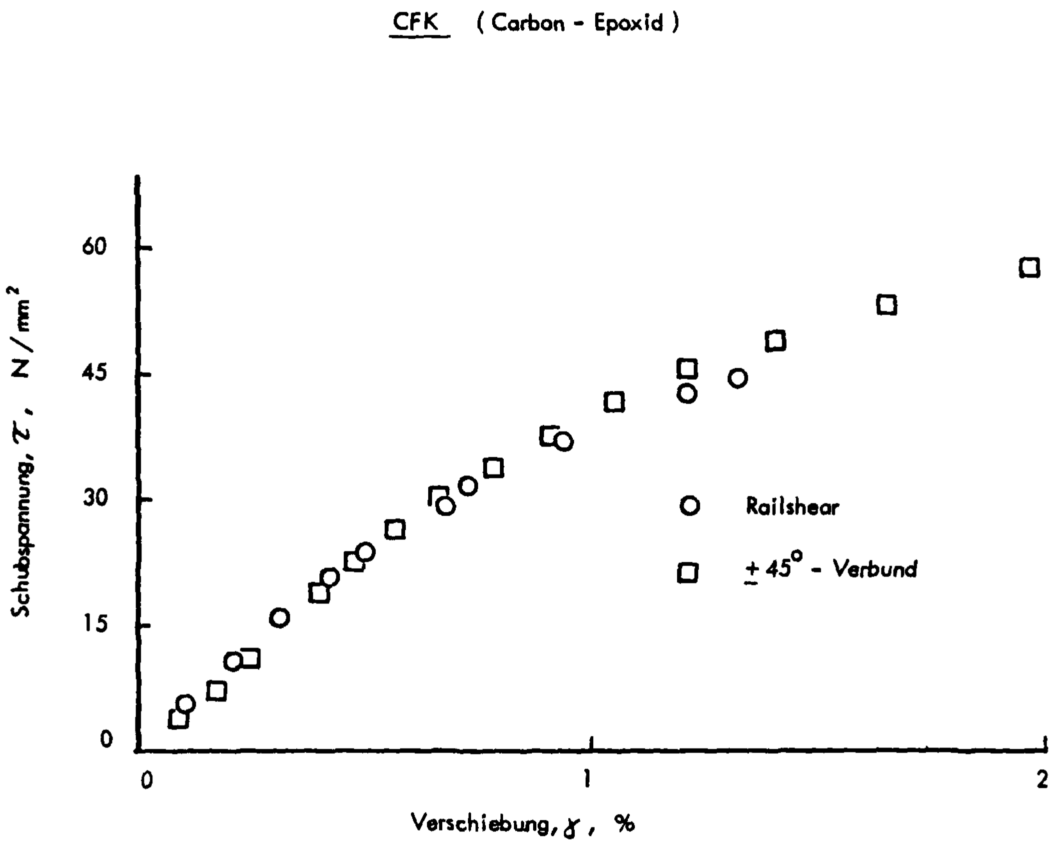
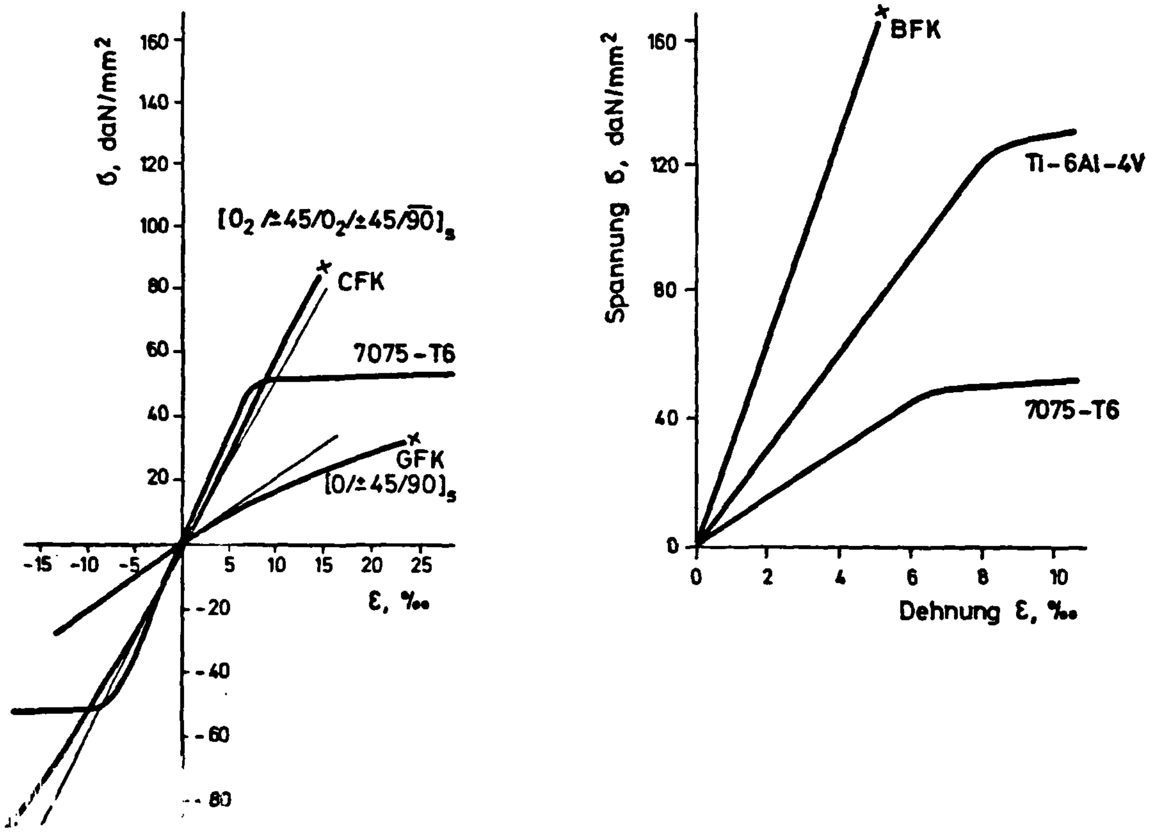


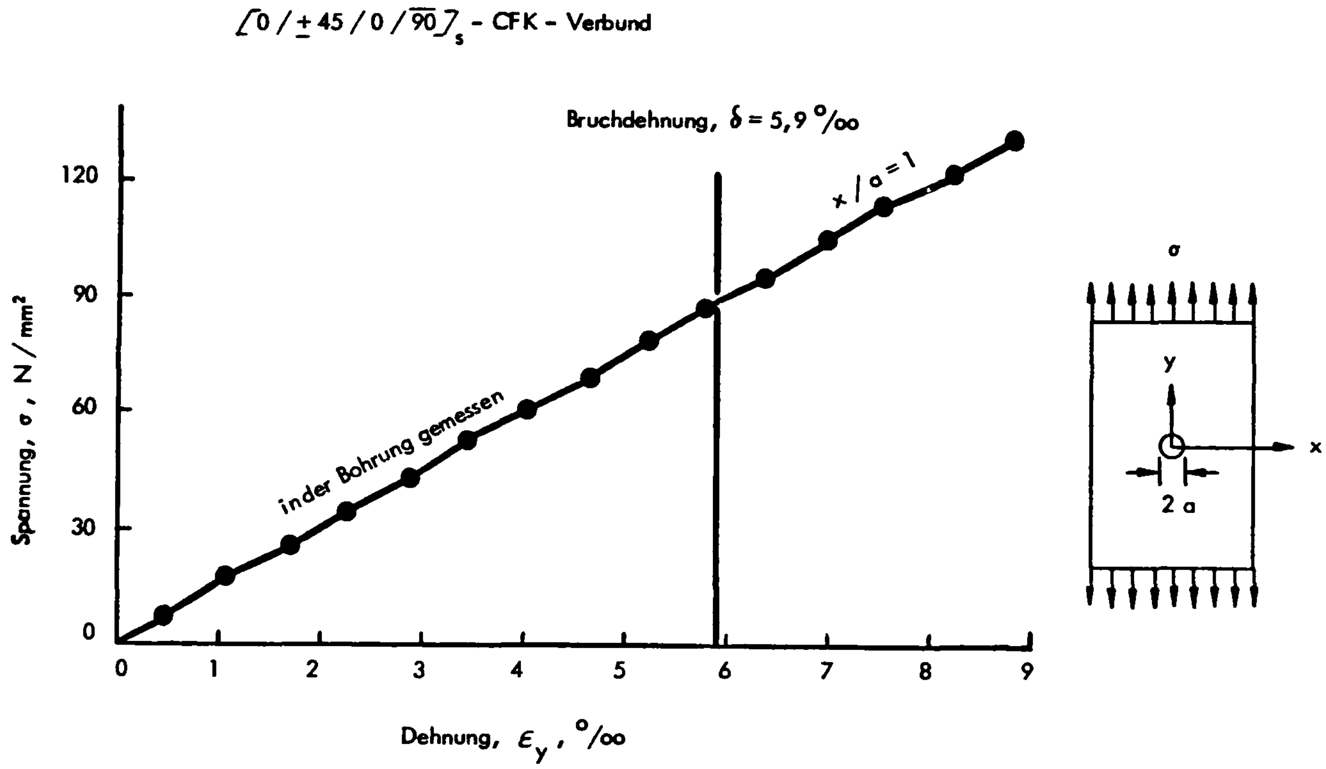
Key:
See Fig 1.6

Fig 1.7 Stress-strain behaviour of CFC under intralaminar shear load (Ref 79)



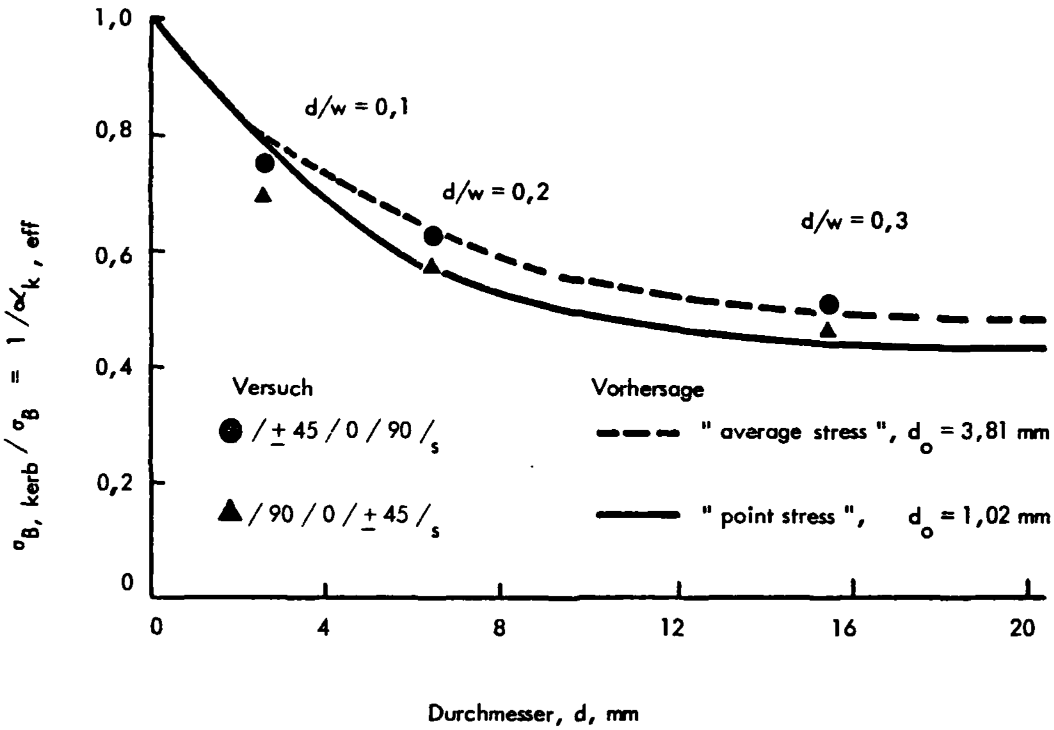
Key:
Spannung = stress
Dehnung = strain

Fig 1.8 Stress-strain behaviour of BFC, CFC, GFC, aluminium 7075 T6 and titanium Ti 64



Key:
In der Bohrung gemessen = measured in the hole

Fig 1.9 Comparison between strain in the hole and breaking strain in an angle ply CFC (Ref 252)

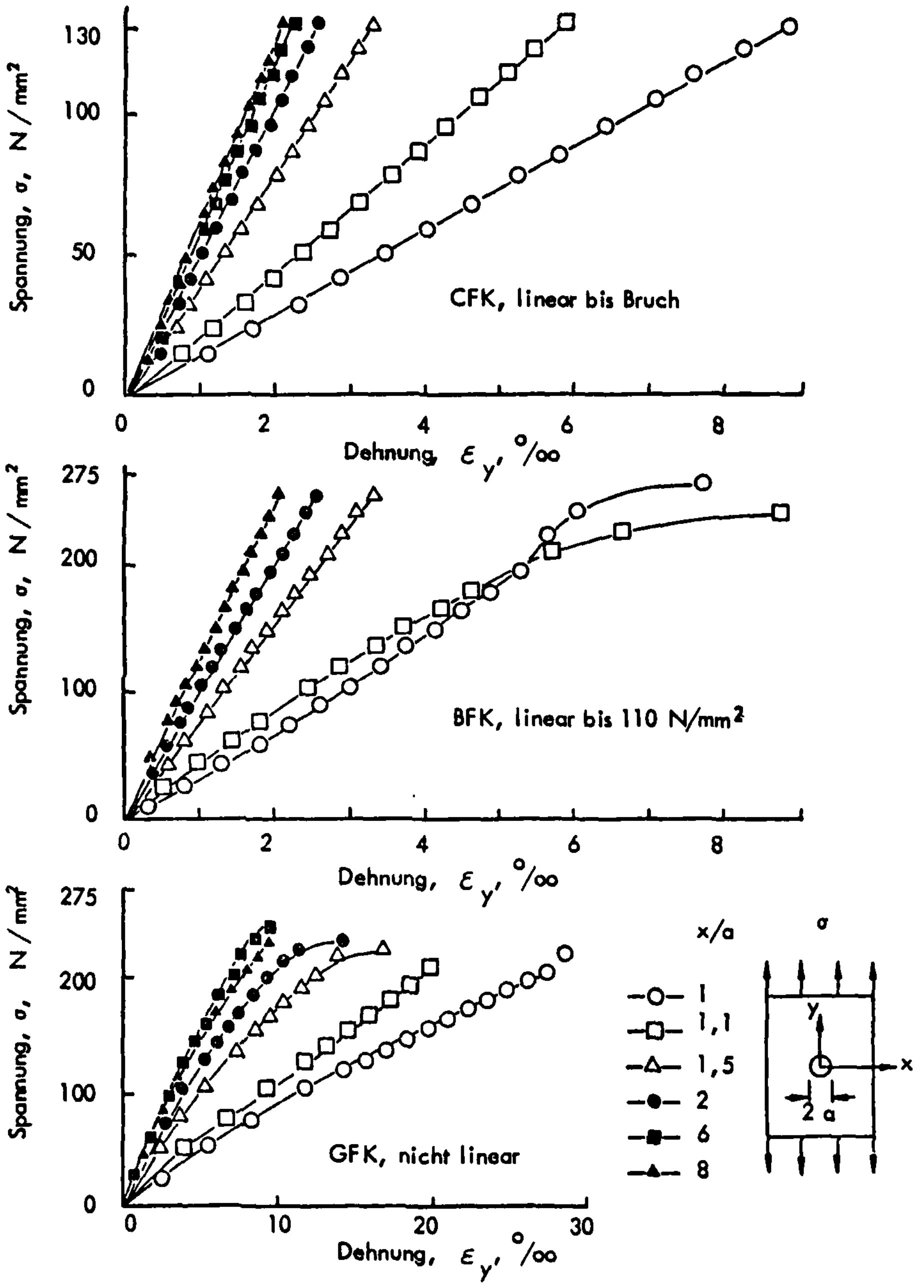


[591]

Key:
Versuch = test
Vorhersage = prediction
Durchmesser = diameter

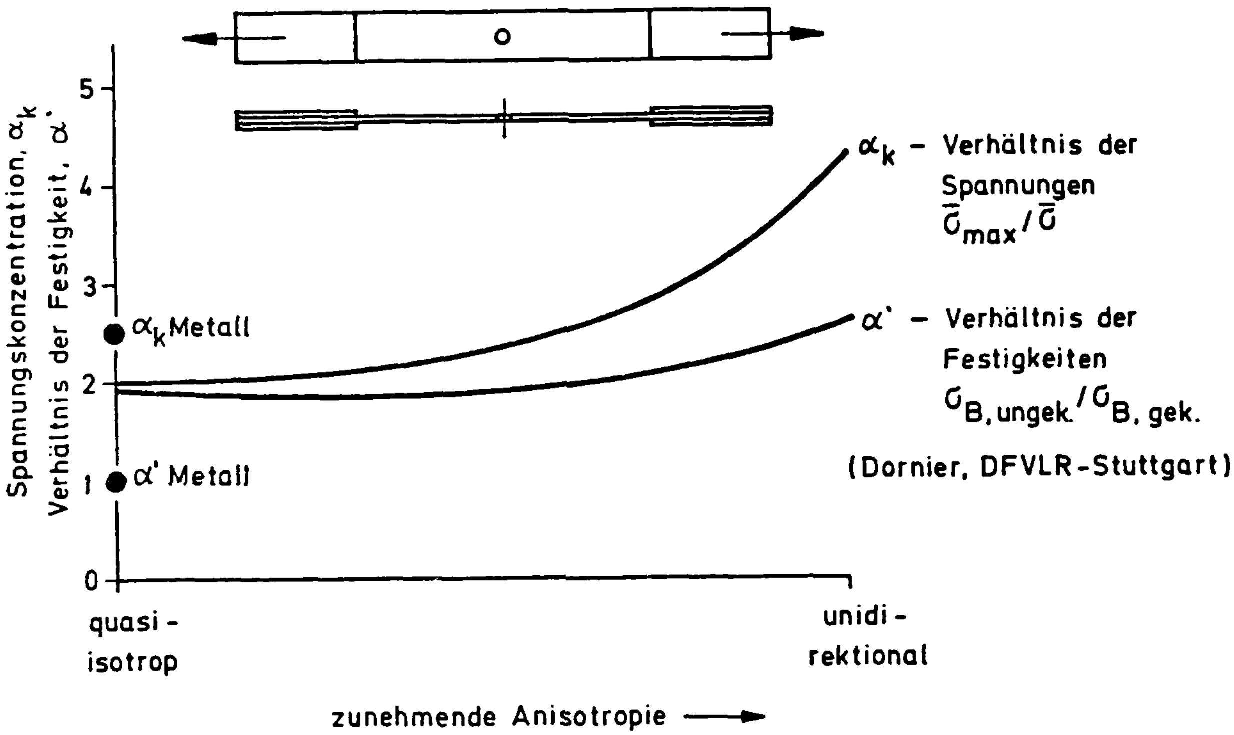
Fig 1.10 Effect of the hole diameter on tensile strength of CFC (Ref 634)

LT 2045



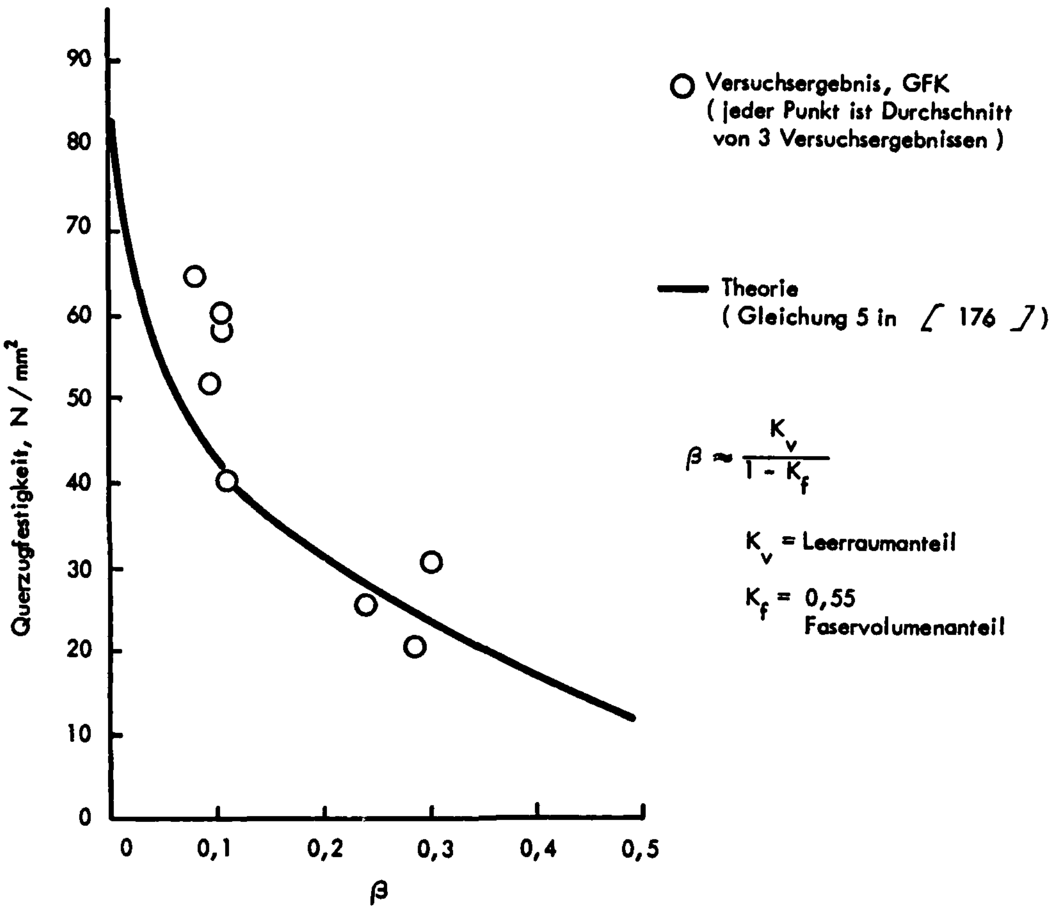
Key:
Linear bis Bruch = linear to fracture
Nicht linear = non-linear

Fig 1.11 Effect of fibre material on the stress-strain behaviour in the hole cross-section of angle ply composite (Ref 252)



Key:
Verhältnis der Spannungen/Festigkeiten = ratio of stresses/strengths
Zunehmende Anisotropie . = increasing anisotropy

Fig 1.12 Dependence of stress concentration and static strength of notched specimens on the laminate structure

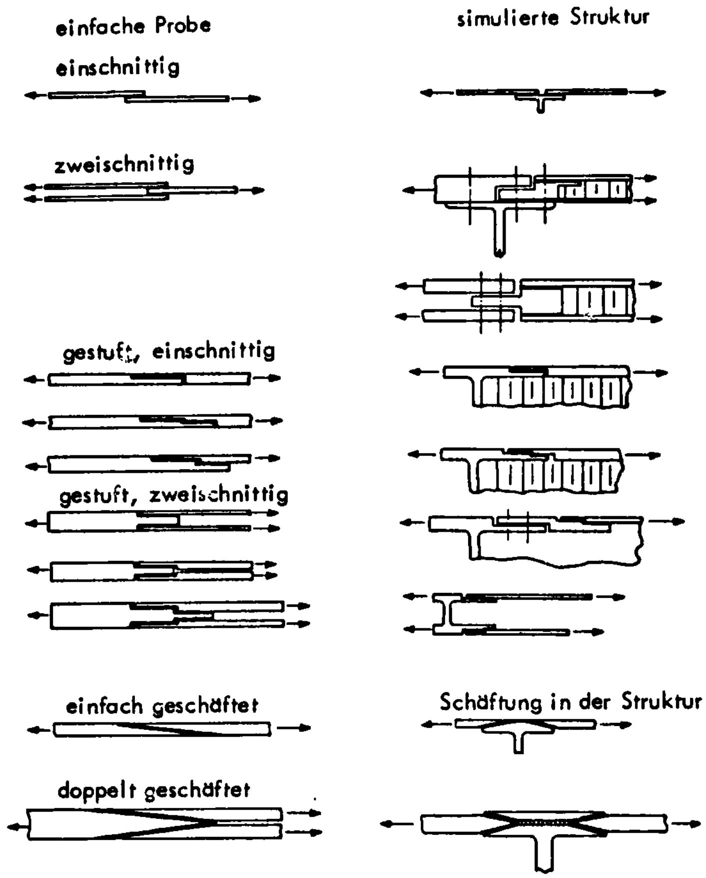


Key:

Versuchsergebnis = result of test

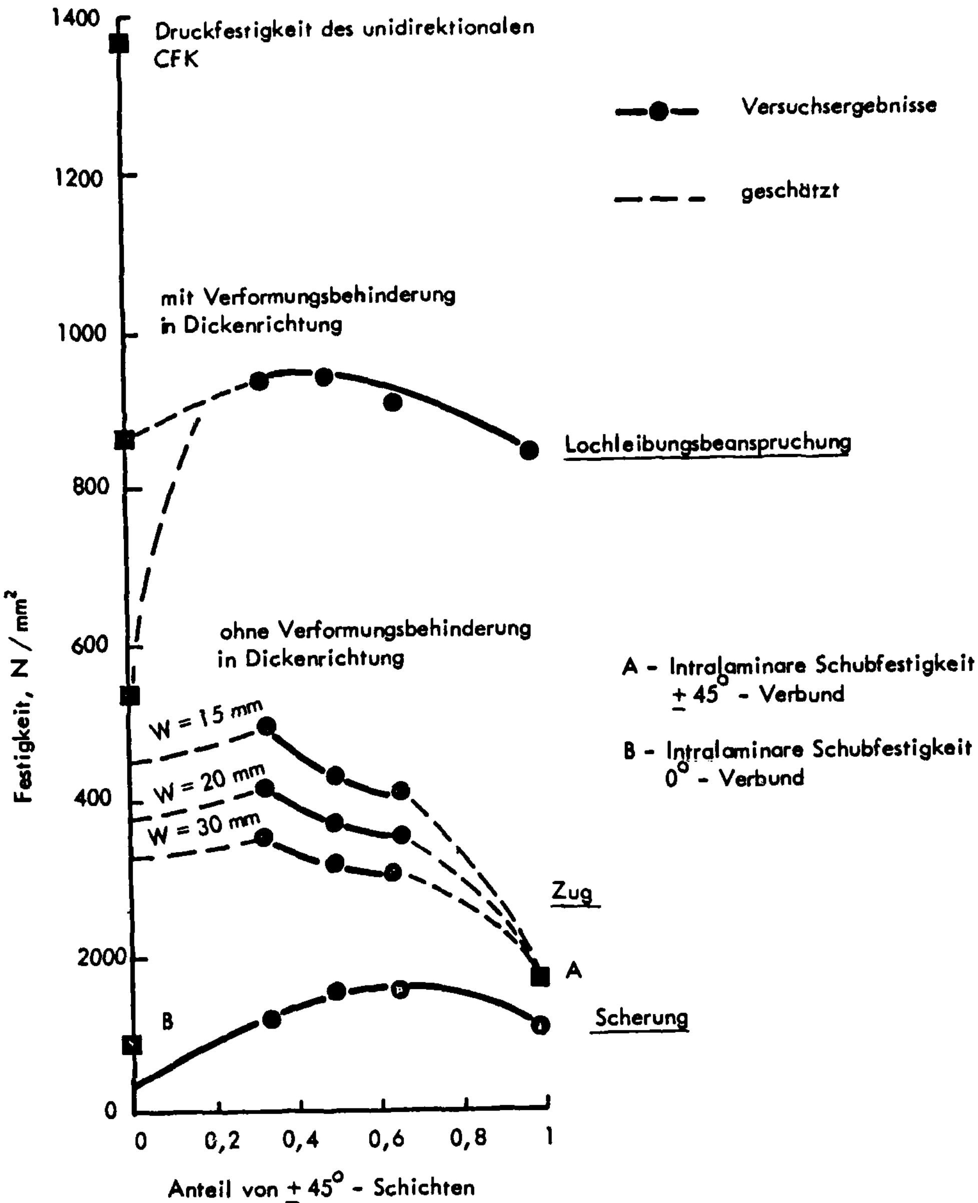
Jeder Punkt ist Durchschnitt von 3 Versuchsergebnissen = each point is the average of 3 test results

Fig 1.13 The effect of void content on transverse tensile strength (Ref 176)



Key:
Einfache Probe = simple specimen
Einschnittig = single shear
Zweischnittig = double shear
Gestuft = stepped
Geschäftet = spliced
Simulierte Struktur = simulated structure
Schäftung = splice

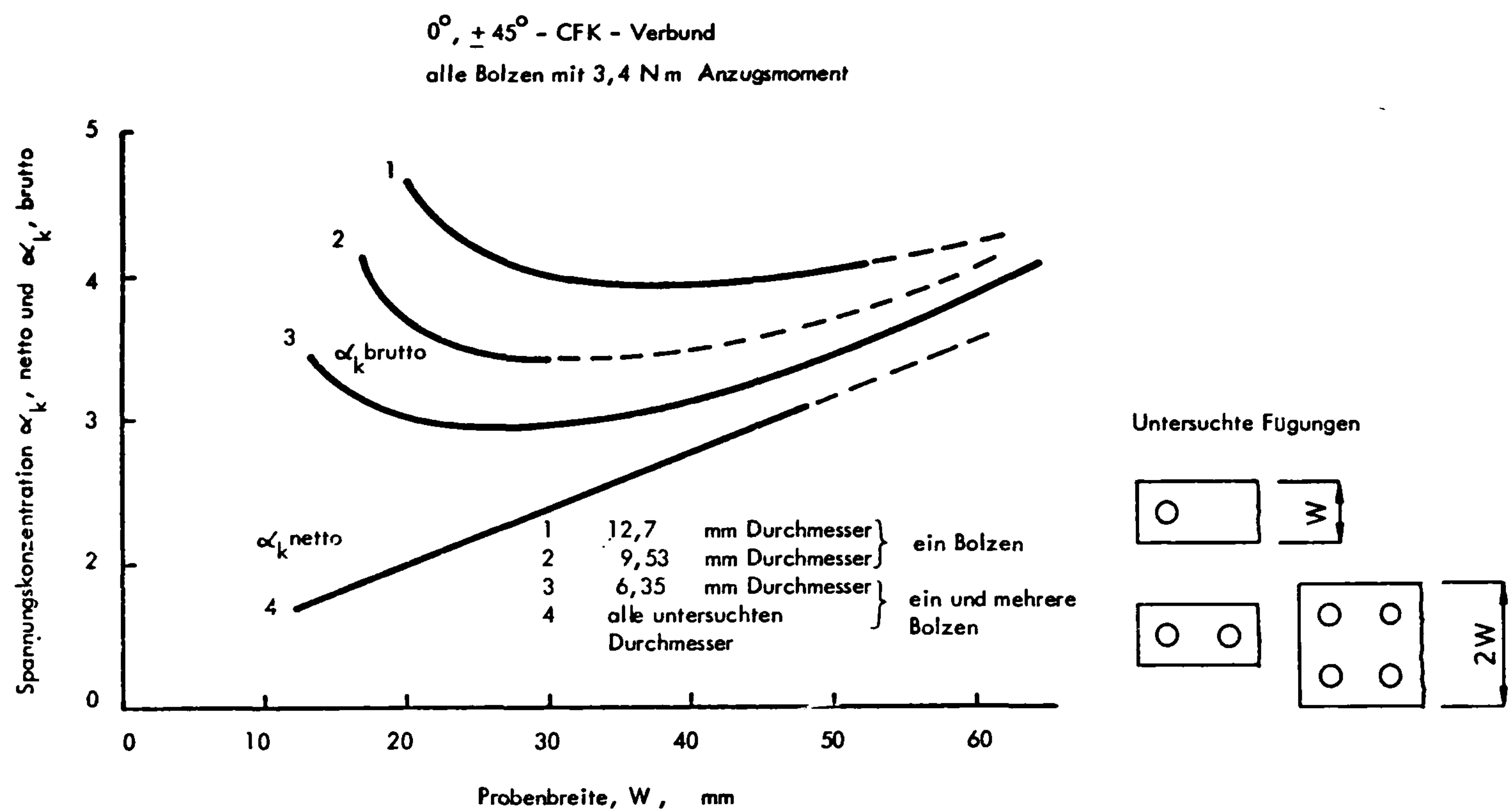
Fig 1.14 Joint configurations



Key:

Druckfestigkeit	= compressive strength
Mit Verformungsbehinderung in Dickenrichtung	= with deformation resistance in direction of thickness
Ohne ...	= without ...
Geschätzt	= estimated
Lochleibungsbeanspruchung	= bearing load
Intralaminare Schubfestigkeit	= *intralaminar shear strength*
Zug	= tension .
Scherung	= shear
Anteil von ±45° Schichten	= proportion of ±45° layers

Fig 1.15 Effect of proportion of ±45° layers on bearing, tensile and shear strength of joints (100% load transfer) of 0°, ±45° CFC (Ref 545)



Key:
Alle Bolzen mit 3,4 nm Anzugsmoment = all bolts with 3.4 nm tightening torque
Untersuchte Fugungen = joints tested
Alle untersuchten Durchmesser = all diameters tested
Ein und mehrere Bolzen = one and several bolts
Probenbreite = specimen thickness

Fig 1.16 Relationship between average stress concentration factor and width of joints (Ref 545)

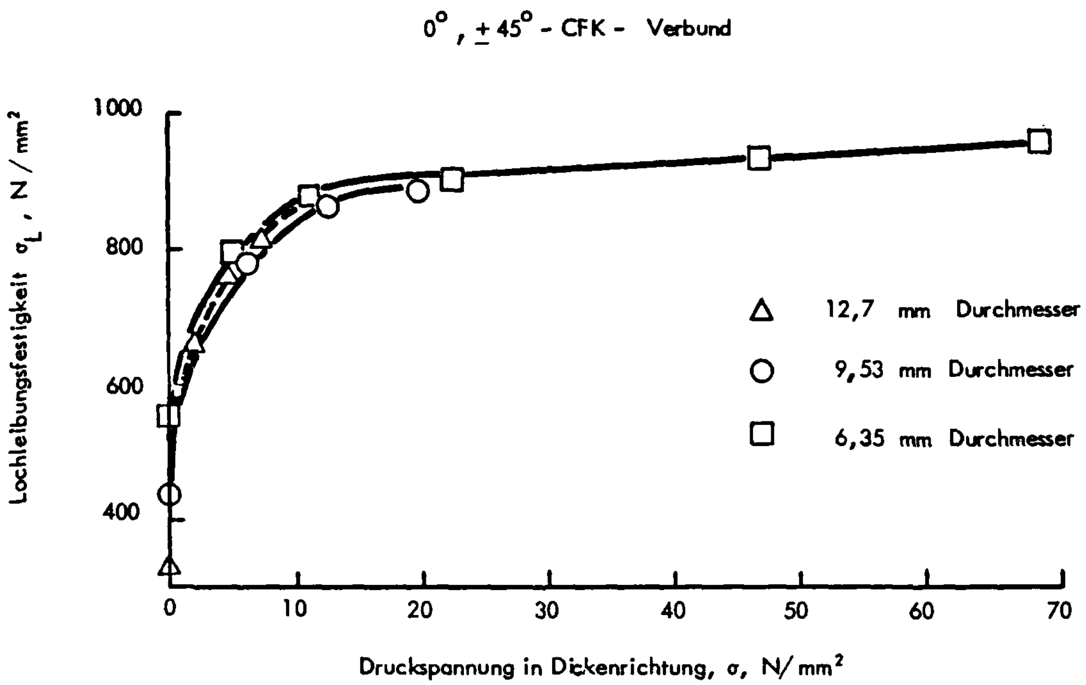


Fig 1.17 Effect of bolt clamping force (compressive stress in direction of thickness) on bearing strength (Ref 545)

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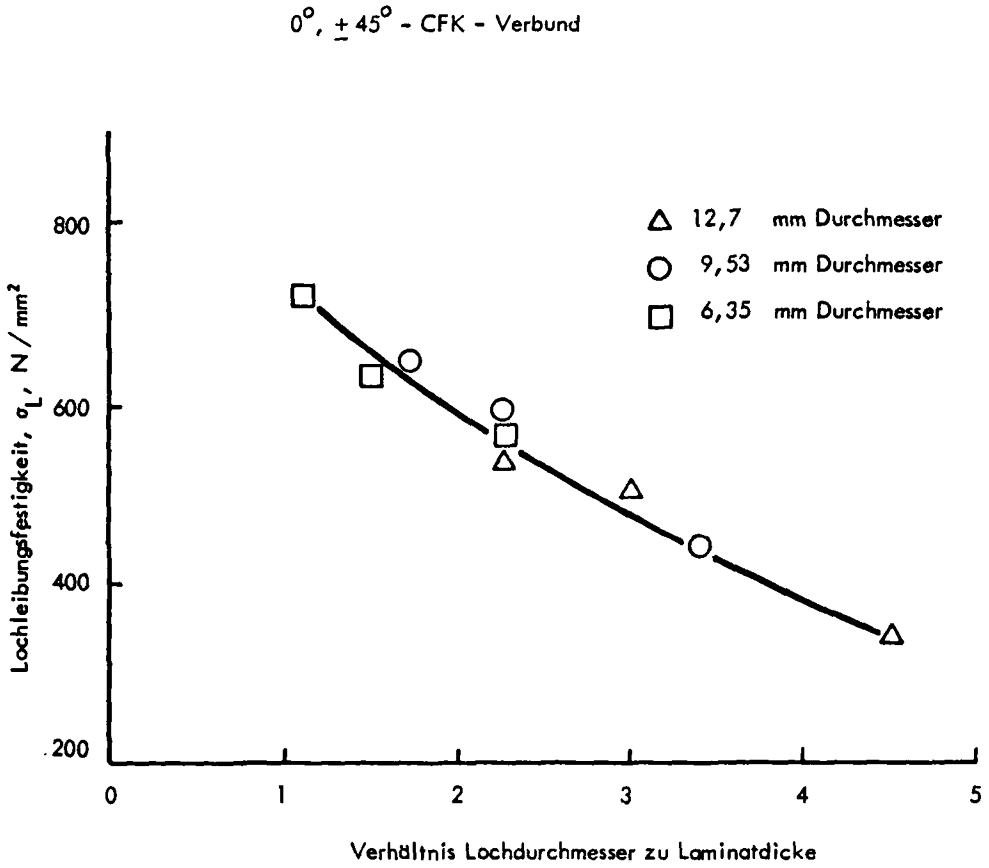
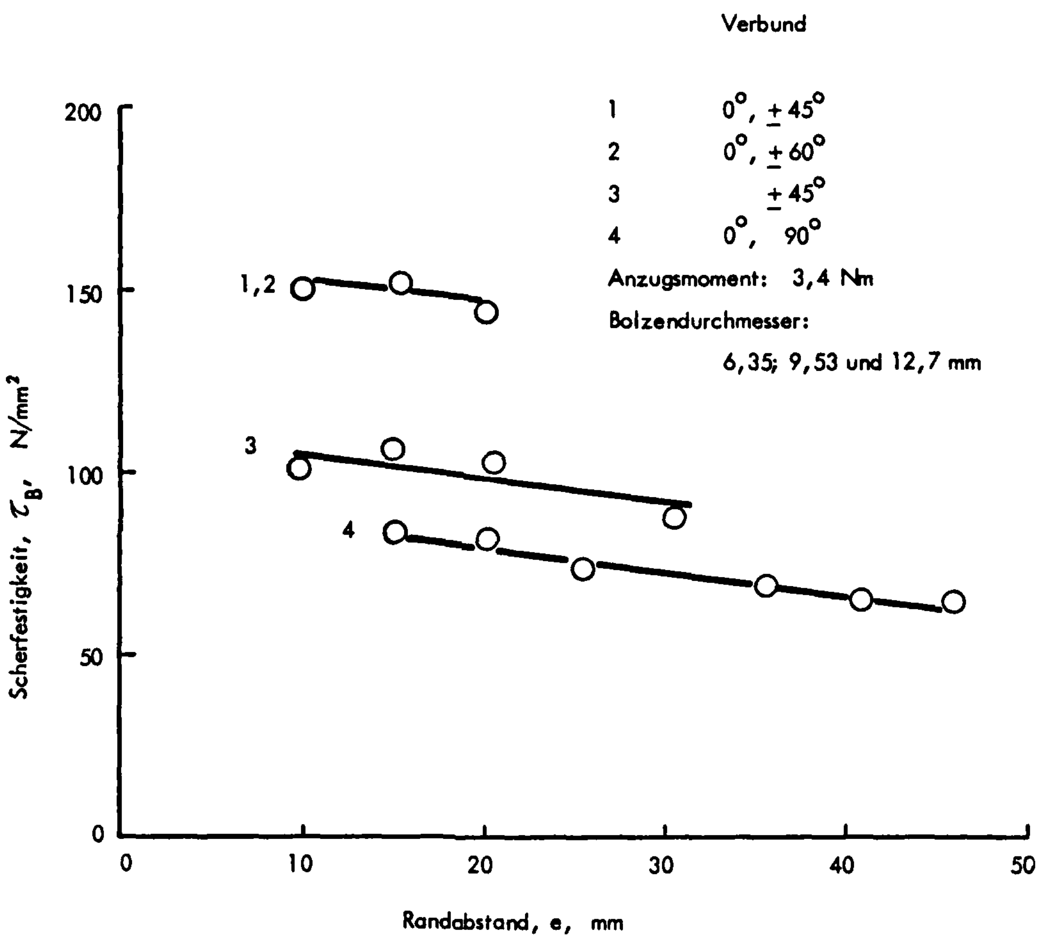


Key:
Verhältnis Lochdurchmesser zu Laminatdicke = ratio of hole diameter to thickness of laminate

Fig 1.18 Effect of laminate thickness on strength of hole without resistance to deformation in direction of thickness (Ref 545)



Key:
Randabstand = edge distance

Fig 1.19 Effect of fibre orientation in the composite on shear strength of joints (Ref 545)

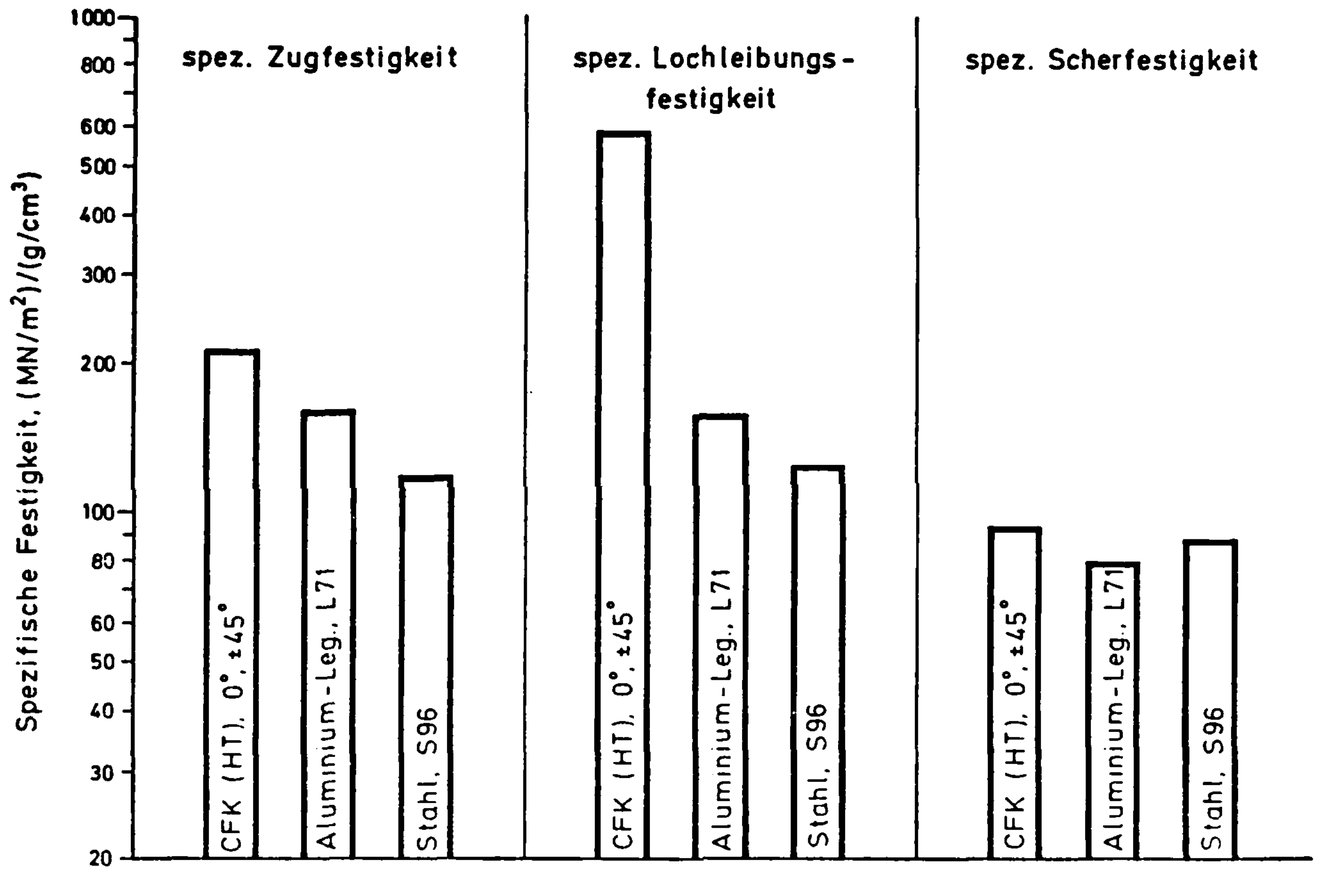
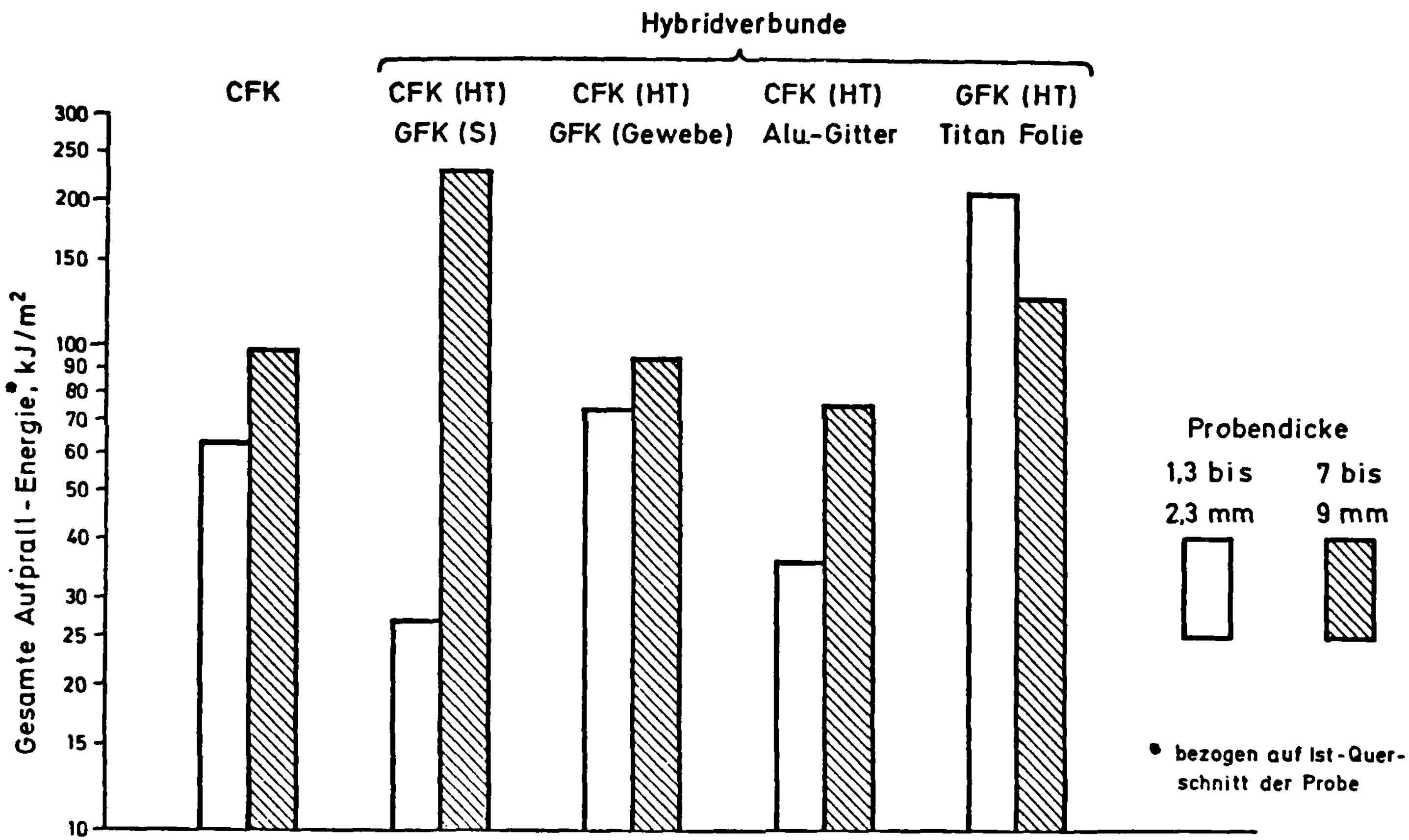


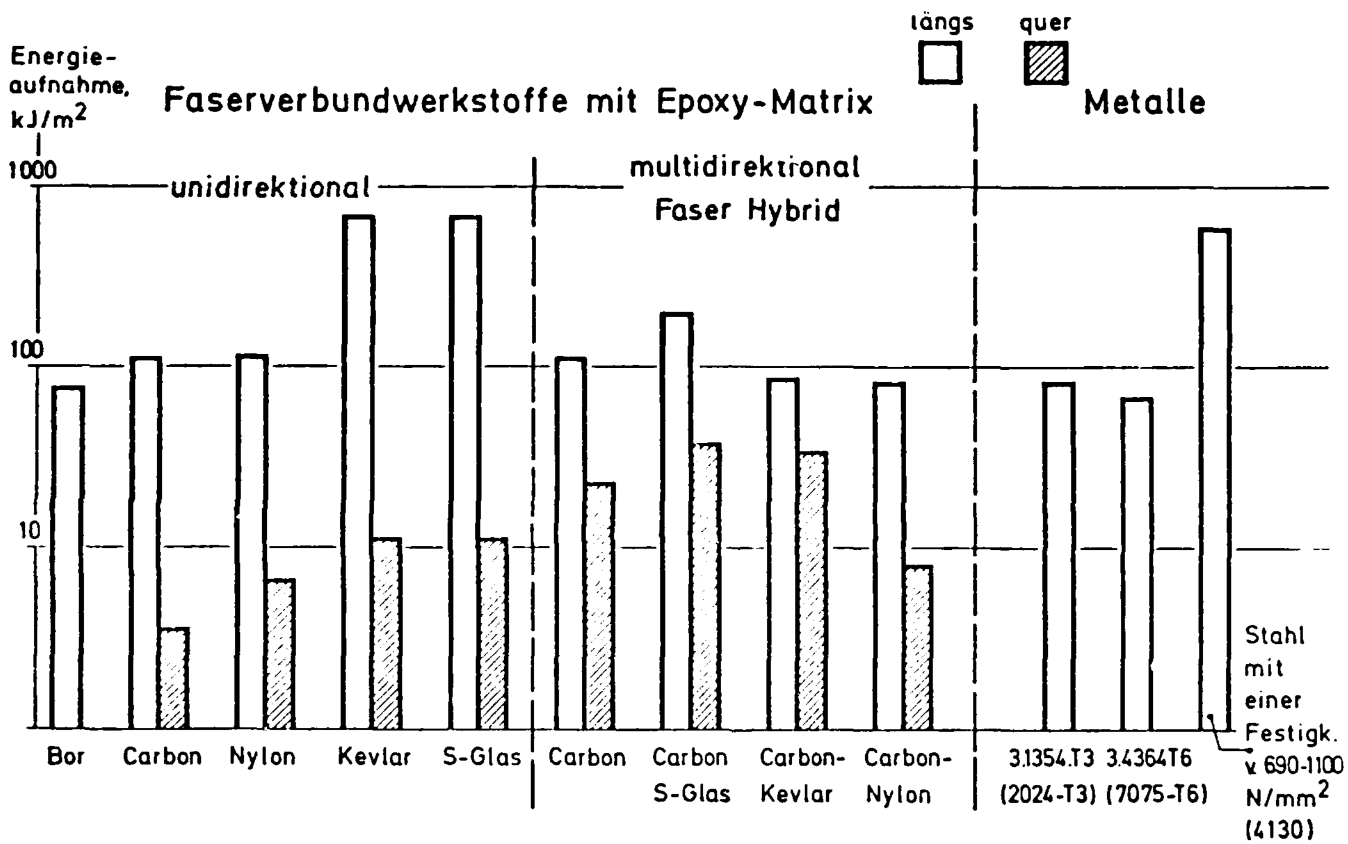
Fig 1.20 Specific tensile, bearing and shear strength of CFC (HTS) in comparison with metals (Ref 545)

[redacted] L[redacted]45



Key:
Gesamte Aufprall-Energie = total impact energy
Bezogen auf Ist-Querschnitt der Probe = related to actual cross-section of specimen

Fig 1.21 Effect of specimen thickness on the energy absorption of various composite materials (Ref 644)



Key:
Energieaufnahme = energy absorption
Längs = lengthwise
Quer = across

Fig 1.22 Notch impact strength (Charpy V) of fibre composite materials and metals (Ref 644)

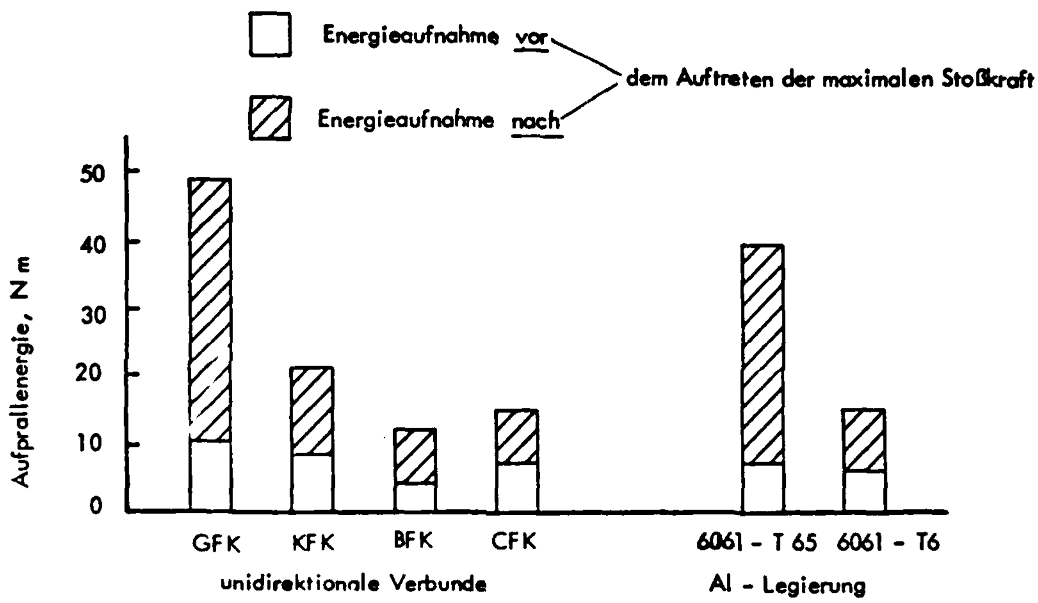


Fig 1.23 Energy absorbed before and after occurrence of maximum impact power (Ref 100)

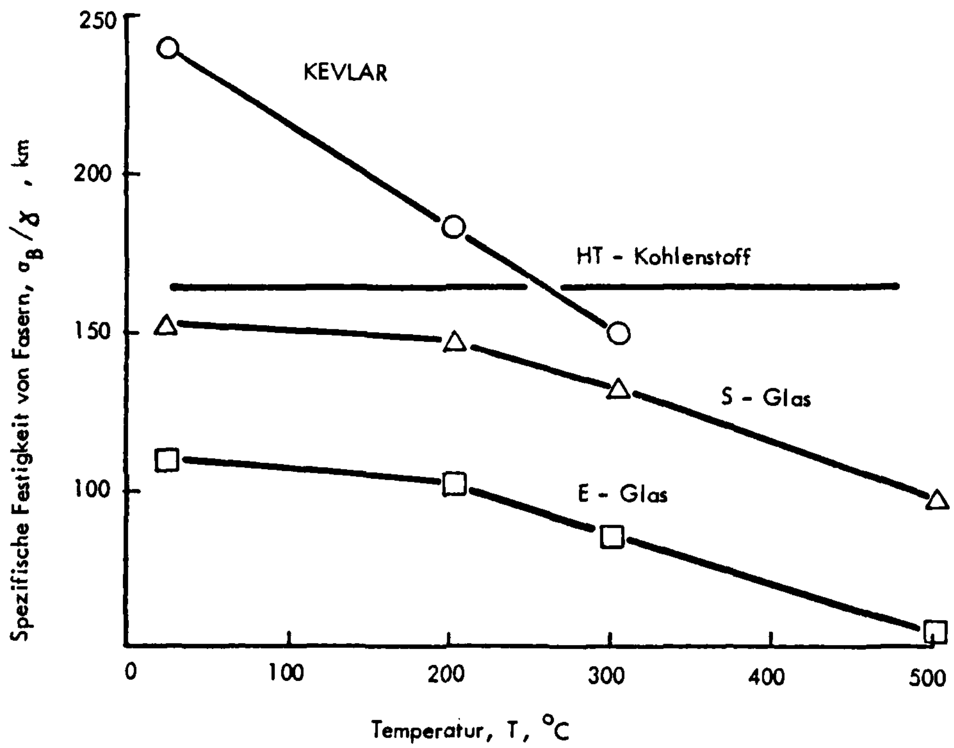


Fig 1.24 Specific tensile strength of fibres as a function of high temperatures (Ref 100)

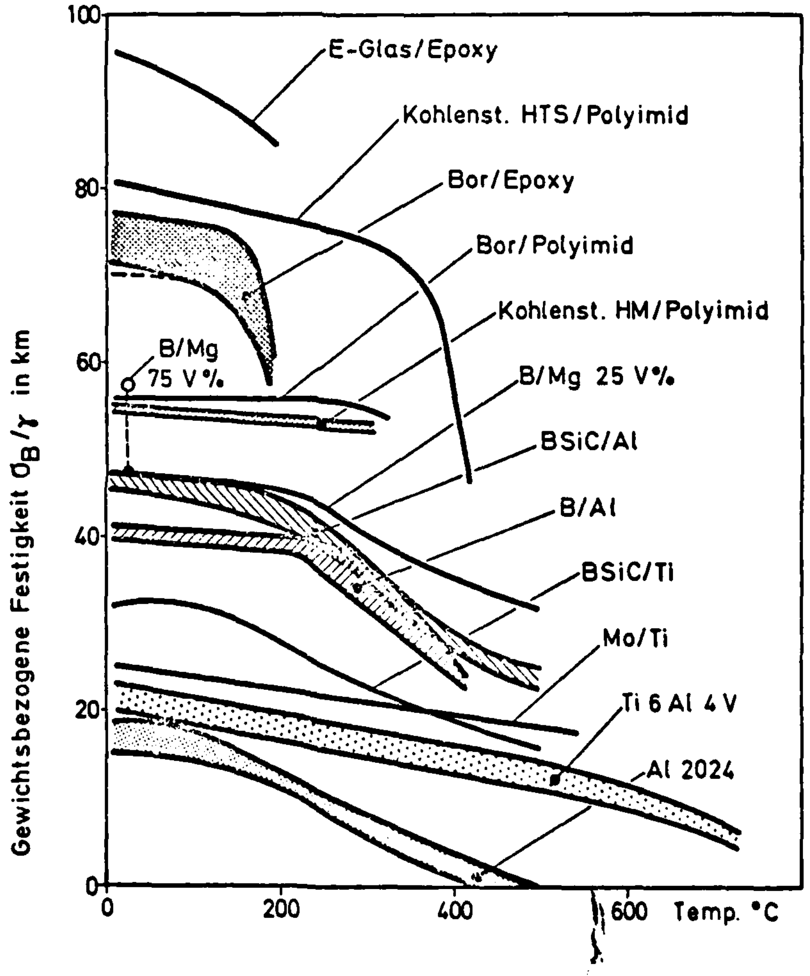
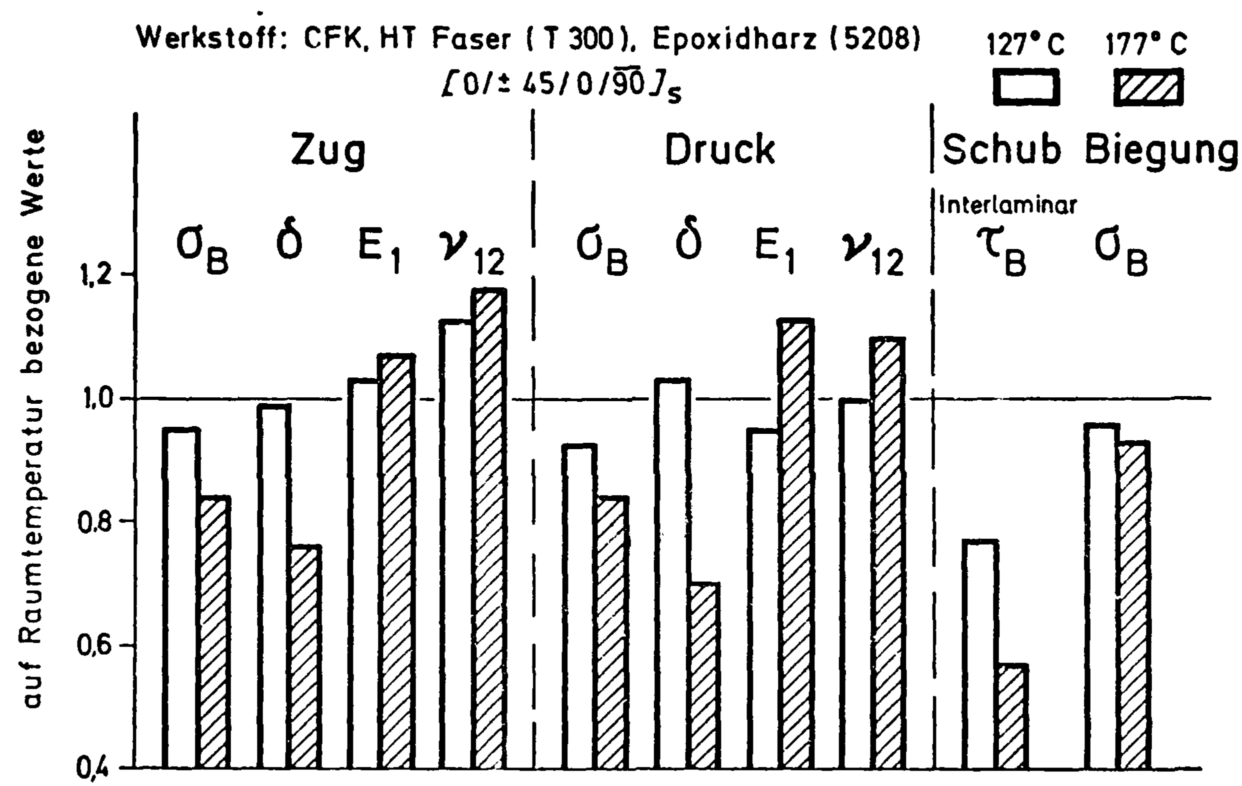
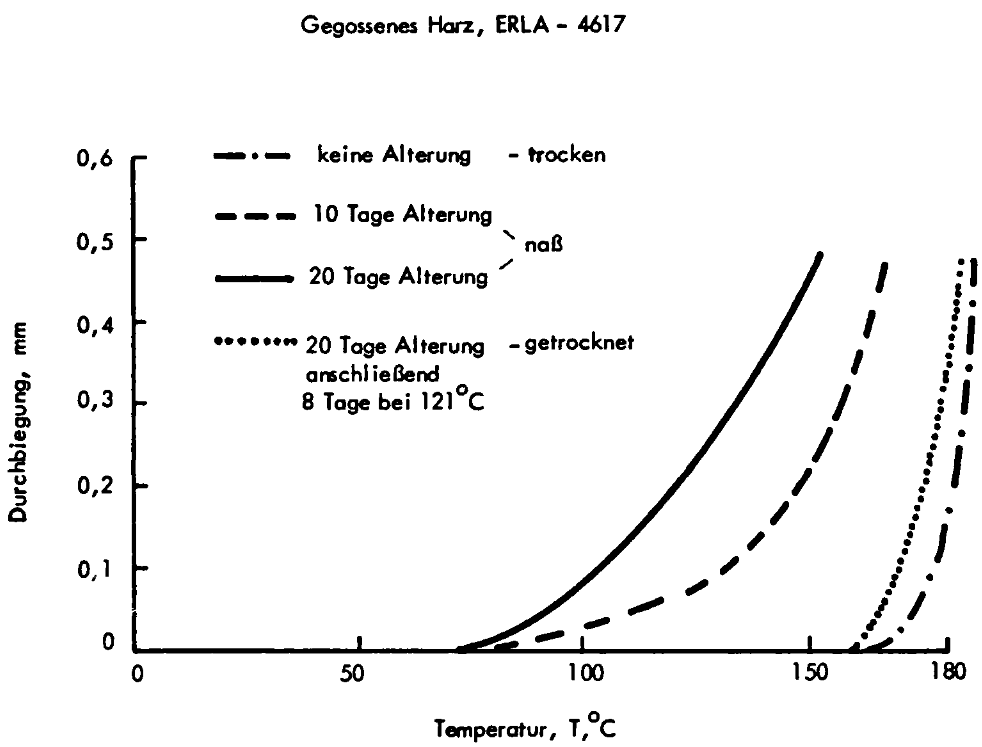


Fig 1.25 Specific strength of composite materials and metals as a function of temperature (Ref 426)



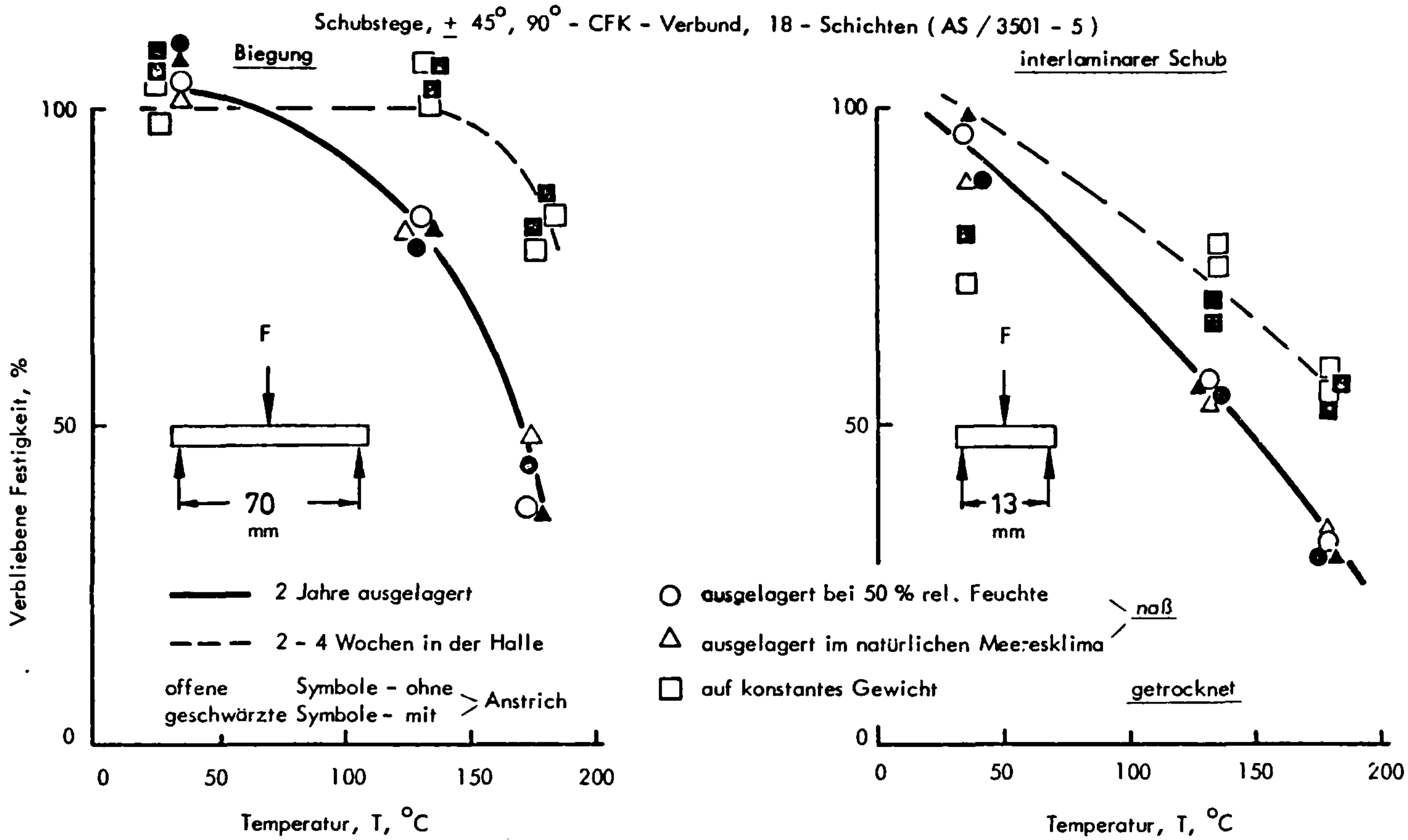
Key:
Auf Raumtemperatur bezogene Werte = values related to room temperature

Fig 1.26 Effect of high temperatures on the static properties of CFC (Ref 516)



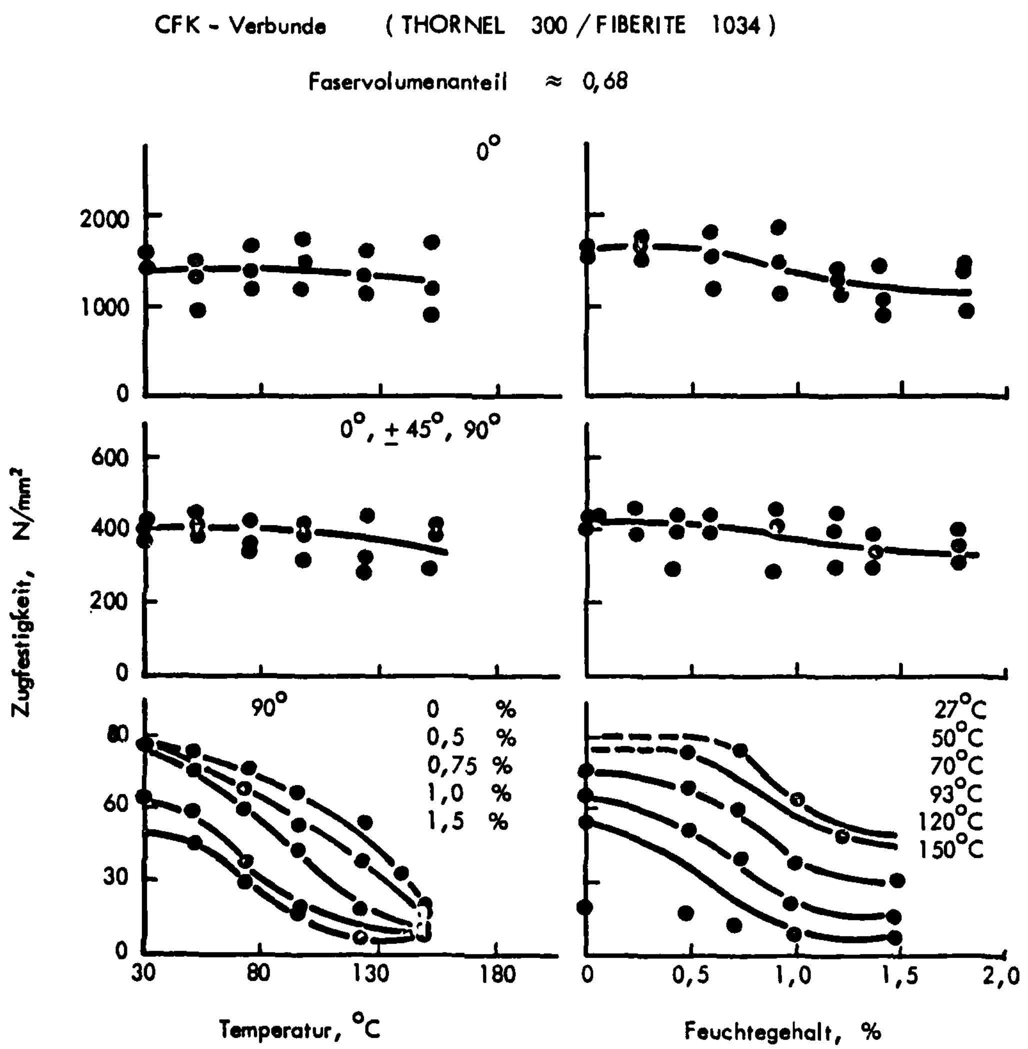
Key:
Gegossenes Harz = cast resin
Keine Alterung = no ageing
Trocken = dry
Nass = wet
Getrocknet = dried
Anschliessend = subsequent

Fig 1.27 Bending as a function of high temperatures for dry, wet and dried resin (Ref 92)



Key:
Verbliebene Festigkeit = remaining strength
2 Jahre ausgelagert = stored 2 years
2-4 Wochen in der Halle = 2-4 weeks in the shed
Offene Symbole = open symbols
Geschwärtze Symbole = filled in symbols
Anstrich = coat of paint, etc
Ausgelagert bei 50% rel. Feuchte = stored at 50% rel. humidity
Ausgelagert im natÜrlichen Meeresklima = stored in natural marine olimate
Auf konstantes Gewicht getrocknet = dried to constant weight

Fig 1.28 Bending and shear strength as a function of high temperatures for wet and dried CFC (Ref 528)



Key:
Faservolumenanteil = fibre volume fraction

Fig 1.29 Effect of temperature (during loading) and moisture content on tensile strength of CFC (Ref 718)

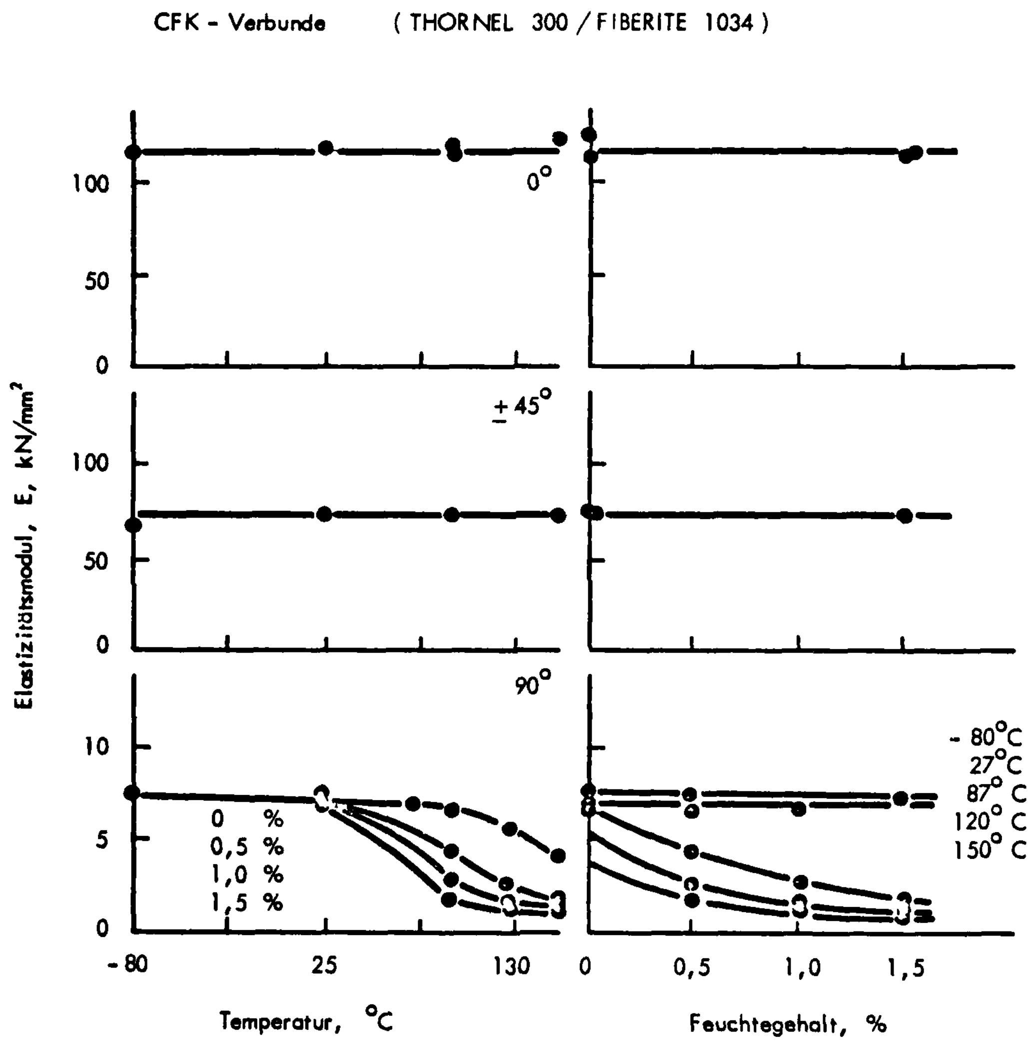
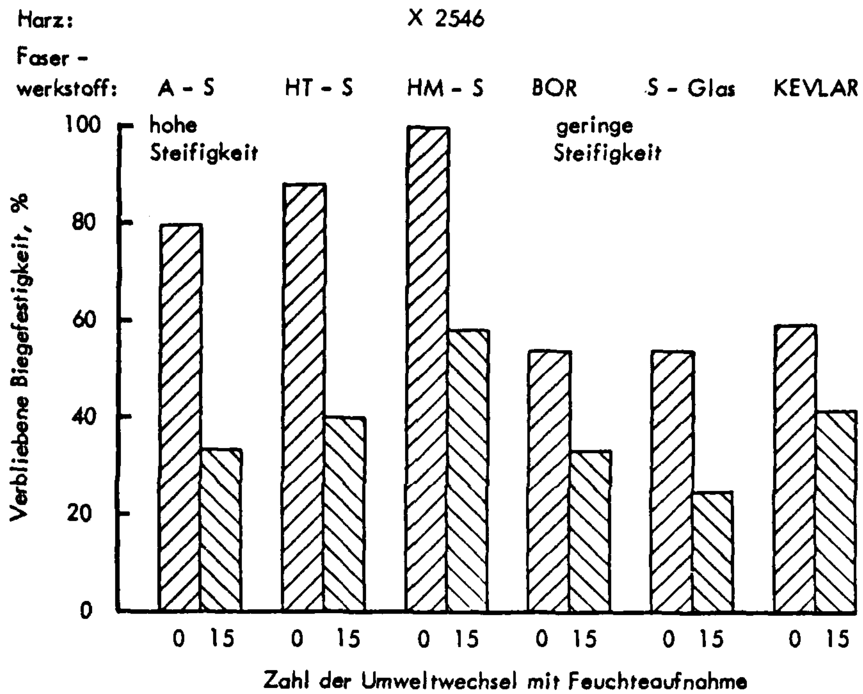


Fig 1.30 Effect of temperature (during loading) and moisture content on the stiffness of CFC (Ref 793)



Key:
Hohe Steifigkeit = high stiffness
Geringe Steifigkeit = low stiffness
Zahl der Unweltwechsel mit Feuchteaufnahme = number of environmental changes with moisture absorption

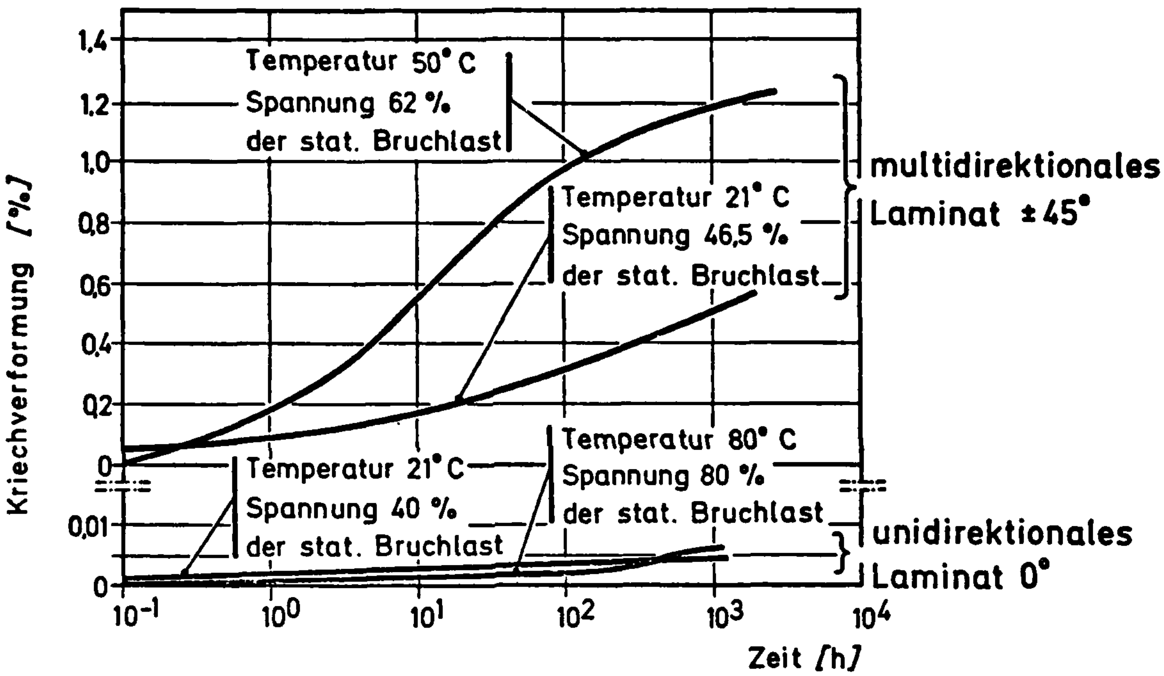
Fig 1.31 Bending strength of uni-directional composites with fibres of differing stiffness and the same resin with and without absorbed moisture at 177^{o}C during loading (Ref 92)

	Bauteil	Werkstoff		Einsatzzeit, Jahre Betrieb	Einsatzzeit, Jahre Versuch	Maximale Feuchte, % Gewicht	Verbliebene Festigkeit, % Zug	Druck	Schub, intral.	Biegung	Schub, interl.	Loch-leibung
Raumtemperatur	E-2A Rotodome	GFK, Gewebe a)		16	-	1,46	92	-	-	89	-	-
	A-6A Radomes	GFK, gewickelt a)		11	-	0,25	98	-	-	94	-	-
	Hautteile	CFK, AS/3501-5	blank	1	-	0,96	-	-	-	100	90	96
				2	-	0,94	-	-	-	100	88	99
			lackiert	1	-	1,34	-	-	-	100	88	98
				2	-	1,21	-	-	-	100	88	98
	Sandwich-Biegeproben	BFK, AVCO 5505		-	20	1,20	-	96	-	-	-	-
	B-1 Höhenleitwerk, Anschluß	BFK/CFK, AVCO 5505 / AS/3501-5		-	20	1,0	100	94	-	-	-	-
+ 127° C	Hautteile	CFK, AS/3501-5	blank	1	-	0,96	-	-	-	78	-	-
				2	-	0,94	-	-	-	81	53	-
			lackiert	1	-	1,34	-	-	-	71	-	-
				2	-	1,21	-	-	-	78	54	-
	Sandwich-Biegeproben	BFK, AVCO/5505		-	20	1,20	-	58	-	-	-	-
	B-1 Höhenleitwerk, Anschluß	BFK/CFK, AVCO 5505 / AS/3501-5		-	20	1,0	100	79	-	-	-	-
	B-1 Höhenleitwerk, Sandwichproben	CFK, AS/3501-5		-	20	1,50	-	-	-	-	-	-

a) GL/B28/MNA/BDMA

Key:
Hautteile = parts of skin
Höhenleitwerk = horizontal tail surfaces
Anschluss = connection
Einsatzzeit, Jahre = operating period, years
Betrieb/Versuch = operational/test
Verbliebene Festigkeit = remaining strength
Blank/Lackiert = clear/varnished

Fig 1.32 Strengths at room temperature and 127°C of fibre composites after real and simulated operating period with moisture absorption (Ref 528)



Key:
Kriechverformung = creep deformation
Spannung 62% der stat. Bruchlast = stress 62% of static breaking load

Fig 1.33 Creep behaviour of CFC (Ref 666)

2 MECHANICAL PROPERTIES UNDER REPEATED LOADING

J.J. Gerharz

Contents

2.1 Introduction

Structures of all types - here aircraft construction is of particular interest - are exposed to changeable loads in operation. In order to ensure the adequate operational safety of a structure during its assumed useful life, it is therefore essential to know its fatigue strength under conditions similar to those obtaining in practice. Important parameters of varying stress are as follows:

- the range of stress in the test cross-section which is bounded by the minimum stress σ_u and the maximum stress σ_0,
- the stress ratio $R = \sigma_u/\sigma_0$. Under varying stress in the purely tensile area $0 < R < 1$, in the purely compressive area $1 < R < \infty$, in the tensile/compressive area $-\infty < R < 0$ with the special cases $R = +1$ (static load) $R = -1$ (purely alternating load), $R = 0$ (tensile stress of varying magnitude) and $R = \infty$ (compressive stress of varying magnitude),
- mean stress $\sigma_m = (\sigma_0 + \sigma_u)/2$, and
- stress amplitude $\sigma_a = (\sigma_0 - \sigma_u)/2$.

Single-stage tests at *constant amplitude* are particularly easy to carry out and are suitable for the assessment of the fatigue behaviour of materials in the form of specimen bars or bar-type joints. Results of single-stage tests and the Woehler lines derived from them form the basis for arithmetical life prediction. In many areas of aircraft construction (*eg* on the fuselage) the operational load is very similar to a single stage load.

Endurance analysis for components which are primarily exposed to operational loads with variable amplitudes (*eg* wing areas) demands so-called flight-by-flight tests. The load sequences used are always representative of a definite operational use of aircraft, they therefore contain the essential characteristics of operational loading.

Over and above these general considerations there are some special features to be noted in the case of fibre composite materials. Thus the pronounced environmental sensitivity of fibre composites with plastic matrix, especially to moisture, renders it necessary to establish the fatigue strength of these fibre composites in different environmental conditions. In addition the drop in stiffness due to cyclic loading has to be investigated - particularly for the design of bearing surfaces. Since geometric notches in fibre composites presumably have a similar effect on fatigue behaviour as in metals, notched bars and joints should be included in the investigations.

The fatigue strength of fibre composites is described in the following sections according to data from the relevant literature, *eg* Refs 49, 254 and 732. These concern almost exclusively results of single-stage tests. The fatigue strength tests with variable amplitude loading which have been carried out to a very limited extent so far concern flight-by-flight tests with load sequences typical of aircraft load bearing structure.

CFC materials are used most frequently in load bearing aircraft structure at present and are therefore central to the following considerations. GFC is dealt with only marginally, although there are more extensive investigations available for this material in Refs 205, 209, 504, 509, 617, 684, 685 and 686. The CFC specimens examined are largely realistic angle ply composites which are adjusted in structure and fibre orientation to the multi-axial stresses in the structures of aircraft, rockets and space ferries. The components designed in fibres for these applications contain many $\pm 45^{o}$ and 0^{o} layers and the 90^{o} layers which are sometimes necessary. The 0^{o} layers lie in the principal load direction, the $\pm 45^{o}$ layers take the shearing stresses and the 90^{o} layers take the lateral stresses if the transverse strength of the 0^{o}, $\pm 45^{o}$ composite is not adequate.

Apart from tests to ascertain the endurance of complete components the behaviour of this composite has hardly been tested with multi-axial loading similar to that obtaining in practice. Most of the fatigue strength tests reported below were performed with uni-axial loading parallel to the fibres of the 0^{o} layer, *ie* with loading in the direction of maximum principal stress on the component. This paper does not discuss fatigue strength on multi-axial loading since this subject was treated in detail in Ref 705.

2.2 Repeated loading at room temperature

2.2.1 Effect of mean stress and stress ratio

The best view of the effect of the mean stress and stress ratio R is provided by presentation of the results of single-stage tests in the so-called Haigh diagram. This type of presentation is therefore described briefly, see Fig 2.1.

In the Haigh diagram the mean stresses σ_m are the abscissa values and the stress amplitude σ_a the ordinate values. The test results are entered as lines of constant life (N = constant). They all meet at $\sigma_a = 0$ and $R = +1$

(abscissa axis) in the range $\sigma_m > 0$ at the static tensile strength and in the range $\sigma_m < 0$ at the static compressive strength. Between these points the lines of constant life run separately at a greater or lesser distance from each other, intersecting the lines of constant R values which radiate from the origin ($\sigma_a = \sigma_m = 0$). The lines of constant maximum stress ($\sigma_0 = \sigma_m + \sigma_a$ = constant) are inclined at -45° to the abscissa axis ($\sigma_a = 0$ = constant), the lines of constant minimum stress ($\sigma_u = \sigma_m - \sigma_a$ = constant) at $+45^\circ$. All constant life lines therefore lie within a right-angled triangle whose hypothenuse is the section of the abscissa between the static strength values and whose short sides are the lines of constant maximum stress $\sigma_0 = \sigma_{Bz}$ (tensile strength) and constant minimum stress $\sigma_u = \sigma_{Bd}$ (compressive strength), see Fig 2.1.

In the metal materials used in aircraft construction the lines of constant life exhibit the same tendency in the range of positive mean stresses as in the range of negative mean stresses, see Fig 2.2. In contrast, the lines of constant life for fibre composite materials (see Fig 2.1) are deflected, even when the stress cycles extend only slightly into the compressive range, in the direction of $\sigma_m = \sigma_{Bd}$ (at $\sigma_a = 0$)[712,785], *ie* the tolerated stress amplitudes drop with increasing proportion of compressive stress (increasing negative minimum stress σ_u).

Fibre composite materials are therefore more sensitive to compression on repeated stressing than metals[785]. A possible physical explanation lies in the stress-strain behaviour typical of the material, which allows greater elongations on compressive stress than on tensile stress (see section 1.2.1). This leads to the stress on the matrix increasing in this fibre composite as the stress cycles increasingly enter the compressive range, and thus to earlier development of damage.

The Haigh diagram also shows as a function of mean stress and fatigue amplitude whether or not the static design covers fatigue strength. For this purpose a line must be drawn parallel to $\sigma_0 = \sigma_{Bz}$ (tensile strength) and $\sigma_u = \sigma_{Bd}$ (compressive strength) of the material at the distance of the static safety factor. The area above these lines is covered by the static design; in the area below these lines fatigue strength governs the design.

The lines of constant life for fibre composites[732,785] show the greatest distance from static strength in the range of $R = -5.0$ to $R = -0.5$, so that tensile-compressive stresses with 30-80% proportion of compression of the whole load cycle can be covered least by static design. The stress ratio at which the distance from static strength is greatest appears to be close to the ratio σ_{Bd}/σ_{Bz} of static strengths, as shown by a comparison of the Haigh diagrams in Fig 2.1.

2.2.2 Effect of notches introduced by the design

The previous section explained the presentation of results of single-stage tests in Haigh diagrams because the effects of stress ratio and mean stress are shown most clearly by this means. In describing the effects of notches, however, presentation of the results of single-stage tests is clearer in the form of Woehler lines which are therefore used hereunder. In the case of Woehler lines at constant values of R either the maximum or minimum stresses are plotted against endurance, depending on which of these two stresses is the greater, see Fig 2.3.

Position and shape of the Woehler lines of notched and joint specimens (no bonded joints) of fibre composites are determined by their static strength and the fatigue strength of the unnotched composite at large numbers of load cycles. As is known by comparing static strengths of notched and unnotched specimens (see section 1), the static strength of notched and joint specimens is below the static strength of unnotched specimens. For example, in drilled specimens with a ratio of hole diameter (d) to specimen width (w) of d/w = 0.2 the ratio of static strengths (unnotched to notched) is approximately 1.5[532,533,794]. In joint specimens these ratios can be up to 3.0[546,794].

The significant effect of the notch on static strength under cyclic load decreases steadily with increasing number of cycles to failure and can disappear completely after a large number of load cycles ($N > 10^6$), see Fig 2.3. This is in contrast to the usual fatigue strength behaviour of metals where the effect of notches on fatigue strength is greatest in the high endurance range. The most suitable method of illustrating notch sensitivity under cyclic load is by plotting β_k against endurance, β_k being the ratio: fatigue strength of the unnotched to fatigue strength of the notched specimen. β_k curves for notched specimens with approximately the same α_k are shown in Fig 2.4. They show the typical trend for metals (ductile aluminium alloy, 7075 - T6) and for stiff fibre composite materials (CFC with realistic laminate structure). Typical for metals is the rise in the β_k curve from value 1.0 at static strength ($N = 10^0$) to a figure in the endurance strength range ($N = 10^7$) which is close to α_k.

The composite material shows the reverse tendency: it falls from a higher value at static strength to 1.0 in the range of endurance strength. The static tensile strength of metals is virtually the same in notched and unnotched condition because of their ability to yield in highly stressed areas.

In fibre composites the falling notch sensitivity with increasing number of cycles to failure points to progressive neutralisation of the notch. This is confirmed by the observed development of longitudinal cracks which run along the fibres in the 0° layers at a tangent to the hole[258,532,625,717,732], see also section 4. The reduction in notch sensitivity can even lead to fractures outside the hole cross-section of the specimens[625].

Because of high notch sensitivity in the short-term strength range all measures to improve the static strength of composites also lead to improvement in fatigue strength; measures which have lead to improvement in static strength of joints are quoted in section 1.2.3. Here clamping force and interference fit of the fastener work out favourably[545,547].

For a comparison of fatigue strength of fibre composites with that of aluminium alloys normally used in aircraft construction it is suggested in Ref 711 that the stress amplitudes ($P_u = 50\%$) tolerated at $N = 10^6$ be related to static strength. For aluminium alloys these proportional values are for unnotched specimens $\sigma_a (N = 10^6)/\sigma_B = 0.15$ to 0.2 and for notched specimens 0.1 to 0.15[711]. In contrast the corresponding values for CFC with realistic laminate structure are 0.4 to 0.5. In this respect fatigue strength of fibre composites with realistic laminate structure is better than that of the aluminium alloys used in aircraft construction.

The facts described above do not apply to bonded joints. The decisive factor here is that in bonded joints the adhesive and the bond adhesive - component prove to be the critical weak spots. In consequence the fatigue strength of the bond (in contrast to bolted joints) is far below the strength of the unnotched fibre composite component. Tests with spliced bonded joints of CFC with an aluminium alloy[716] under fluctuating tensile stress show a greater drop in fatigue strength as compared with static strength, see Fig 2.5.

2.2.3 Deformation behaviour under cyclic stress

Cyclic stress causes a drop in stiffness in fibre composites with a plastic matrix. With an increasing number of load cycles in single-stage loads or with increasing number of flights in flight-by-flight loads (see next section) deformations increase during cyclic stress which corresponds to a drop in the secant modulus. This change in the deformation behaviour during and as a consequence of cyclic stress of fibre composites has been observed frequently[141,184,202,205,218,224,530,625,717,729,732].

Thus there is a clear connection between falling cyclic strength and progressive damage of fibre composites. It has been noted, for example, that the development of longitudinal cracks in notched specimens of fibre composites increases the drop in stiffness[732,794].

Results of other tests indicate that the increase in deformation of the fibre composite measured during or after cyclic stressing is always accompanied by damage[141,206,415,732]. In this connection it was observed in Ref 732, that with the occurrence of delamination, deformation increased sharply at the end of the life. The increase in deformation and corresponding drop in stiffness are generally at their maximum at the beginning and end of the life span[785]. The accompanying damage processes are presumably different. While the increase in deformation at the end of life is probably connected with delamination, at the beginning of cyclic stress it is presumably caused predominantly by the appearance of cracks which run parallel to the fibres in the individual layers relatively early in life, see section 4. The connection described between development of damage and increase in deformation applies both to unnotched and notched specimens with holes. At the same time the drop in stiffness of notched specimens is generally greater than that of unnotched specimens, which can be explained by the additional damage at the notch, see section 4.

In jointed specimens the increases in deformation on load transfer by the fastener can be considerably greater than in notched and unnotched specimens; moreover the greater increase in deformation in single-shear joints is more critical than in double-shear, since it occurs earlier as a comparison in Fig 2.6 shows. Local bearing stress is the cause of the increase in deformation in joints. The alternating bearing stresses lead to major bulges in the holes due to damage on the stressed hole wall. The X-ray picture in Fig 2.7 gives an indication of the type and intensity of this damage in the composite. The clearly visible faults such as matrix cracks parallel to the fibres and delaminations emanate almost exclusively from the stressed hole wall. Development of the faults caused by bearing stress can presumably be delayed by interference fit of the fastener and by clamping force generated between bolt head and nut.

Major expansions in the hole in the joint lead to major displacements between the connected components and may result in the structure being unfit for operation because of excessive deformation. The drop in stiffness of the composite alters the damping behaviour of the structure and has an adverse effect on its stability. For this reason the changes mentioned in deformation and stiffness with increasing stress cycles must be taken into account in the design, especially where the structure is designed for stiffness and stability. Further the loss in stiffness should be borne in mind in respect of the aeroelastic behaviour of the structure. Woehler lines for constant drop in stiffness or constant increase in deformation can be used as relevant data, see section 7.

2.2.4 Effect of flight-by-flight loading

Section 2.1 referred to the importance of fatigue strength tests under variable amplitude loading typical of service.

In Refs 113, 162, 517, 532, 613, 614, 668 and 785 tests were carried out with flight-by-flight loading simulating the load sequence on the wing skin of fighter aircraft. In these the effect of flight-by-flight loading on residual strength was the main feature tested, see section 3. Within the life span to be demonstrated in the test no fractures occurred at the design levels when the investigation concerned the lower wing skin, which is critical in metal construction methods[113,162,517], and where stresses are predominantly in the tensile range, see section 7.

As anticipated, flight-by-flight loading on the upper wing skin with stresses well into the compressive range proved to be more damaging[532,785]. Nevertheless fatigue strength of the unnotched and notched CFC tested in Refs 532 and 785 under flight-by-flight loading is still adequate for the upper wing skin if at a life span of approximately $5 \cdot 10^4$ flights the usual static safety factor of 1.5 is demanded, see Fig 2.8. As to how far this applies to other components as well depends inter alia on the content of the load spectrum.

An extreme example is single-stage loading and a spectrum with maximum envelope (rectangle). Assume the case that a component is subjected to a purely alternating load ($R = -1$) with constant amplitude and that only one load cycle occurs per flight. On this assumption the Woehler line for $R = -1$ as in Fig 2.3 only covers approximately $2 \cdot 10^4$ flights at a stress level representing a static safety factor of 1.5; this is normally inadequate as evidence of fatigue life.

Flight-by-flight loading leads to a smaller drop in fatigue strength in joint specimens as compared to static strength than might have been expected after the results of single-stage tests. Particularly for joint specimens, however, premature increase in deformation during cyclic stress is to be expected in flight-by-flight loading[794].

2.2.5 Effect of laminate structure and fibre material

According to observation of damage (see section 4) all damage in composites starts either in the matrix or in the fibre/matrix bond ('interface'), *ie* these areas of the composite are critical. Fatigue tests on carbon, boron, glass fibre uni-directional composites produce virtually no drop in fatigue strength as compared with static strength[49,728]. Thus the fatigue strength of a composite depends on the stresses in the matrix and the 'interface'. These are particularly heavy if inter- and intralaminar shear stresses as well as normal stresses perpendicular to the fibre occur, which for their part depend on the type of external loading (axial, bending, shear), orientation of the fibres and arrangement of the layers in the composite (laminate structure).

If due to changes in the laminate structure all the above stresses are increased for the same cyclic load, then faults in the matrix and/or 'interface' occur earlier and to an increased extent. This is also indicated by an increased drop in stiffness. Examples of this are tests in Ref 794 and investigations in Ref 712 on CFC with 0^{o}, $\pm 45^{o}$ and 90^{o} layers; specimens of these composites were tested with load in the 0^{o} direction and also tested with load in the 90^{o} direction. The greater load on the matrix and the interface for the 90^{o} load direction increased the rise in deformation considerably towards the end of the life span and reduced fatigue strength in the entire load cycle range by an approximately constant factor, see Fig 2.9.

The above interrelationships also explain the greater fatigue strength sensitivity of GFC. Since glass fibres are far less rigid than carbon and boron fibres, stresses on the matrix in the layers with fibres in the direction of loading are great enough to cause a major drop in fatigue strength compared with the static strength of GFC, for example, under varying tensile stress. The comparison of Woehler lines for uni-directional CFC and GFC and a CFC/GFC hybrid in Fig 2.10 with constant stress ratio $R = 0.1$ clearly shows the effect of fibre stiffness described[530].

It has been pointed out that the laminate structure can have a major effect on fatigue strength and deformation behaviour under cyclic stress. The test results available at present can therefore only be used for the design of fibre composite components made of the particular composite examined. An exclusively experimental determination of fatigue strength characteristics for all customary composites does not appear to be justifiable in view of the large number of tests which would be required. Other courses must therefore be taken. One possibility would be to produce design data with specimens of simply structured composites in which one only of the above critical loadings occurs.

The local stresses would then have to be calculated for the component whose life was to be predicted. With these stress values the life span could be predicted separately for the various types of failure. Thus the results of fatigue tests would no longer be compared with the nominal stresses in the critical cross-section of the component, but related to the local stress which initiates the damage. Generally applicable design data could be set up for such a concept.

This procedure will of course encounter considerable difficulties, since the failure of the component does not occur when the first damage develops; the component remains serviceable until the fault or combination of faults has reached a certain dimension. The calculation of the combined stressing for interactive faults will present great difficulties.

2.2.6 Summary

To summarize, the following can be stated on the subjects mentioned:

Effect of mean stress and stress ratio:

- CFC is more sensitive to compression in cyclic loading than metals,
- the drop in fatigue strength as compared with static strength is greater under alternating loads ($-5 \leq R \leq -0.5$) than under cyclic load in the tensile range ($-0.5 < R < 1.0$) and cyclic load in the compressive range ($1.0 < R < \infty$); fatigue strength is least covered by static design under alternating load.

Effect of notches introduced by design

- stress concentrations reduce fatigue strength in the short-term strength range; their effect disappears at the higher endurances,

- slope and curve of the Woehler lines for notched and joint specimens can be estimated from static strength and endurance strength of unnotched specimens of the same composite,
- measures which improve the static strength of fibre composites with stress concentrations dictated by the design also increase their fatigue strength,
- a greater drop in fatigue strength as compared with static strength must be expected in bonded joints than in bolted joints.

Deformation behaviour under cyclic stress:

- during cyclic stress on angle ply composites with a plastic matrix deformation increases and stiffness drops,
- at the beginning and end (shortly before fatigue fracture) of stress cycling the increases in deformation are greatest,
- increase in deformation/loss of stiffness are caused by damage in the matrix and the interface,
- the increase in deformation is greater in specimens with holes than in specimens with uninterrupted cross-section,
- in joints with load transfer by bolts increases in deformation due to bearing stresses are generally considerable, especially in single-shear joints.

Effect of flight-by-flight loading:

- multi-stage stresses with flight-by-flight load sequences lead to a drop in fatigue strength compared with static strength in the life range of interest only if a major proportion of load cycles lies in the compressive range,
- in flight-by-flight loading for the upper wing skin (predominantly in the compressive range) it is to be anticipated that the drop in fatigue strength as against static strength will be covered by the static design,
- in joints flight-by-flight loading (as single-stage loading) can lead to major increases in deformation.

Effect of laminate structure and fibre material:

- changes in laminate structure which increase the stress in the matrix and fibre/matrix bond reduce the fatigue strength of the composite,

- composites with fibres which are less stiff than carbon and boron fibres are more sensitive to cyclic stress,
- the range of tests for the establishment of design data for angle ply composites with realistic structures will depend on whether success is achieved in deriving the fatigue strength behaviour of a composite from the fatigue strength behaviour of the individual layers.

2.3 Effect of the environment on fatigue strength

2.3.1 Environmental conditions

Environmental conditions such as temperature, humid air or liquids can be of great significance to the strength properties of fibre composites with a plastic matrix. The effect of these environmental conditions on fatigue strength will be discussed in this section.

The effect of the following media is important in connection with the use of fibre composites in aircraft:

Air with alternating moisture content
Fuel/kerosene
De-icing media
Water/rain
Hydraulic fluids
Methanol

Of these 'air with alternating moisture content' is for aircraft the medium which occurs most frequently and has the longest lasting effect. It is therefore the subject of most tests. In space flight equipment, however, the effect of extremely low temperatures is prominent.

Most components are subject in operation to the combined effect of the medium, temperature and mechanical loading, while a typical correlation of mechanical loads and environmental changes occurs for much equipment. In aircraft, for instance, during the ground-air-ground load cycle on the wing the environmental change room temperature, high relative humidity → -55 °C → room temperature, at high relative humidity occurs.

For economic safe design it must be possible to indicate the behaviour of the fibre composite component under such operational conditions. For this tests with environmental *and* load simulation are necessary.

Most of the tests performed on the effect of environmental exposure on fatigue strength can be classified in the following groups:

(i) environmental exposure before repeated loading only (ageing),
(ii) environmental exposure during repeated loading only, and
(iii) environmental exposure before and during repeating loading.

Environmental conditions are either constant or they change, *ie* in temperature and/or humidity.

2.3.2 Effect on fibre composites with extremely stiff fibres

Uni-directional 0° composites

When uni-direction composites are stressed parallel to the fibres in the tensile range the effect of the environment on fatigue strength is generally slight[316,516]; for instance a drop in fatigue strength of 10% was noted only after 1000 hours at 98% relative humidity and $50^{\circ}C$. In other environmental conditions such as high test temperature or temperature and environmental cycles before loading fatigue strength even increased very slightly. Fragments of the 0° specimens showed increasing disintegration at higher temperatures, which points to increasing failure of the matrix and reduction in shear load-carrying capacity of the fibre/matrix bond; this is also indicated by many longitudinal cracks and drawn out fibres on the fracture face. The drop in shear load carrying-capacity of uni-directional composites caused by environmental effects is confirmed by investigations in Ref 144. Storage and bending stress on short-term bending specimens in water reduced life to 50% and dropped stiffness considerably, see Fig 2.11. The moisture absorbed led to great losses in stiffness due to the high shear stresses in short-term bending specimens and impairment of the matrix and fibre/matrix bond by the moisture, which showed itself in rapidly increasing flexure.

Multi-directional composites

In contrast to uni-direction 0° composites, the stress on the matrix and the fibre/matrix bond is relatively high in multi-directional composites even where the load direction is parallel to the fibres of the 0° layers. This is the reason for the greater environmental effect on multi-directional composites which increases with a falling proportion of 0° layers.

Of the various environmental conditions high temperatures during cyclic loading have the most effect[313,516], *eg* a drop in fatigue strength was found in

0°, ±45°, 90° CFC of up to 20%[516]. Presumably the drop increases the nearer the test temperature approaches the glass equilibrium temperature of the resin[517]. In environmental exposure before cyclic loading (ageing) the drop in fatigue strength found in Ref 516 was only half as much; other tests produced no, or very little effect of ageing[316,538,704].

In simulating practical environmental conditions, therefore, care should be taken to see that high temperatures which occur in operation during cyclic stress also occur in the test simultaneously with cyclic stressing. Simplified simulation by heating before stressing would lead to results on the unsafe side.

The drop in fatigue strength due to the environment is however in general greater when the cyclic stress has a compressive component[316,712] than when it is exclusively in the tensile range. The reason is that the weakening of the matrix and fibre/matrix bond caused by environmental stress are brought more into play by compressive stresses. As shown by observations of damage in Ref 516, this is especially the case when delamination occurs to an increased extent (see section 4). There it was established inter alia that moisture absorption promotes delamination and that ageing with moisture and temperature changes leads to extensive delamination.

<u>±45° composites under axial stress</u>

±45° composites are considered separately from the other multi-directional composites since the high intralaminar shear stresses severely tax the fibre/ matrix bond under axial loading. Correspondingly these composites react strongly under axial loading to any weakening of the fibre/matrix bond. This is also the reason for the reductions in life by factors of more than 10 established after moisture absorption through ageing with temperature and humidity cycles in Ref 369. This fracture mechanism is confirmed by fractographs in which frequent debonding (failure of the fibre/matrix bond) on the fracture faces was observed. The weakening of the fibre/matrix bond was probably caused by the penetration of moisture. This is supported by the fact that when moisture absorption was inhibited by a surface coat of polyurethane paint there was no reduction in life under the same test conditions. The environmental stresses in this case were environmental changes lasting 6 weeks before cyclic loading which simulated environmental conditions for a modern fighter aircraft mission.

Adhesive bond

A stepped BFC - titanium 64 adhesive joint with one bolt per stage was stressed with a flight-by-flight load sequence for the wing underside and a superimposed temperature cycle per flight with real-time sequence (on average 0.225 Hz, test series A). In addition tests were performed with real-time flight-by-flight stressing without temperature change (test series B) and tests with higher test frequency (on average 3.7 Hz, test series C). For test series A, B and C mentioned the mean life values were as

$$1 : 1.3 : 2.2$$

and the scatter of life as

$$1 : 1.2 : 1.7 .$$

Thus the effect of frequency appears to exceed that of temperature change. However, the scatter is wide so that further investigations are needed to confirm this impression[149].

2.3.3 Effect on GFC

In GFC high temperatures during cyclic stress significantly reduce fatigue strength in contrast to CFC[391]. The drop increases as the load amplitude extends increasingly into compression.

Ref 684 reports on flat specimens of 0° and $\pm 45^{\circ}$ GFC laminates which were stored at room temperature for 5000 hours in water before single-stage loading (R = 0) in air. Fatigue strength of the exposed $\pm 45^{\circ}$ specimens drops in the entire life range examined $10^4 < N < 10^7$, while the fatigue strength of the exposed 0° specimen approaches that of the dry specimen in the high-cycle range. When the GFC specimens were only placed in water during cyclic loading the specimens were able to absorb more moisture at the lower levels because of the longer duration of the test than at the upper levels. Accordingly the fatigue strength of the 0° and the $\pm 45^{\circ}$ laminate dropped sharply at the lower levels in this case. It was also found that GFC laminates absorb more moisture under cyclic stress than in the unstressed condition. Investigations of specimens with varying pre-damage n/N indicate the interaction of growth of damage with moisture absorption and similarity with the phenomena of corrosion fatigue in metals. A fractographic examination which shows that the fibre/matrix bond is disintegrated by penetrating moisture[648] helps to explain this fact.

2.3.4 Conclusions on the effect of the environment on fatigue strength

The investigations on the effect of the environment on fatigue strength show that in all fibre composite materials the environmental effect on fatigue strength is governed by the reaction of the fibre/matrix bond to environmental conditions. Negative effects of the environment on the strength of fibre composites become more distinct the more the fibre/matrix interface is stressed by the external load. Where no effect or positive effects of the environment were observed the stresses on the interface were slight, as for example the unidirectional laminate stressed axially in fibre direction. Here presumably the life span between the first fibre fracture and total fracture is extended if environmental damage on the interface reduces the load concentration in those fibres which are in the immediate proximity of a torn fibre.

2.3.5 Effect of low and cryogenic temperatures

During cyclic stress 0^{o} and 0^{o}, 90^{o} CFC laminates with HM and HT fibres were cooled to $-40^{o}C$ with vaporising nitrogen[316]. The effect of low temperatures in the endurance strength range examined is slight at $R = -1.0$. For HM fibre specimens a minor drop was registered in fatigue strength at $N = 10^{6}$ and for HT fibres a minor rise compared with the room temperature results.

Ref 529 reports on fatigue strength investigations on multi-directional BFC at $-197^{o}C$, dipped in liquid nitrogen, in tensile and compressive ranges ($R = +0.1$ and $R = +10.0$). The effect of low temperature on fatigue strength was slight here too. The low temperature improved the fatigue strength of the BFC specimens to a minor extent.

In contrast, tests on GFC roving specimens with 69% fibre component at $4^{o}K$ ($-269^{o}C$ in liquid helium) at $R = 0.1$ showed up to 10 times longer life than at room temperature[315].

Summarizing, investigation of the effect of low temperatures on fatigue strength shows the following tendencies:

- in CFC and BFC the effect of low temperatures during cyclic stressing is negligible,
- in GFC composites low temperatures during cyclic stressing can materially increase fatigue life.

2.4 References to the literature

In this section the references evaluated in the preceding sections are classified according to contents.

Section 2.1

Introduction:

49, 205, 209, 254, 504, 509, 617, 684, 685, 686, 705, 732.

Section 2.2.1

Effect of mean stress and stress ratio:

49, 712, 732, 785.

Section 2.2.2

Effect of notches introduced by the design:

44, 63, 96, 166, 174, 176, 252, 253, 258, 343, 377, 420, 532, 533, 545, 546, 547, 591, 620, 622, 625, 634, 711, 716, 717, 732, 763, 766, 770

Section 2.2.3

Deformation behaviour under cyclic stress:

98, 141, 144, 153, 158, 182, 184, 202, 203, 204, 205, 206, 216, 218, 224, 227, 234, 245, 254, 280, 315, 316, 317, 338, 390, 415, 474, 500, 529, 530, 531, 535, 625, 717, 729, 732, 785, 794.

Chapter 2.2.4

Effect of flight-by-flight loading:

113, 162, 517, 532, 613, 614, 668, 785, 794.

Section 2.2.5

Effect of laminate structure and fibre material:

7, 49, 530, 712, 728, 794.

Section 2.3.1

Environmental conditions:

271, 356, 427.

Section 2.3.2

Effect on fibre composites with stiff fibres:

144, 149, 316, 369, 399, 516, 517, 528, 538, 553, 618, 704, 712.

Section 2.3.3

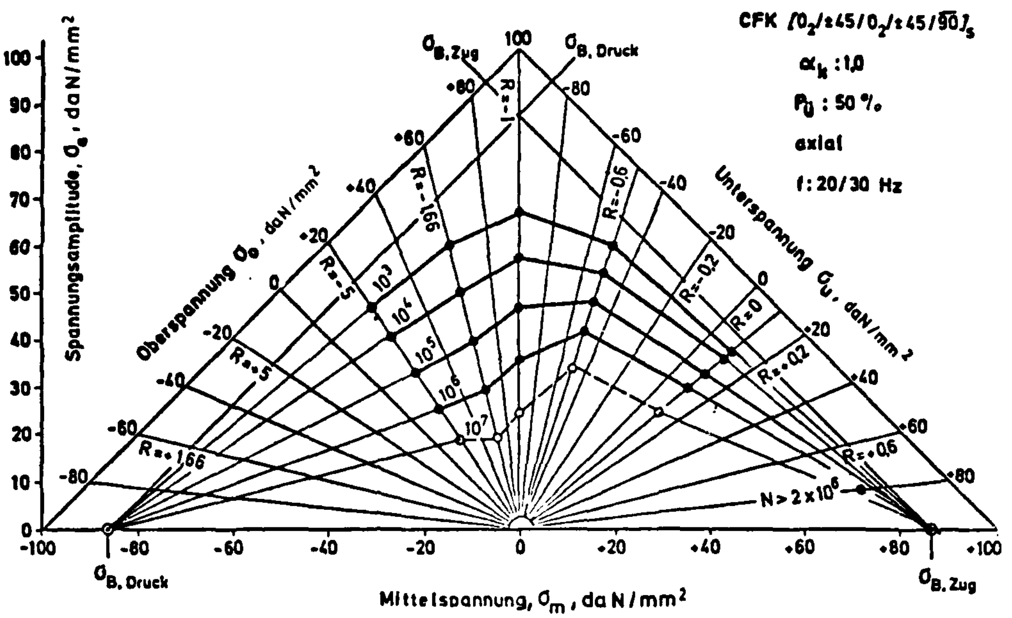
Effect on GFC:

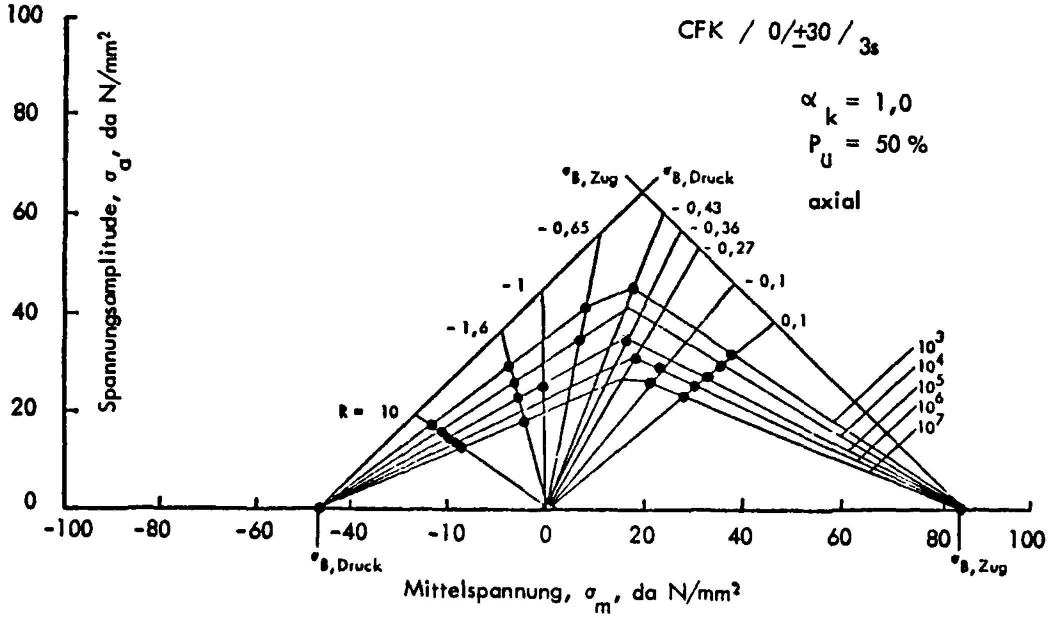
113, 209, 334, 391, 648, 684.

Chapter 2.3.5

Effect of low and cryogenic temperatures:

315, 316, 529.





Key:
Oberspannung = maximum stress
Unterspannung = minimum stress
Mittelspannung = mean stress
Druck = compression
Zug = tension

Fig 2.1 Course of constant life lines in the Haigh diagram for unnotched CFC angle ply (Ref 732)

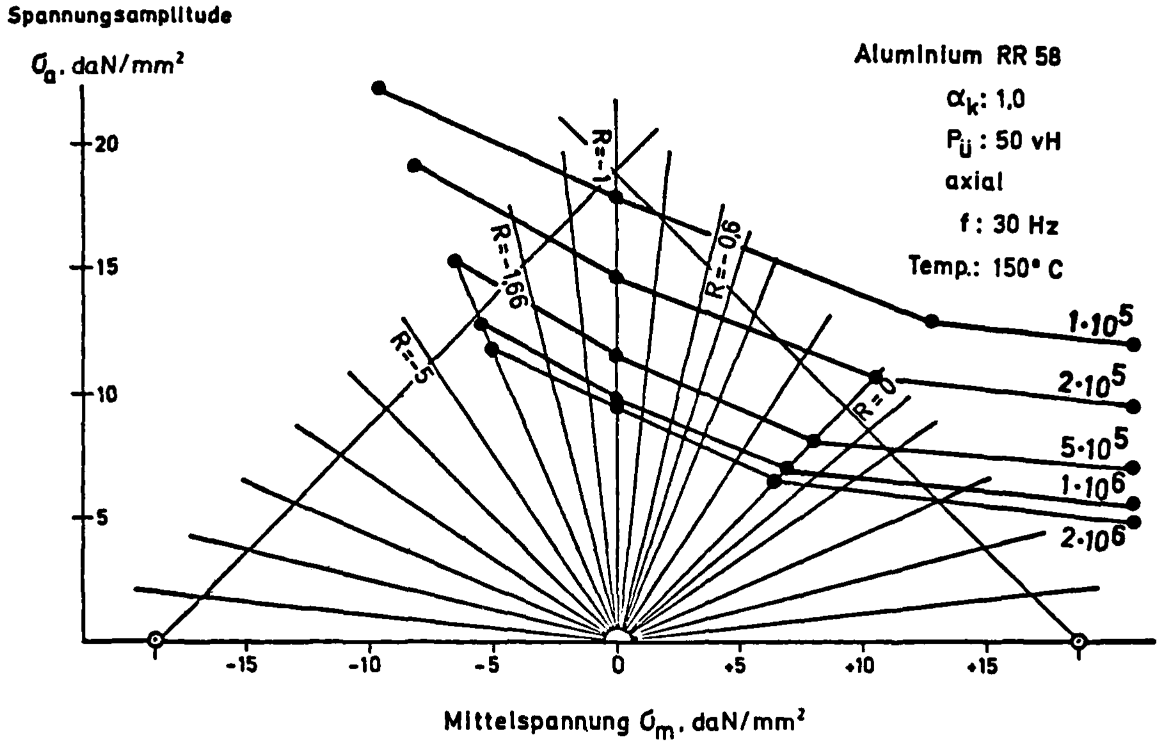
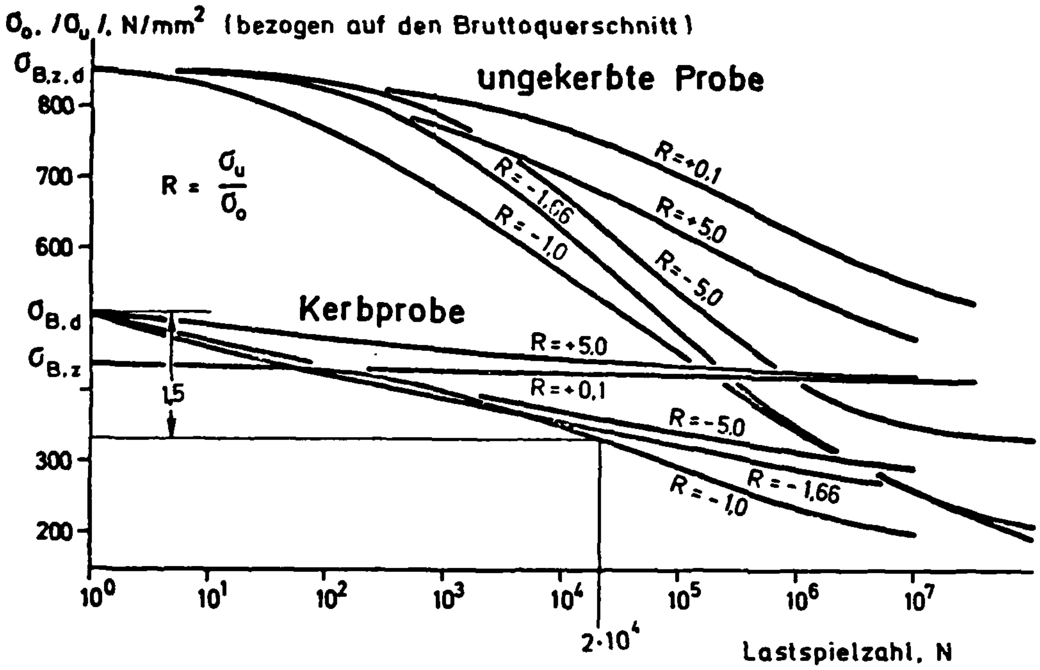
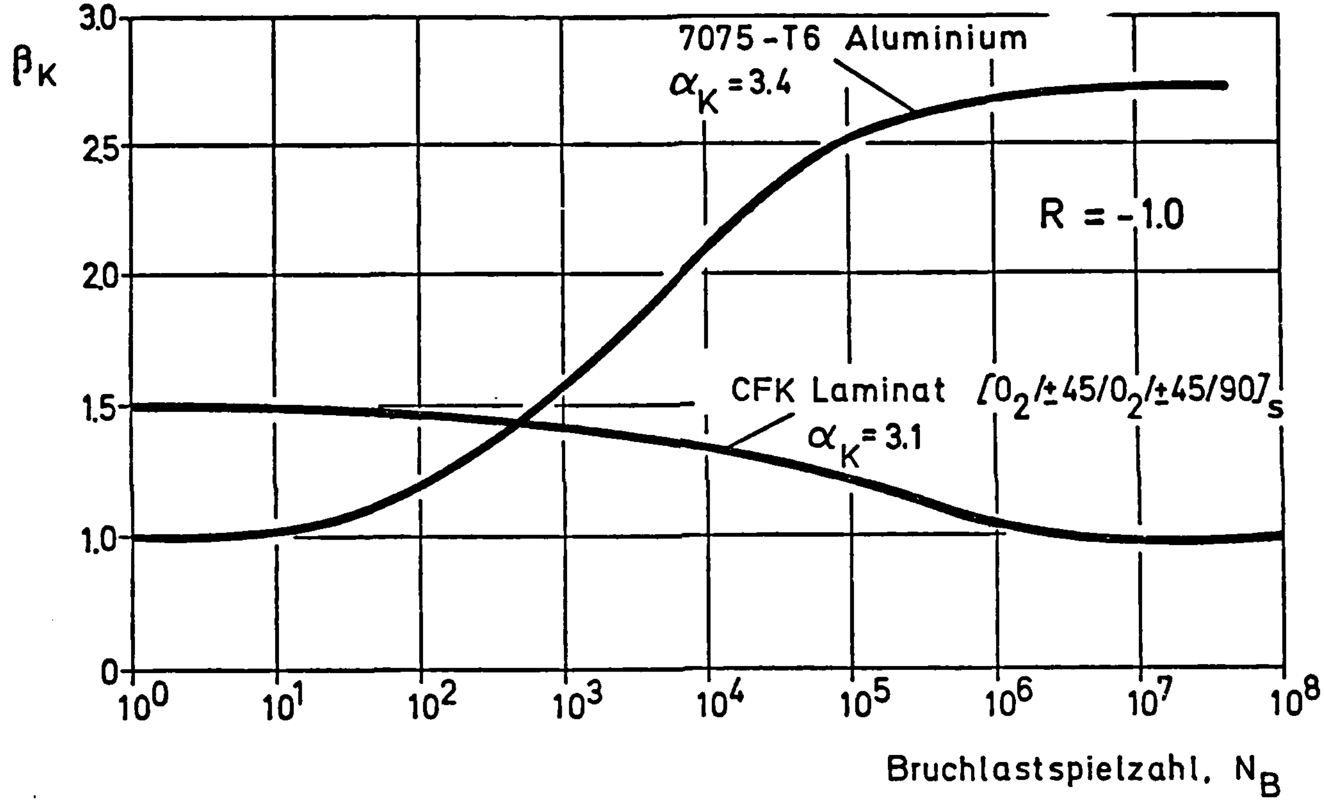


Fig 2.2 Haigh diagram of unnotched aluminium RR 58, axial load



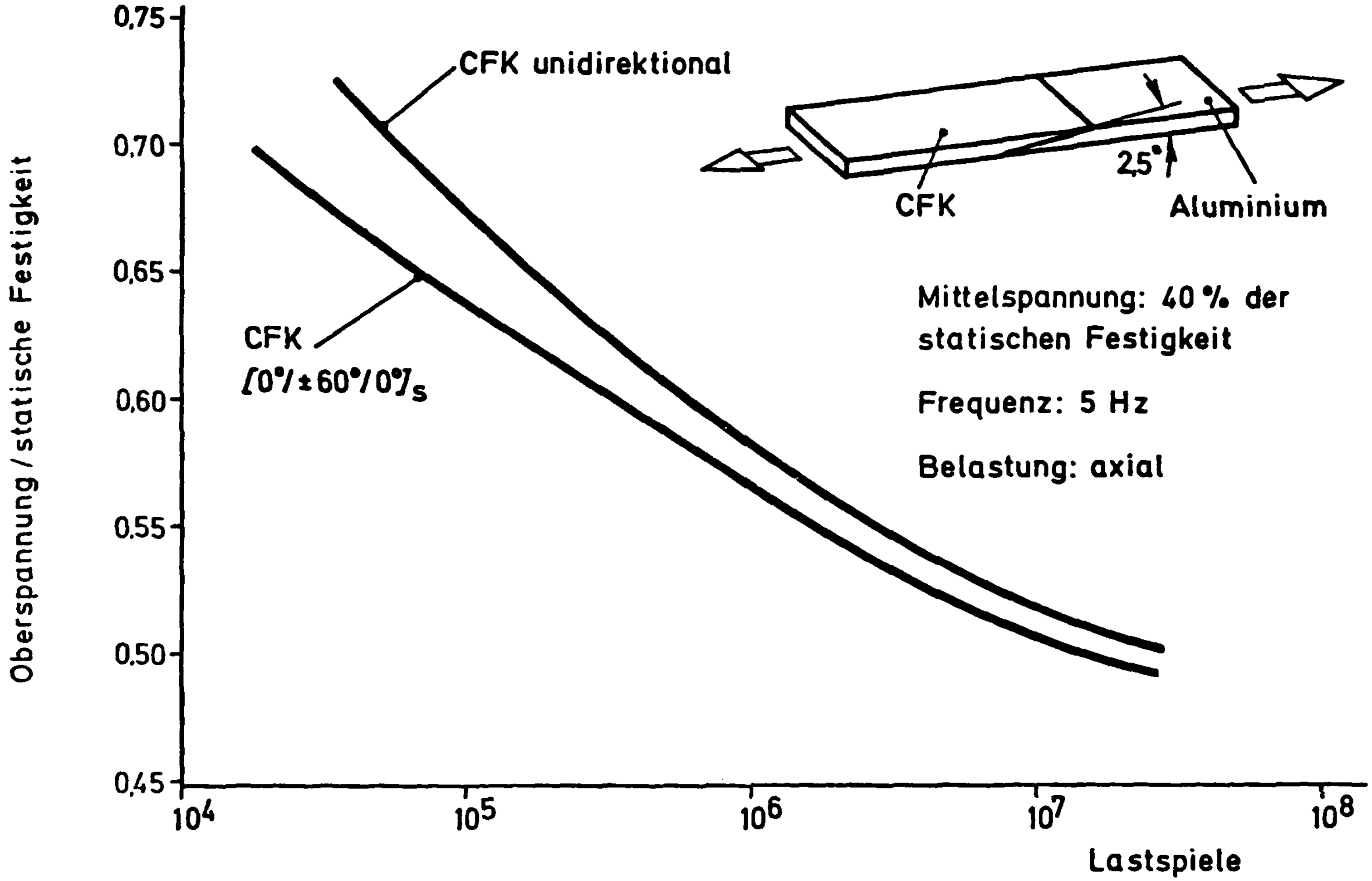
Key:
Bezogen auf den Bruttoquerschnitt = related to gross cross-section
Ungekerbte Probe = unnotched specimen
Kerbprobe = notched specimen
Lastspielzahl = load cycle

Fig 2.3 Wöhler lines of notched and unnotched CFC specimens



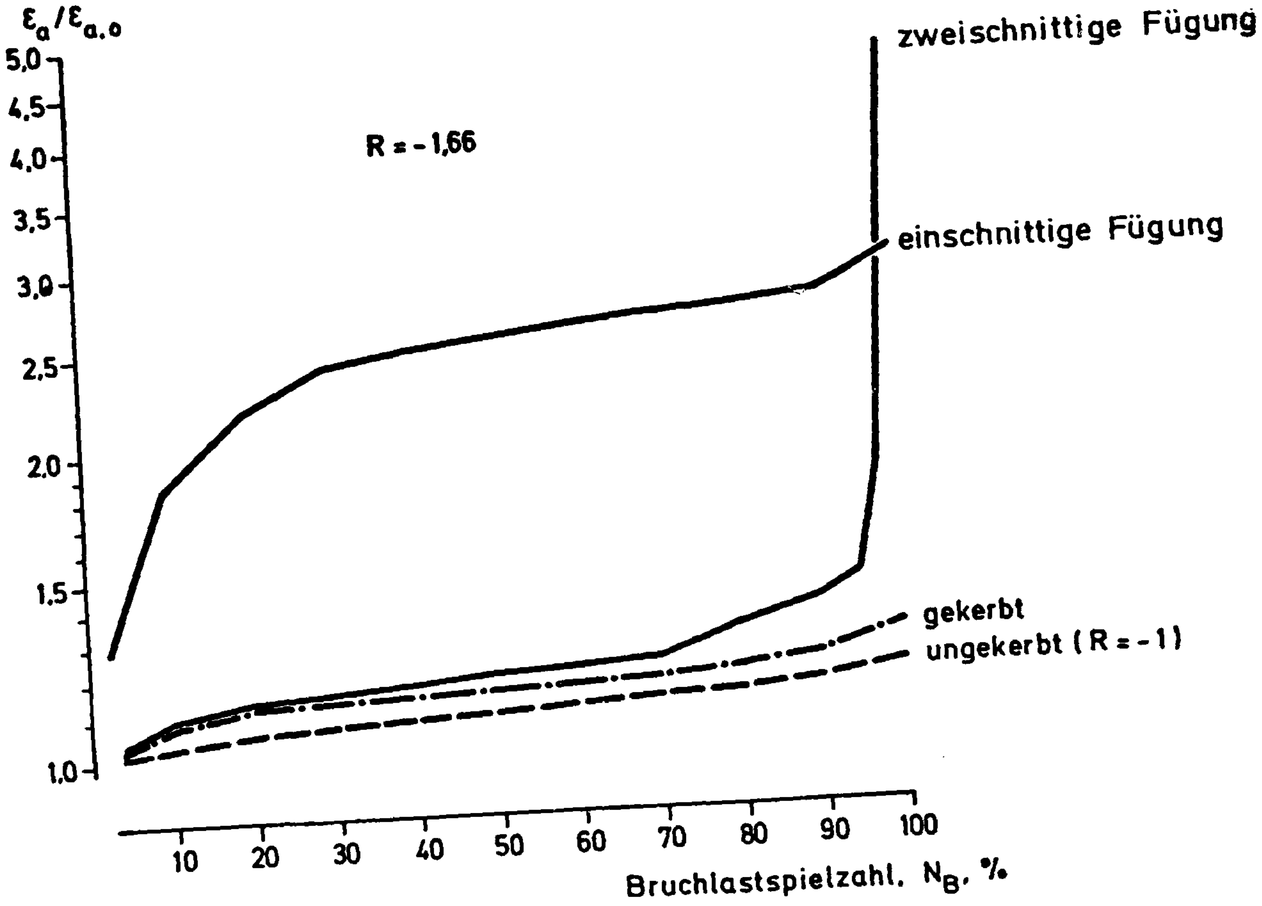
Key:
Bruchlastspielzahl = number of cycles to failure

Fig 2.4 Notch sensitivity of CFC



Key:
Oberspannung/statische Festigkeit = maximum stress/static strength

Fig 2.5 Fatigue strength of spliced adhesive bond CFC aluminium (Ref 716)



Key:
Zweischnittige Fügung = double-shear joint
Einschnittige Fügung = single-shear joint

Fig 2.6 Increase in deformation during cyclic stressing (Ref 794)

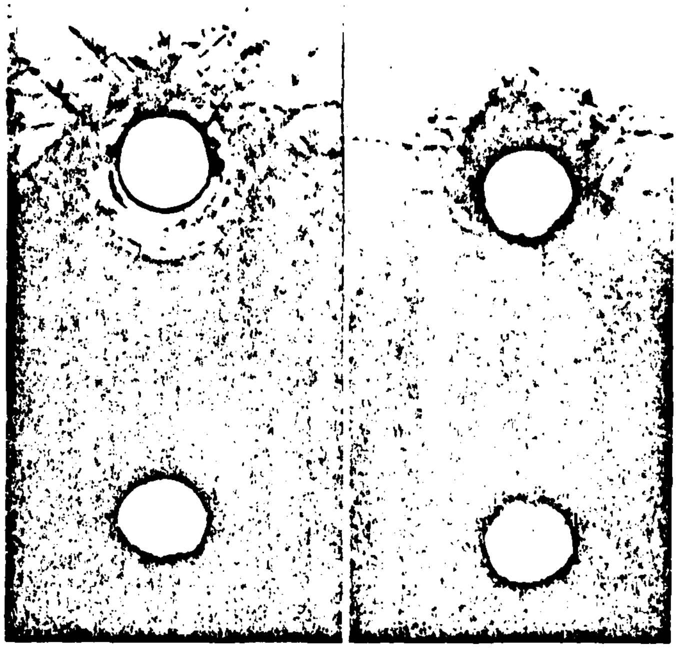
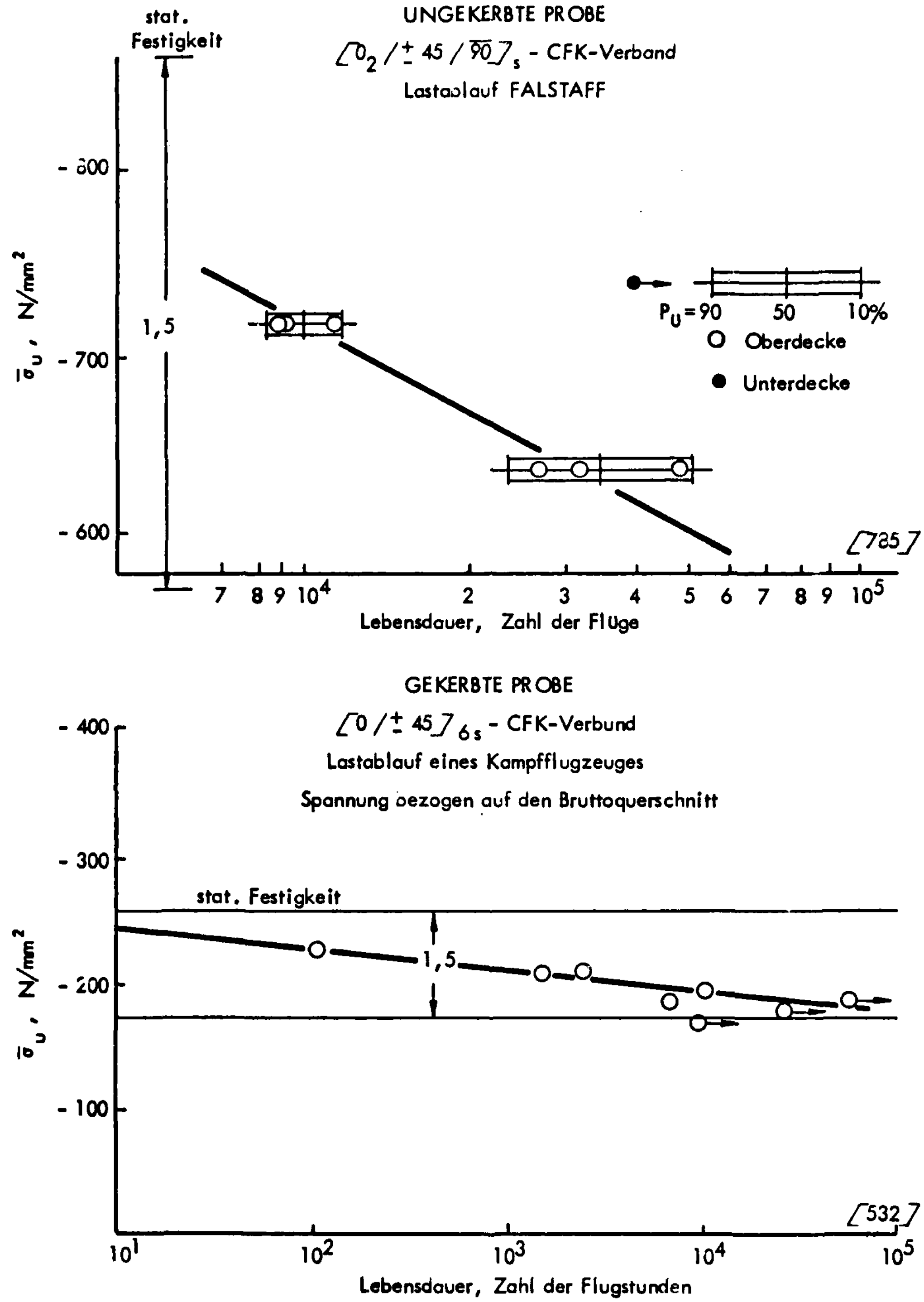


Fig 2.7 Damage after repeated bearing stress on application of varying tensile stress to joint (Ref 162)



Key:

Oberdecke	= upper skin
Underdecke	= lower skin
Lebensdauer, Zahl der Flüge	= fatigue life, number of flights
Lastablauf eins Kampfflugzeuges	= load cycle of a fighter aircraft
Spannung bezogen auf den Bruttoquerschnitt	= stress related to gross cross-section
Zahl der Flugstunden	= number of flight hours

Fig 2.8 Fatigue strength of unnotched and notched CFC specimens under flight-by-flight loading

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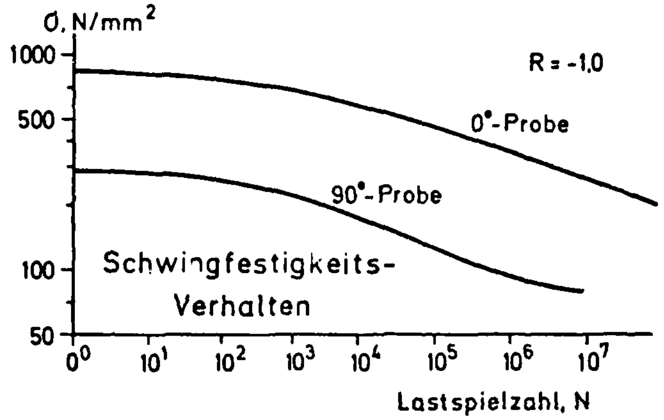

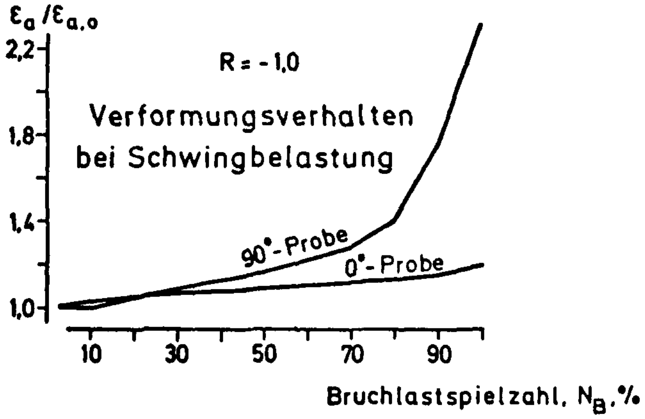

Key:
Schwingfestigkeitsverhalten = fatigue strength behaviour
Verformungsverhalten = deformation behaviour

Fig 2.9 Effect of laminate structure

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LITERATURE RESEARCH ON THE MECHANICAL PROPERTIES OF FIBRE COMPO--ETC(U)
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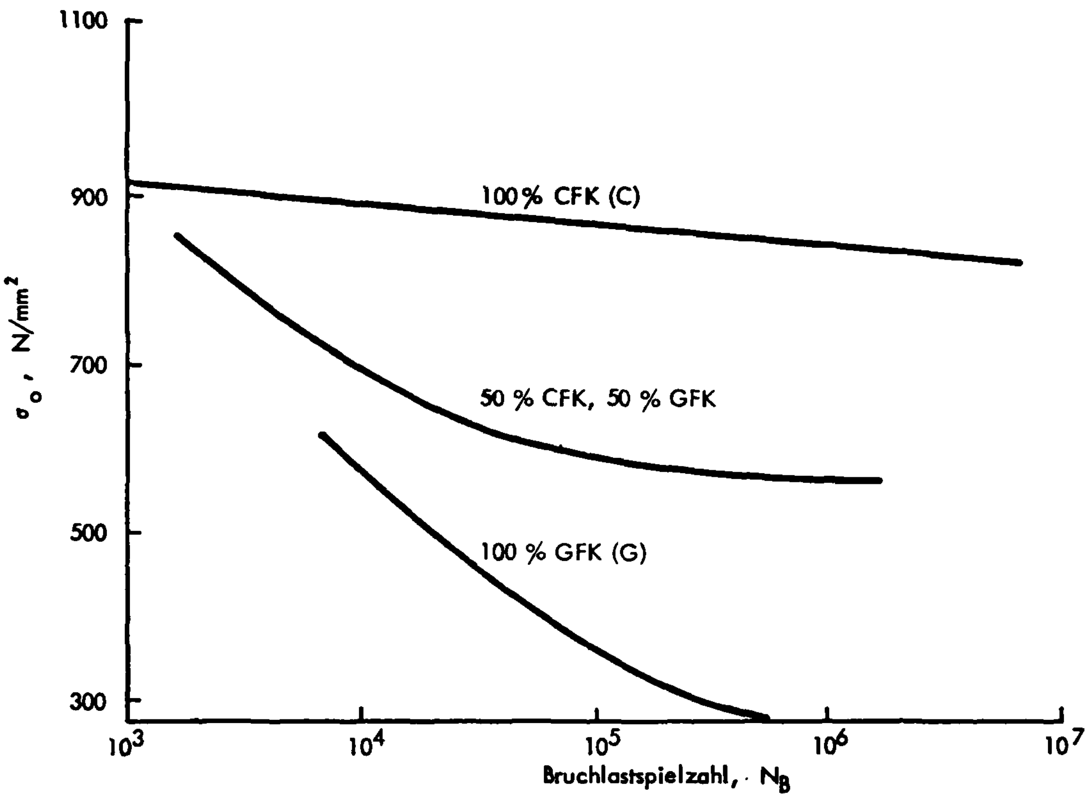

Fig 2.10 Effect of fibre material on the fatigue strength of uni-directional composites (Ref 530)

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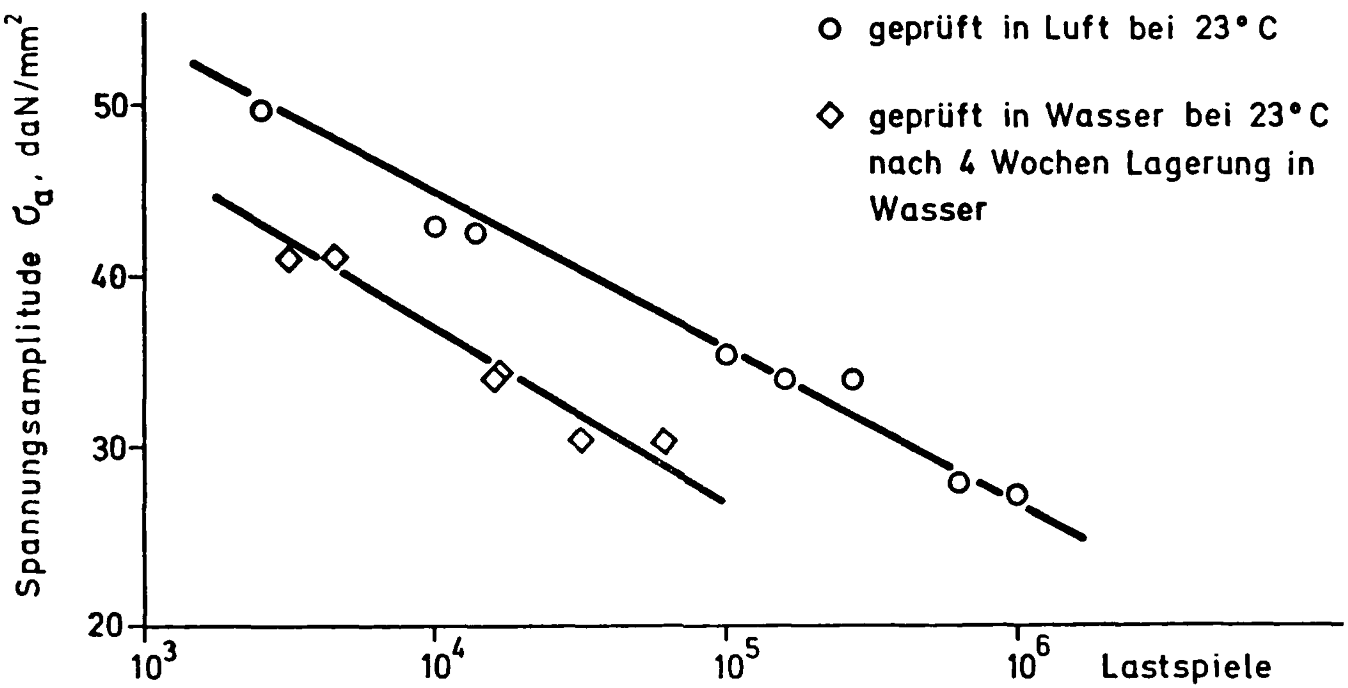

— Biegeversuche an unidirektionalem CFK bei R = -1

— Versagenskriterium 50 % Steifigkeitsverlust

Key:

Geprüft in Luft bei 23°C	= tested in air at 23°C
Gepruft in Wasser bei 23°C nach 4 Wochen Lagerung in Wasser	= tested in water at 23°C after 4 weeks' exposure in water
Biegeversuche	= bending tests
Versagenskriterium 50% Steifigkeitsverlust	= failure criterion 50% loss of stiffness

Fig 2.11 Effect of moisture on fatigue strength of CFC (Ref 144)

3 MECHANICAL PROPERTIES IN PRE-DAMAGED CONDITION

J.J. Gerharz

Contents

3.1 Introduction

Evidence of reliability of structure must include consideration of the drop in static strength after:

(a) repeated loading,

(b) discrete damage (from turbine blades, hailstones, birds, projectiles).

Investigations reported in the literature on the effects of preliminary loads (a) and preliminary damage (b) on the mechanical properties of fibre composite materials are evaluated below.

3.2 Mechanical properties after preliminary loading

The effect of preliminary loading on static strength, stiffness and Poisson's ratio is closely linked to the damage caused in preliminary loading[523]. Establishment of the crack density and residual strength as a function of the load cycle, *eg* in orthotropic 0°, 90° GFC demonstrates this connection[49,224,391], see Fig 3.1. It was observed in all the investigations that residual strength drops predominantly with the increase in cracks parallel to the 0° fibres. While cracks perpendicular to the load direction in the 90° fibres appear with the first stress cycles and are more numerous than cracks parallel to the direction of loading in the 0° layers, they have no direct effect on residual strength. Their indirect effect consists in acting as 'starters' for longitudinal cracks[224]. Similar types of damage can have varying degrees of effect in the residual strength test under different loads; if delamination is caused in an angle ply specimen, for instance, this will have a major effect on compressive and bending strength and a lesser effect on tensile strength and axial stiffness.

As shown in section 4, the predominant types of damages are determined largely by laminate structure and shape of specimen (notched, unnotched); while the state of damage (in development of damage) depends on the duration of load application (stress cycle, number of flights) and on the load level.

3.2.1 Effect of type of stressing in preliminary loading and residual strength test

Analysis of the damage tolerance of fibre composite component is also affected by the type of load occurring. Sendeckyi (AFFDL) for instance points out in Ref 523 that delaminated layers have a different effect on residual tensile strength from that on residual compressive strength. Experience shows that delaminated layers buckle under compressive stress, which leads to a drop in residual compressive strength compared with static compressive strength. A distinct difference between compressive and tensile residual strength is shown by the results of Wurtinger[209] with flat specimens of 0°, 90° GFC. Fluctuating compressive stress which leads to more extensive delaminations than fluctuating tensile stress reduces compressive residual strength more than tensile. Tensile residual strength drops to the same extent after fluctuating tensile or compressive stress. The difference between compressive and tensile residual strength due to the type of damage is independent of the cause. Delaminations caused by the impact of hard objects also produce a greater drop in compressive strength[533].

If less extensive delaminations occur during preliminary loading the differences between tensile and compressive residual strength are also less, as shown by the results of residual strength tests of unnotched flat specimens of 0°, ±45°, 90° CFC in Ref 729 and of 0°, ±45°, 90° GFC in Ref 254.

Sendeckyi further points out that with numerous matrix cracks in ±45° layers the transverse residual tensile strength of 0°, ±45° composites will decline compared with the static transverse tensile strength, whereas these cracks will not reduce the static longitudinal tensile strength[517].

3.2.2 Effect of laminate

In uni-directional composites repeated stresses in the fibre direction do not produce a drop in residual strength[530,765]. Loads which severely stress the matrix and the fibre/matrix bond cause damage (matrix cracks, de-bonding) which, *eg* in uni-directional composites reduce the residual bending strength[144] and in ±45° composites the tensile strength in the 0° direction to a greater extent than in 0°, 90° composites[209]. As in the above examples, the differing degree of stress on the matrix and fibre/ matrix bond also explains the difference in residual strength between unnotched specimens of 0° CFC and 0°, ±45°, 90° CFC under axial load in the 0° direction. As shown by a comparison of the mechanical properties in Fig 3.2, a significant drop in tensile strength after preceding fluctuating tensile stress occurs only in the multi-axial composite.

Corresponding to the differing development of damage in notched angle ply specimens with and without 90° layer (see section 4), the residual strength is also different. While in 0°, ±45° composites without a 90° layer residual tensile strength after preceding cyclic loading in the fluctuating tensile range is greater in CFC[113,517,732], and in BFC is either the same[717] or greater[625] than the static tensile strength, after a sufficiently long duration of stress it becomes less than the static tensile strength in 0°, $\pm 45^{\circ}$, 90° composites. As a function of the load cycle the comparison of Fig 3.3 (composite with 90° layer) with Fig 3.4 (composite without 90° layer) shows the different development of damage and correspondingly different change in residual strength. The increasing compliance of the specimen in both cases is worth noting; however it is less in the composite without a 90° layer in accordance with the extent of damage. In composites with a 90° layer, damage leading to fracture extends over the width of the specimen, while in CFC it emanates from the outer edges of the specimen, Fig 3.3[732] and in BFC increases from the edge of the hole towards the edge of the specimen[717]. In composites without a 90° layer faults propagate from the edge of the hole along the 0° fibres which border on the hole. Section 4 deals in detail with development of damage in notched specimens.

3.2.3 Effect of notch

Among the most striking deviations from the behaviour of metals is the increase in residual tensile strength of notched fibre composite specimens with the duration of cyclic stress, compared with static tensile strength. The literature contains relevant examples in Refs 113, 324, 508, 517, 625, 717 and 732, relative test results are shown in Figs 3.3 to 3.5. Fig 3.5 shows residual strength after flight-by-flight loading for notched and unnotched BFC specimens, and Figs 3.3 and 3.4 the effect of single-stage cyclic stress on residual strength and modulus of elasticity, in connection with the damage arising during initial loading. As frequently observed[508,517,625,717,730,732], longitudinal cracks occur in the matrix along the 0° fibres bordering on the root of the notch under repeated loading. The formation of these longitudinal cracks and zones of damage in the root of the notch are clearly connected with the observed increase in strength. For example, it was observed in Ref 517 in CFC specimens with a surface notch that residual strength becomes greater than static strength as soon as the zone of damage caused by repeated loading exceeds that occurring under monotonic load. Tests on GFC specimens with an edge notch[508] also reveal this dependence on the size of damage. Zweben shows on a model[114] that shear cracks along 0° fibres generate a release of stress in areas of concentrated stress by redistributing it. As is also confirmed experimentally[625], stress concentrations reduce as a function of the length of the longitudinal cracks and the laminate structure[114].

Reifsnider[782] introduces an image for damage development which characterises damage in two categories, a 'wear-in' and a 'wear-out' process. The 'wear-in' process comprises those phenomena occurring in the material under stress which increase its load capacity (residual strength), *eg* development of damage at a hole. The 'wear-out' process comprises those phenomena occurring in the material under stress which reduce its load capacity, *eg* fibre breakage, matrix cracks. Naturally not all events contribute exclusively to the 'wear-out' or 'wear-in' process. Many events which reduce stress concentration at the hole reduce at the same time the load capacity of the material in this area. This model image is presented in Fig 3.6.

The 'wear-in' and 'wear-out' processes are damage defined as such by their effect on the residual strength (load capacity) of the fibre composite. At the same time, however, stiffness drops monotonically with increase in damage (fibre cracks, matrix cracks, delamination, etc). The 'wear-in' - 'wear-out' mode model does not cover this drop in stiffness so it cannot be used in cases where loss in stiffness of the structure is a criterion of failure.

3.2.4 Effect of number of cycles and load level

Initial cyclic stresses are applied in different ways in residual strength tests:

(a) with different numbers of cycles at one level[49,113,530,732],

(b) with constant numbers of cycles but at different levels[202,205,209,508],

(c) with constant cycle ratio N_V/N_B at different levels[224,391,533], where N_V is the number of cycles of initial stressing and N_B the number of cycles to fracture, and

(d) with numbers of cycles of constant survival probability, *eg* $P_{\ddot{u}}$ = 90% in Ref 729 at different levels.

In initial stress type (a) the level can be above or below the endurance strength and thus determines the effect of the initial stress. In initial stress type (b) the load level of the initial stress is limited by the fatigue strength at the proposed number of initial stress cycles. Initial stress tests with defined ratios such as the number of cycles of initial stress to mean number of cycles to fracture (test type (c)) or with number of cycles of constant survival probability (test type (d)) supply comparable results. Test results on unnotched

GFC specimens[50,224] show, first, that for constant N_V/N_B the drop in residual strength increases with reducing stress levels and, secondly, as expected the drop in residual strength increases with increasing ratio N_V/N_B. This finding is illustrated in Fig 3.7.

If during initial stressing fatigue fractures occur due to a high value of N_V/N_B, the following consideration should be included in the interpretation of the test results according to Ref 779.

It is assumed that the individual specimen takes the same order in fatigue strength as in static strength. The 'weakening' (wear-out) of individual specimens due to fatigue stress will then on the one hand lower the mean residual strength and, on the other hand, 'elimination' of weak specimens through fatigue fracture in initial stressing will raise average residual strength. Consequently the event predominating in the test ('weakening' or 'elimination') determines the difference between the static strength and the mean value of the residual strength of the specimens not fractured during initial stressing. In principle, 'weakening' preponderates if N_V/N_B values are low, and 'elimination' if N_V/N_B values are high.

3.2.5 <u>On prediction of residual strength (see also sections 5 and 7)</u>

Arithmetical methods of prediction are based on the models referred to above. Neither the so-called 'wear-out' concept[113] nor other concepts[613,728,779] agree satisfactorily with experimental results. The 'wear-out' concept dealt with most often in the literature is based on model ideas which according to experience apply only to unnotched components, *ie* where the residual strength drops monotonically. In these cases there are some reports of good agreement between theory and experiment[161,778,784]. The 'wear-out' concept is based on the following three assumptions:

(i) Growth of damage is defined by:

$$\frac{dC}{dt} = MC^r$$

where
- C = size of damage zone
- t = time
- M = $AD^r \sigma_{max}^{2r}$
- A = constant
- D = effective compliance
- r = growth rate exponent
- σ_{max} = maximum stress

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(ii) The fracture criterion is defined by:

$$K = \sigma\sqrt{C}; \qquad K = \text{toughness} .$$

(iii) The mode of damage remains constant.

The last assumption is an essential condition for correlation of the two equations in the concept. The fact that these assumptions apply only with reservations to fibre composite materials when, for example as reported in section 4, local instabilities lead to fracture through delamination, raises justified doubts as to the applicability of these methods of prediction to fibre composite components. At present, therefore, conservative inspection intervals are laid down for fibre composite components[551], the length of which is decreed by empirical means, *eg* after residual strength tests of components in operation which can be removed and replaced easily[551,727]. Further concepts for the residual strength prediction of damage fibre composites with the aid of methods based on linear elastic fracture mechanics are introduced and discussed in section 5.

3.3 Mechanical properties after impact of objects

Fibre composite materials are used for fan and compressor blades as well as for aircraft structures which are exposed to impact of hard or soft objects.

In practice impact of hard objects includes:

- gravel and hail,
- falling tools,
- rough handling,
- ejected engine parts,
- bombardment with small calibre weapons, and
- striking rocket parts.

Birds represent a common impact of soft objects. The energy which is transmitted to a material on impact can cause elastic and non-elastic deformation. According to tests by the RAE[421] CFC and GFC are not inferior to metals in regard to their capacity to absorb energy elastically, GFC is actually superior to metals in this respect. As regards non-elastic absorption of energy there is a marked difference between metals and fibre composites:

- metals can undergo plastic deformation before fracture,

- modern fibre composites such as CFC consist of brittle fibres in a brittle matrix and can therefore absorb energy non-elastically only by fracture phenomena.

In these fracture phenomena local longitudinal cracks are formed[421,643,772,780], *ie* cracks parallel with the fibres and transverse cracks, *ie* cracks across the fibres (fibre fracture). Naturally local damage impairs the mechanical properties of the composite. After impact of an object on a fibre composite component different demands may be imposed, depending on the purpose for which the component is used, including:

(a) high energy absorption capacity, *ie* total impact energy should be absorbed,

(b) absorption of a large part of the impact energy before damage occurs, and

(c) high residual strength and stiffness after impact.

To demonstrate (a) conventional resilience tests are suitable and for (b) falling weight and shot tests. The results of tests from the literature which concern (b) and (c) are discussed below. These are tests aimed at establishing the remaining mechanical properties after impact. For investigations on the behaviour of fibre composites at and after impact of an object on the surface of fibre composite components (impact resistance) the impact is simulated in various ways in the laboratory. Falling weight tests are described in Refs 106, 421, 533, 643, 771 and 780 and tests with firing equipment in Refs 108, 146, 186, 421, 423, 733, 773, 774, 777 and 780. Impact energy is varied by changing the weight in falling weight tests and by changing the speed of the projectile in shot tests. There is no standardised procedure at present for carrying out falling weight and shot tests.

3.3.1 Effect of impact energy

A diagrammatic presentation of the drop in residual strength and progression of damage over the speed of impact is shown in Fig 3.8. No drop in residual strength is observed where impact energy is low. The range of this impact energy (Range I) in which no damage is produced may be very small after impact of hard objects on brittle materials (materials without plasticising capacity)[186]. In contrast, it may be very wide after impact of small soft objects, such as hard rubber spheres[186], see Fig 3.9. This range of impact energy depends also on plate thickness and material. It is wider for thicker plates[643], and for less brittle materials[186,421,733,771,777,780]. For example, Range I is wider for metal materials generally used in aircraft construction and for titanium matrix composites than for aluminium matrix composites[146,733,777]. In the case of composites with plastic matrix it is wider for Kevlar fibres and Kevlar-carbon hybrid composites than for pure carbon fibre composites[421,780], as shown in Fig 3.10.

Once damage occurs (chiefly cracks) tensile strength drops rapidly with increasing impact energy to a minimum value (Range II, Fig 3.8). The greatest drop in strength coincides with the maximum extent of damage (Range III). In Range III the impulse transmitted to the plate is greatest and it causes the greatest damage, *ie* the largest cracks or delaminations[421,780], see Fig 3.11. At higher impact speeds (Range IV) the object penetrates the plate completely, and the remaining hole diameter is virtually independent of the impact energy[774] and thus the residual strength has reached a constant value. Results of comparative tests show that material parameters such as stress-strain behaviour, fibre-matrix bond, mixing of layers of different fibre materials also have a strong influence in Ranges II to IV on residual strength. It is a general rule for composites that residual strength in these ranges increases with rising breaking elongation of the fibre, for this reason residual strength behaviour of CFC with HT fibre is better than that of CFC with HM fibre and glass and Kevlar composites are better in this respect than CFC. Hybrid composites of CFC with GFC or KFC layers laminated to them therefore have higher residual strengths than pure CFC composites, see Fig 3.10.

3.3.2 Effect of loading

The impact energy at which the greatest drop in residual strength occurs varies with the material and the type of stressing (tensile, bending, shear)[421]. Prestressing the target plate has a considerable influence on the minimum value of residual strength. As results in Ref 186 and Fig 3.9 show, the minimum value (Range III) gets smaller with increasing prestressing until finally the prestressing suffices to cause fracture of the plate on impact[733,773].

In Range IV, in which the impact object penetrates the plate, the brittle behaviour of modern fibre composite materials leads to clean punctures in comparison with metals, while the size of the hole corresponds to the impact object, *eg* to the diameter of the striking steel spheres[733]. The residual strength of fibre composite specimens with impact puncture is therefore equal to the strength of the notched specimen with a drilled hole.

In Range II the residual fatigue strength also drops in accordance with residual tensile strength. Comparative tests between boron-aluminium and Ti 64 show a far smaller decline in residual strength for the titanium alloy[146]. In this range the drop in compressive strength is also greater than that of tensile strength. For instance in Ref 533 with barely visible external damage twice the drop in compressive strength was observed in $[0, \pm 45, 90]_{3s}$ CFC. Comparison with the strength of notched specimens with different sized holes showed that,

> in tension the residual strength of the specimen with impact damage is as great as the strength of a specimen with a 3mm hole, while in compression it is even less than the strength of a specimen with a 25mm hole (specimen width 76 mm).

The difference between residual fatigue strength under varying tensile load (R = +0.05), alternating load (R = -1.0), or varying compressive load (R = +10.0) was of the same order, see Fig 3.12. This effect of the type of loading is again based on the interaction of the predominant type of damage (delamination, debonding, matrix crack, fibre fracture) and the type of stressing (tensile, compressive, bending, shear). This relationship and its importance for determining the damage tolerance of fibre composite components was discussed in section 3.1.

3.4 Summary

The essential findings on the mechanical properties of fibre composites in a pre-damaged condition are enumerated below. Damage after initial stressing and that caused by the impact of objects may be of different types. In determining damage tolerance it is therefore necessary to note that residual strength, residual stiffness and other properties to be tested vary with type and direction of loading, since this affects the damage incurred.

The most important results from investigations of the pre-loading of fibre composite materials are:

- in notched specimens the residual strength can increase after repeated initial loading,
- in open hole specimens of CFC angle ply with 90° layers residual strength rises at first with increasing damage at the hole, but then drops with the start of damage at the edge of the specimen; without the 90° layer residual strength rises to the static strength of the unnotched specimen,

- if the notch is sharp residual strength rises, as for the open hole specimen, but the static strength of the unnotched specimen is not achieved[732],
- in unnotched specimens the drop in residual strength is greater at low stress levels than at high,
- stiffness drops monotonically in unnotched and notched specimens with increase in damage.

The most important results from investigations of fibre composite specimens after impact of hard and soft objects are:

- compared with metals the drop in residual strength is greater,
- the specific tensile residual strength after impact is greater in fibre composites than metals[773],
- important parameters are:

  $\sigma - \varepsilon$ behaviour of materials, plate thickness, hardness of striking object, fibre/matrix bond,
- the residual strength in Range III (Fig 3.8) of plates stressed on impact is less than that of unstressed,
- residual compressive strength drops more in Range II (Fig 3.8) than residual tensile strength,
- the residual strength of the composites increases with rising breaking elongation energy of the fibres.

3.5 References

In this section the references evaluated in the preceding sections are classified according to contents. Papers mentioned in the text are underlined.

Section 3.2

Mechanical properties after preliminary loading:

49, 224, 391, 523.

Section 3.2.1

Effect of type of stressing in preliminary loading and residual strength test:

49, 96, 144, 202, 205, 209, 224, 254, 523, 530, 729, 737.

Section 3.2.2

Effect of laminate:

204, 209, 391, 530, 717, 732, 783.

Section 3.2.3

Effect of notch:

113, 324, 508, 517, 547, 625, 717, 727, 732, 782.

Section 3.2.4

Effect of number of cycles and load level:

49, 50, 202, 205, 209, 224, 391, 508, 530, 729, 732, 779.

Section 3.2.5

On prediction of residual strength:

113, 161, 613, 728, 778, 784.

Effect of preliminary load and environment:

218, 517, 530, 727.

Effect of preliminary load and preliminary damage (impact):

186.

Drop in stiffness:

141, 184, 202, 205, 218, 224, 530, 625, 717, 729, 732.

Section 3.3

Mechanical properties after impact of objects:

332, 421, 423, 643, 772, 780, 106, 108, 146, 186, 533, 643, 733, 771, 773, 774, 777, 780.

Section 3.3.1

Effect of impact energy:

106, 146, 186, 421, 733, 773, 774, 777, 780.

Section 3.3.2

Effect of loading:

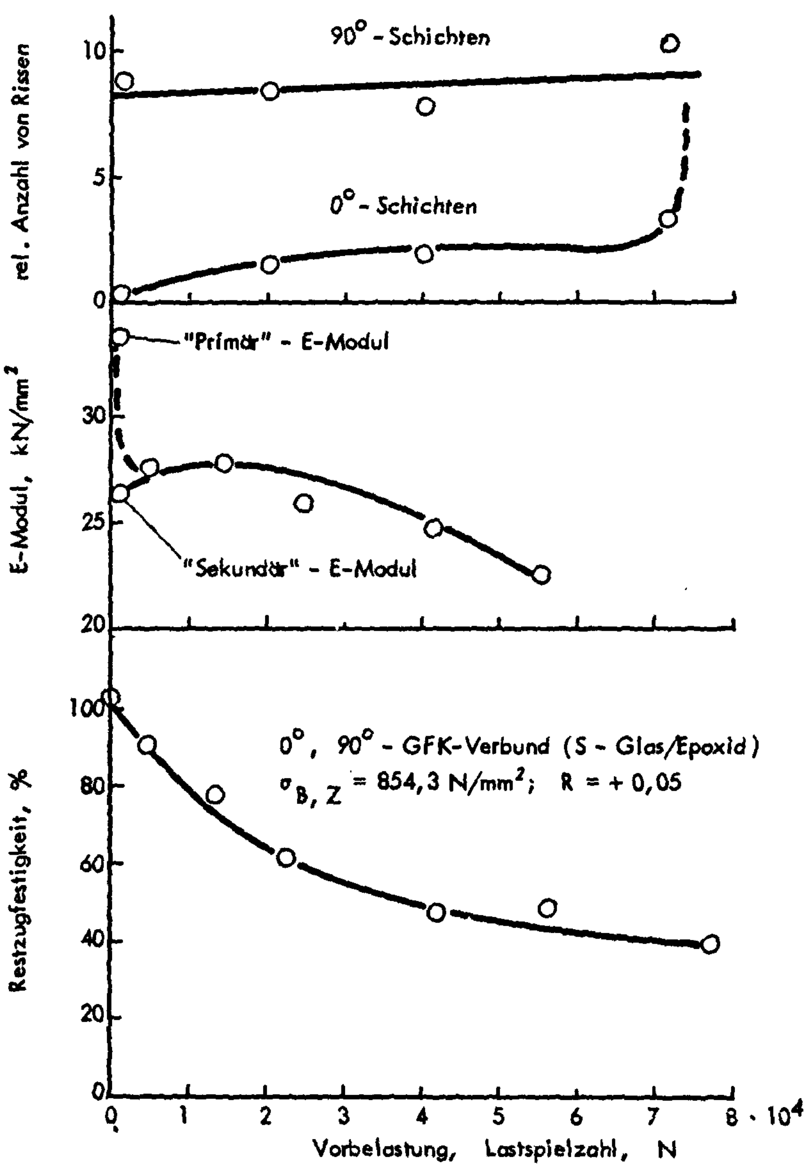
108, 146, 186, 533, 773, 733, 421.

Effect of material on residual strength after impact:

106, 108, 146, 186, 427, 643, 733, 773, 774, 777, 780.

Residual strength after lightning strike:

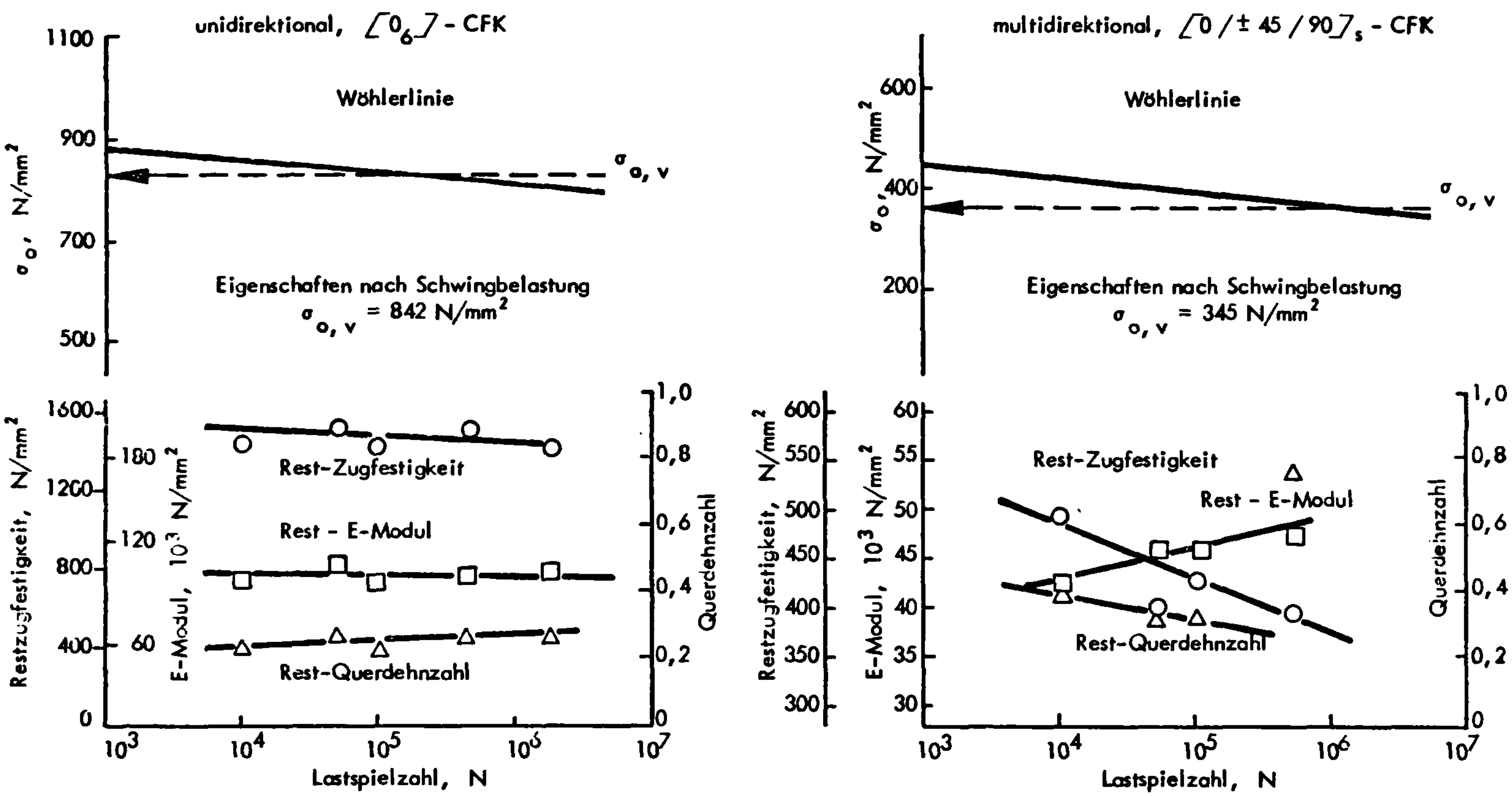
94.



Key:
Restzugfestigkeit = residual tensile strength
Rel.Anzahl von Rissen = rel. number of cracks
Vorbelastung = preliminary loading
Lastspielzahl = number of cycles

T 2

Fig 3.1 Relationship between crack density, modulus of elasticity and residual strength as a function of the number of cycles (Ref 49)



Key:
Querdehnzahl = transverse strain coefficient

Fig 3.2 Mechanical properties after cyclic loading (R = 0.1; f = 30 Hz) of uni-directional and multi-directional CFC (Ref 530)

sT 2

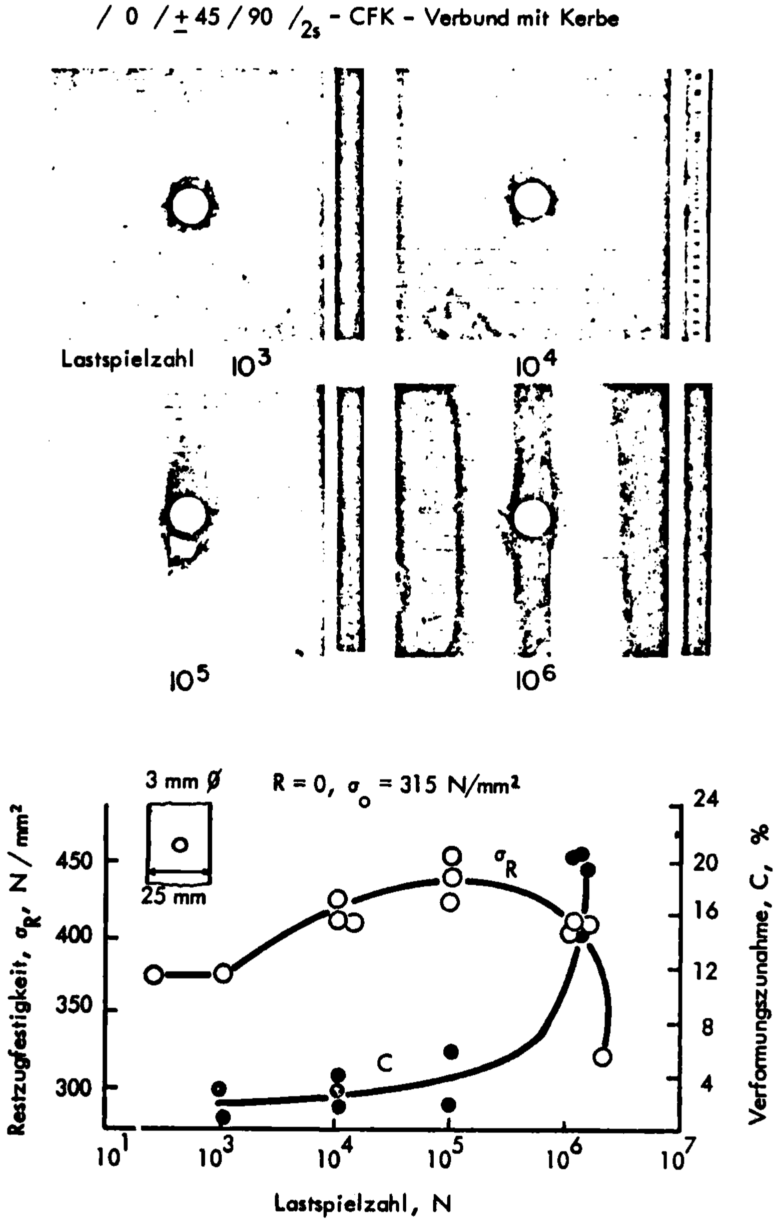
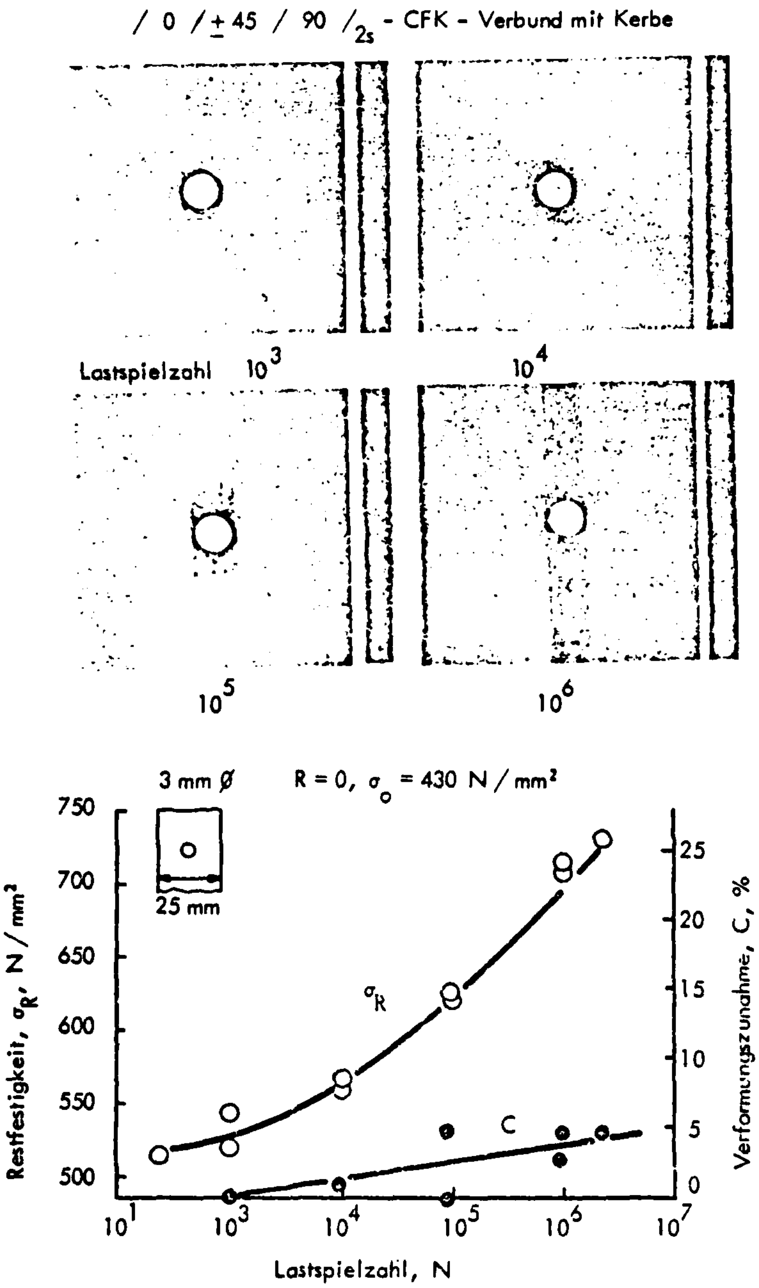
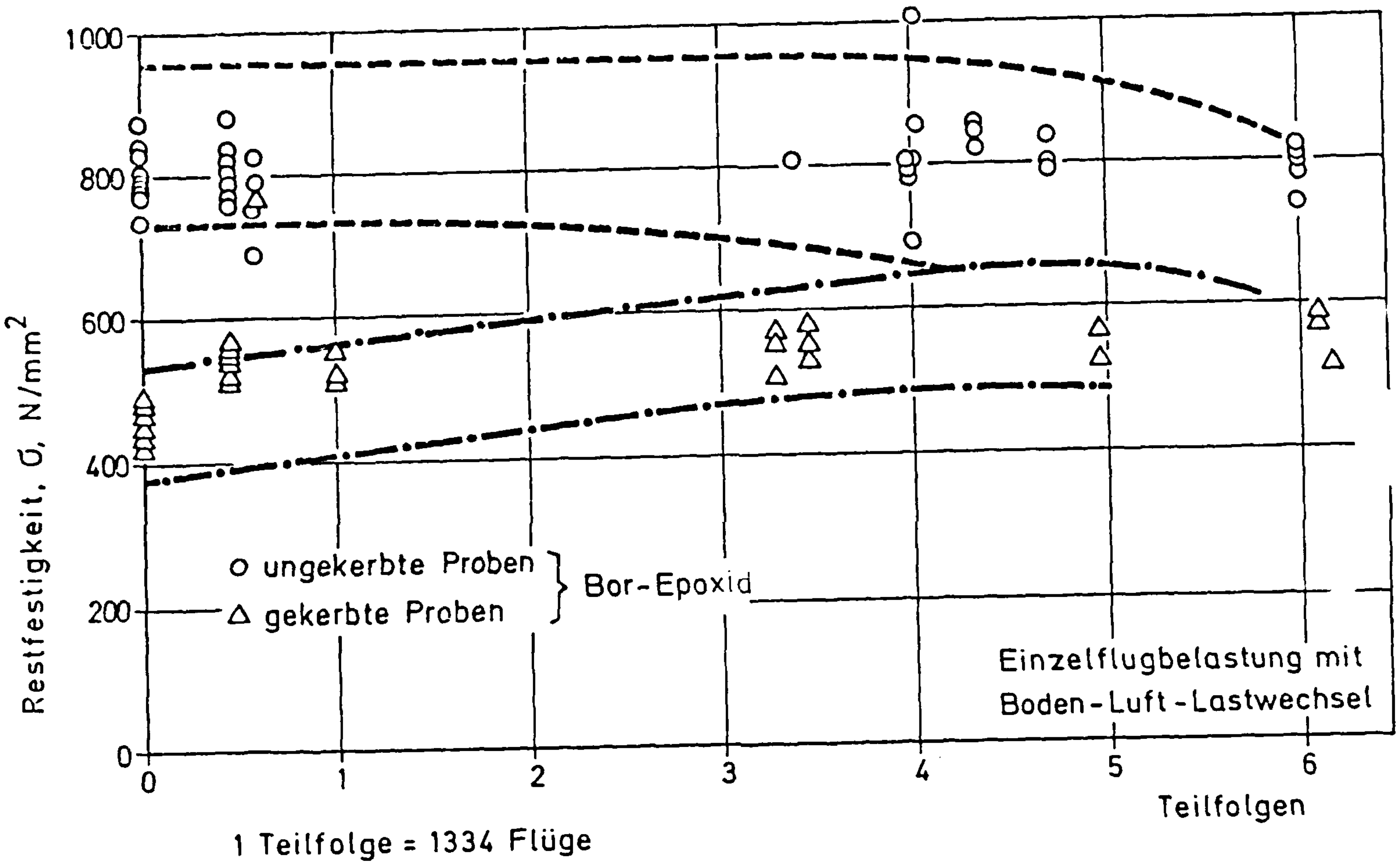


Fig 3.3 Relationship between residual strength, strain increase and development of damage as a function of the number of cycles (Ref 732)



Key:
Verformungszunahme = increase in deformation

Fig 3.4 Relationship between residual strength, strain increase and development of damage as a function of the number of cycles (Ref 732)



Key:
Einzelflugbelastung mit Boden-Luft-Lastwechsel = flight-by-flight loading with ground-air load cycle
Teilfolgen = partial sequences

Fig 3.5 Residual strength after cyclic loading (Ref 113)

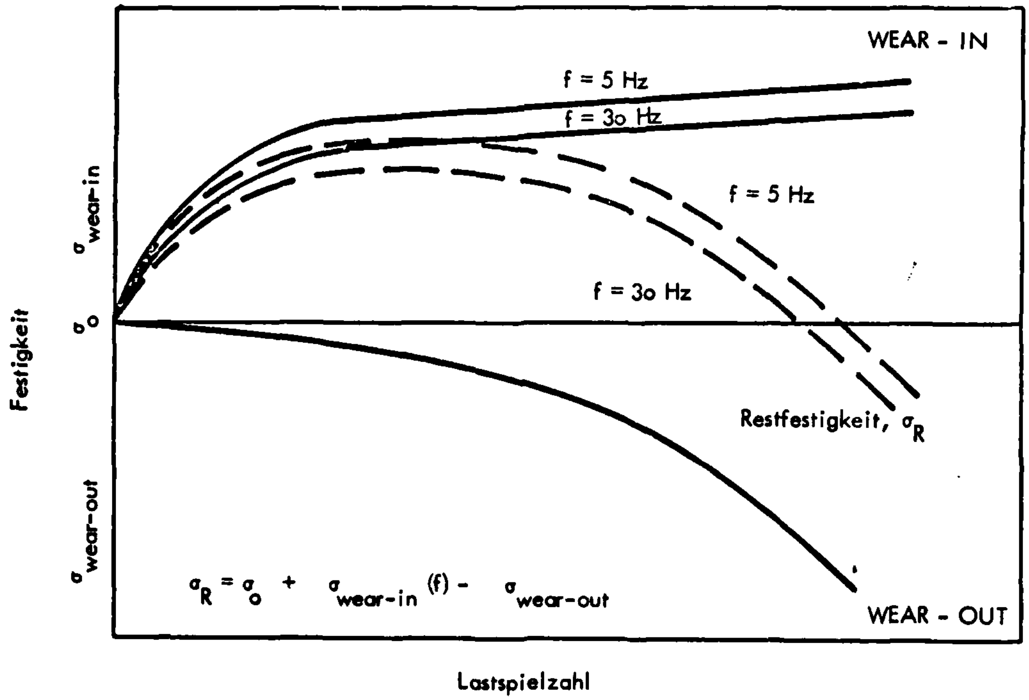


Fig 3.6 'Wear-in' - 'wear-out' model (Ref 782)

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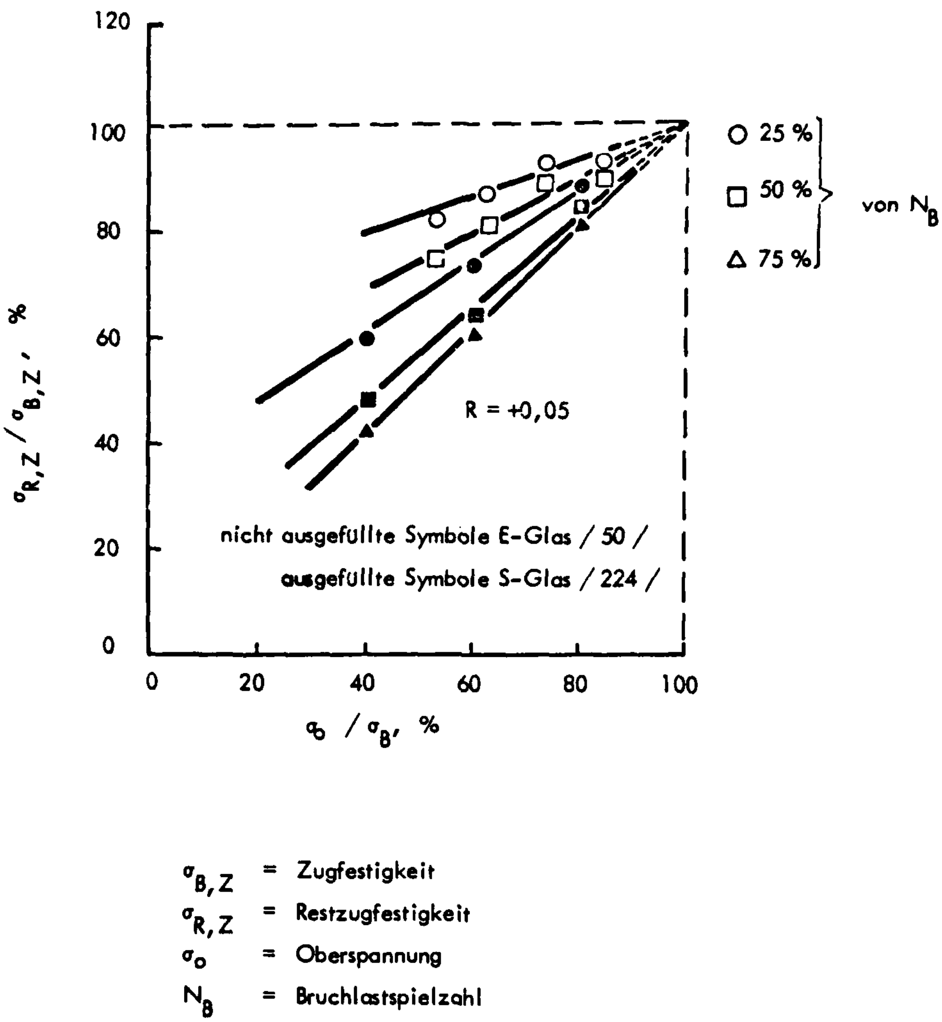
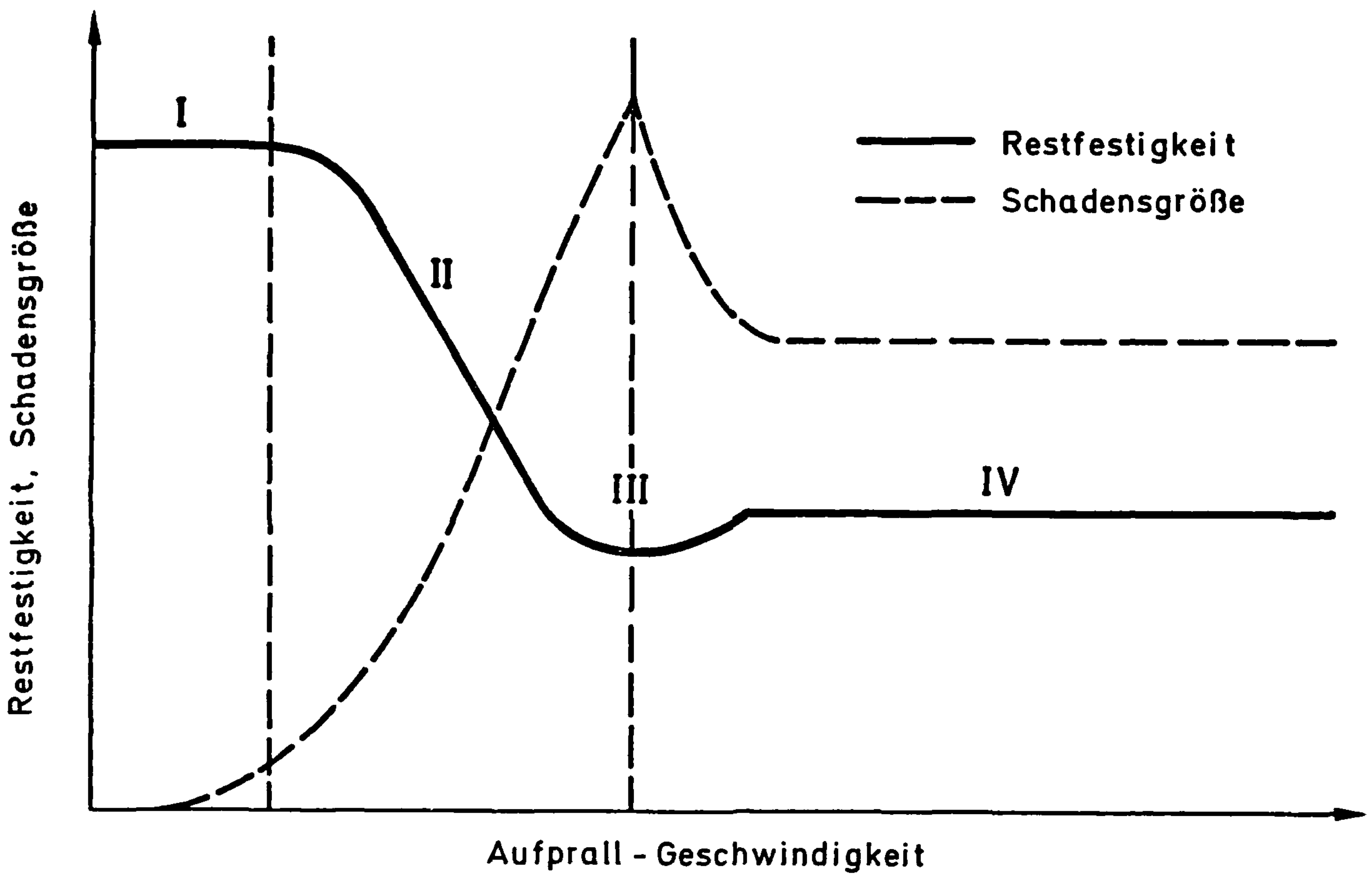


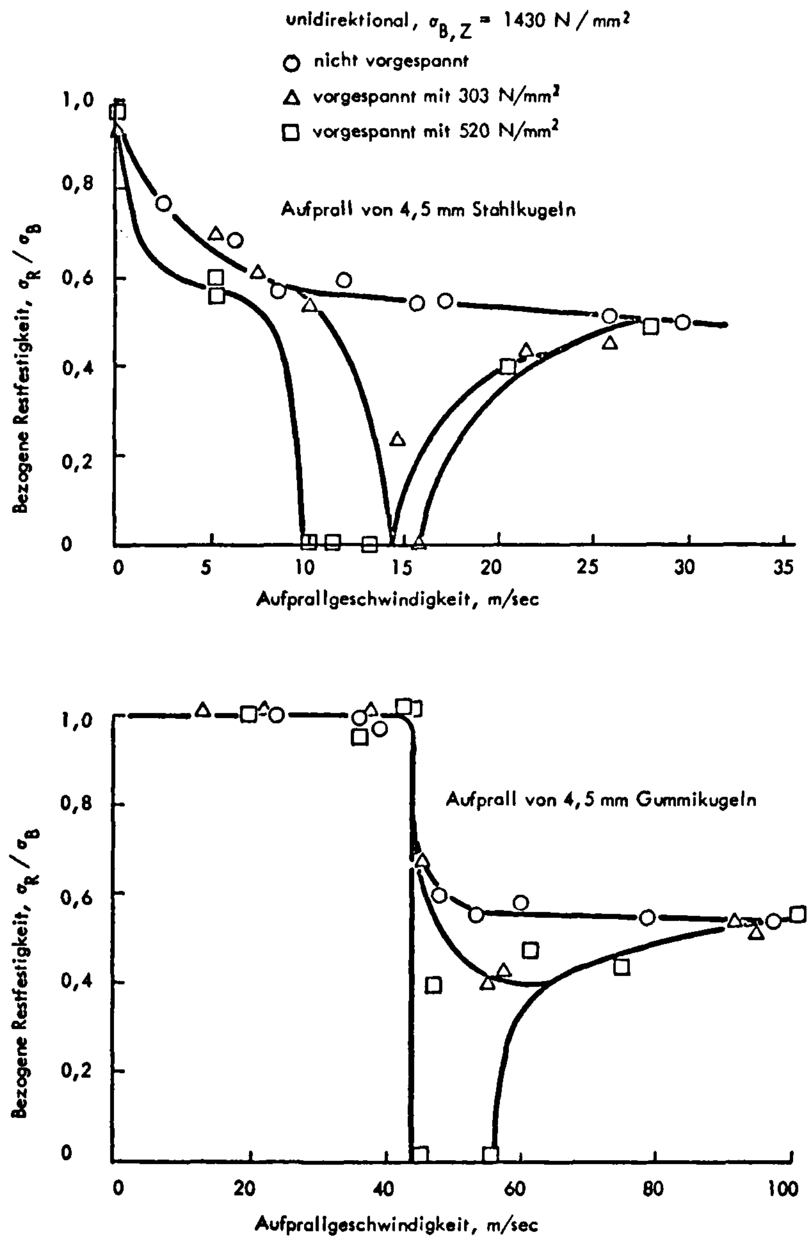
Key:
Ausgefüllte Symbole = filled symbols
Nicht ausgefüllte Symbole = open symbols
Zugfestigkeit = tensile strength
Restzugfestigkeit = residual tensile strength
Oberspannung = maximum stress
Bruchlastspielzahl = number of cycles to failure

Fig 3.7 Effect of number of cycles and load level of initial loading on residual tensile strength of 0°, 90° GFC



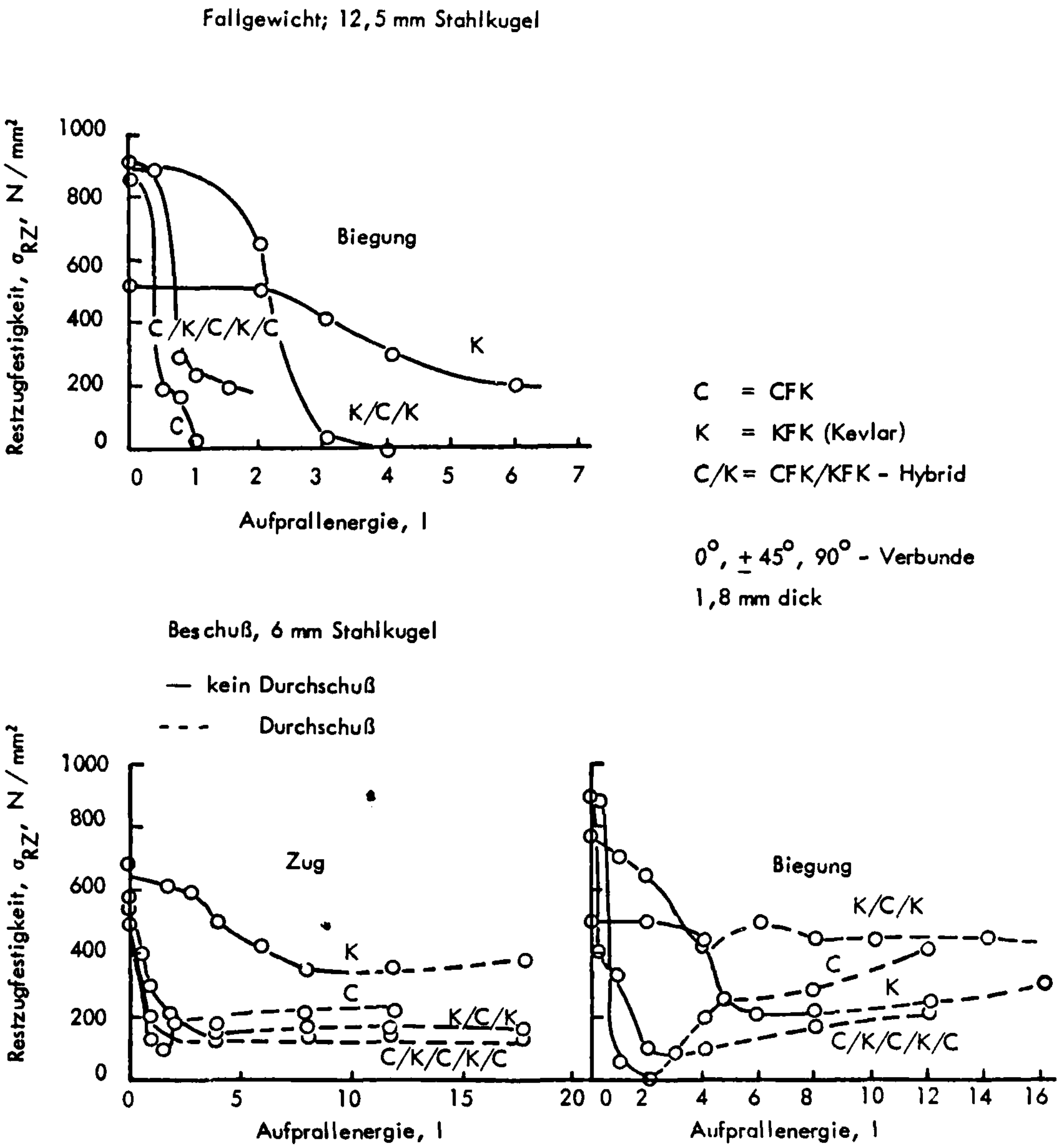
Key:
Aufprall-Geschwindigkeit = speed of impact

Fig 3.8 Residual strength and extent of damage under impact (Ref 777)



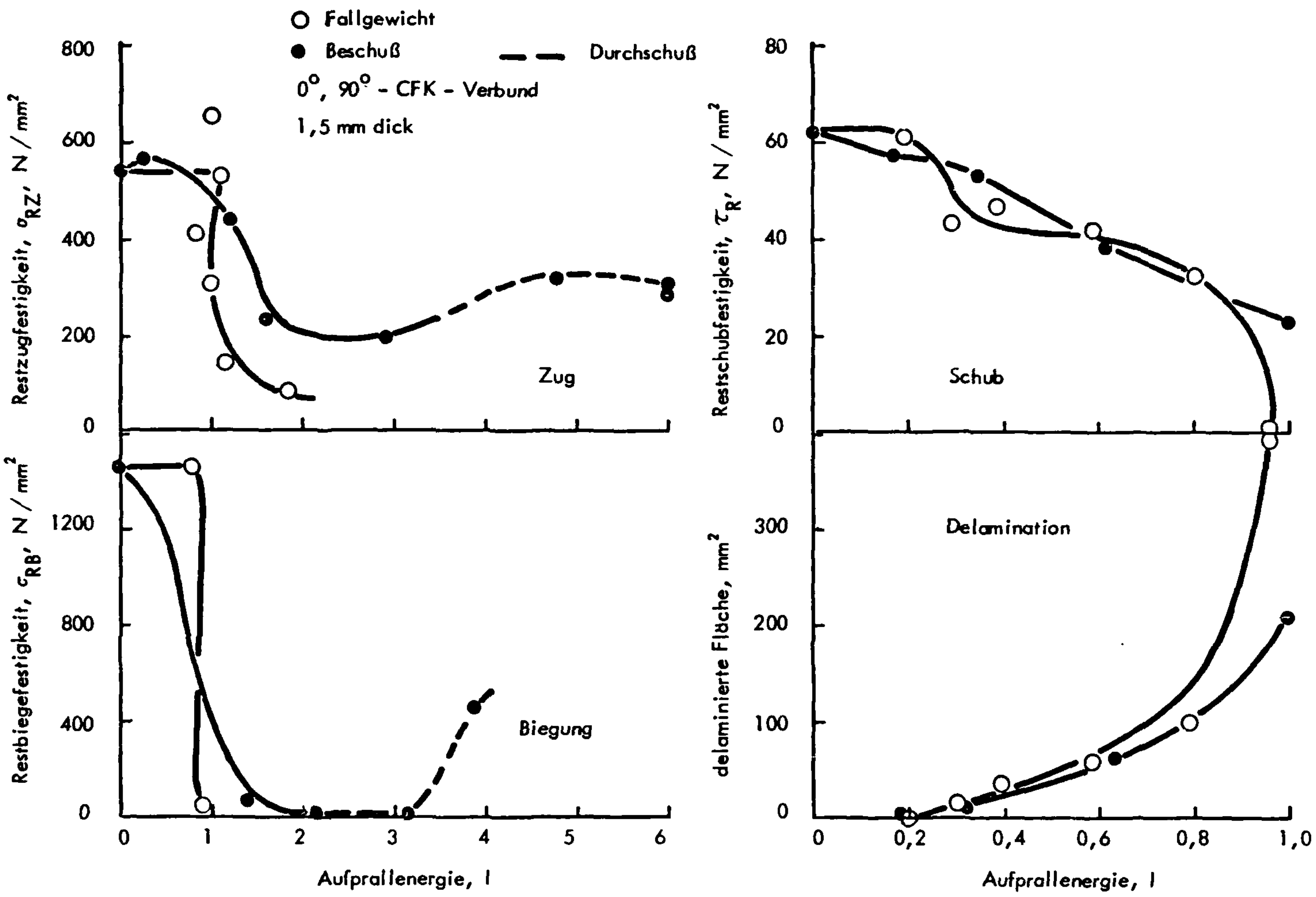
Key:
Bezogene Restfestigkeit = related residual strength
Aufprall von 4.5mm Gummikugeln = impact of 4.5mm rubber spheres

Fig 3.9 Residual strength after impact of hard and soft objects on pre-stressed and non pre-stressed boron/aluminium plates (Ref 186)



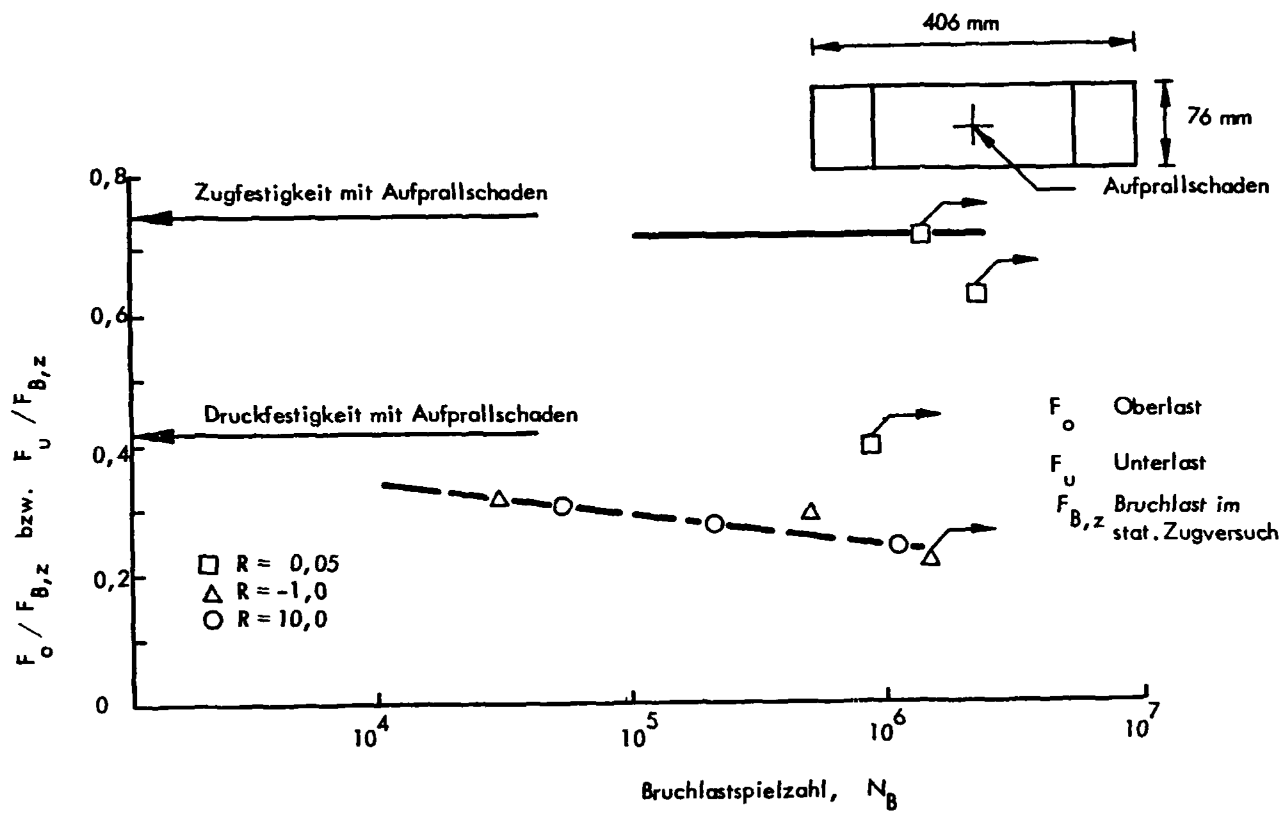
Key:
Fallgewicht; 12.5mm Stahlkugel = falling weight; 12.5mm steel sphere
Biegung = bending
Kein Durchschuss = no penetration

Fig 3.10 Effect of impact energy on the residual tensile strength of angle ply composites (Ref 780)



Key:
Beschuss = bombardment

Fig 3.11 Drop in residual strength in range III (see Fig 3.8) in conjunction with delaminated area (Ref 421)



Key:
Aufprallschaden = impact damage
Bruchlast im stat. Zugversuch = breaking load in static tensile test

Fig 3.12 Residual fatigue strength of impact-damaged CFC as a function of the R value (Ref 533)

4 DEVELOPMENT OF DAMAGE

J.J. Gerharz

Contents

4.1 Review of types of damage, indications of damage and methods of observation

Damage occurring in fibre materials as a result of loading and partly of environmental exposure lies within the micromechanical range and consists of:

- fibre crack,
- fibre fracture,
    - local transverse cracks through fibre and matrix,
- matrix crack,
    - across the layer level along the fibres,
    - in layer level, interlaminar (delamination) or intralaminar,
- interface crack,
    - local longitudinal cracks along the fibres of a layer,
    - debonding.

In the development from the initial damage to fracture, combinations of the faults mentioned always occur so that the total damage process runs on all three levels. A similar process occurs in metals in the generating phase of the macro crack within the grain boundaries[728].

Damage can also be recognised by local heating, sound emission and changes in stress-deformation behaviour which occur on loading and can be measured.

Furthermore,

- loss of stiffness,
- change in resonant frequency,
- change in damping behaviour, and
- change in residual strength,

of prestressed specimens are indications of damage.

The development of damage described in the literature with crack observations and fracture image analyses is reported below. Conclusions on the predominant type of damage and further subsidiary types of damage can be drawn from the appearance of the fractured laminate. This also makes possible inferences on the presumed development of damage, see Refs 623 and 728. The fracture faces are examined either without or with magnification.

Position and extent of local damage before fracture are ascertained effectively by the following means:

- X-ray with the contrast medium Tetrabromethane (TBE)*,
- C-scan ultrasonic test,
- thermography.

Through the penetrating contrast medium X-rays show:

- matrix cracks across the layer level, and
- matrix cracks in the layer level (delamination),
- interface cracks, and
- fibre fractures in boron fibres (without contrast medium).

C-scan ultrasonic tests discover:

- matrix cracks in the layer level (delamination),

and thermography shows the increased generation of heat at the site of damage.

Cracks on the edges of notches and outside edges of the specimen or running through the outer layer (across the layer level) can be observed visually[626,775]. Under load and with suitable lighting visual detection of such damage is facilitated. Because of the transparency of glass fibres visual observations of damage development in GFC are very productive since both matrix cracks and fibre fractures are visible. Many investigations of damage processes in fibre composites have therefore been carried out with GFC[141,166,209,224,254]. More recent investigations deal predominantly with CFC and BFC[517,523,532,623], and[625,717,258] since methods using X-ray with contrast medium and ultrasonic test with C-scan equipment have proved suitable. Additional visual observations of, *eg* the narrow sides of specimens and fracture image analyses give information on the development of damage through the thickness. Findings on the development of damage under monotonic and repeated loading of unnotched and notched specimens are summarised below.

4.2 Development of damage under monotonic stressing

4.2.1 Unnotched specimen

As mentioned in section 6, interlaminar stresses occur in angle ply composites at the edges of specimens depending on the laminate structure. If these exceed the strength of the matrix, delaminations occur at the edge of the specimens. The delaminations observed in Refs 191, 523, 547 and 626 between 90^{o} and 45^{o} layers during tensile stressing were preceded by lateral cracks in the 90^{o}

* TBE is poisonous. It can affect development of damage on extended contact with resin and fibre[730], therefore it is necessary to rinse out the penetrating contrast medium before further stressing.

layer. Starting from the ends of the lateral cracks, delamination cracks are generated at the edges; at the same time these edge cracks can alternate from one side of the 90^o layer (via the 90^o cracks) to the other side[626]. Delamination is also detected by an abrupt change in the slope of the $\sigma - \varepsilon$ curve shortly before fracture. Elongation was measured with adhesive strain gauges[547]. The angle ply composites [90^o, ±45, 0], also tested in Refs 191 and 626 with external 90^o layers showed only slight interlaminar stresses under axial load in 0^o direction and therefore did not delaminate.

Angle ply composites without 90^o layers, with 45^o layers separated by 0^o layers showed extensive delaminations between 0^o and 45^o layers after tensile fracture[728,729].

In static compression tests angle ply composites of $[0, 45, 90, -45]_2$ structure fractured from buckling of the outer layers after their delamination[728]. In the CFC mentioned so far the authors found no fibre fractures prior to the fracture of the entire composite.

However, fractures of glass fibres were observed in uni-directional GFC[141,219]. It was possible to follow the entire damage process during monotonic tensile stressing by visual observation of the illuminated specimen with an optical microscope. The following faults were visible in the order of their appearance:

(i) fractures of the weakest fibres,

(ii) microcracks at the surface of the resin, normal to the load, matrix crazing*,

(iii) rapid increase in fibre fractures,

(iv) fusion of fibre fractures by transverse crack, followed by shear cracks along the fibres, and

(v) total fracture.

Local stress due to impact of hard objects at low speed and objects at high speed and the resulting damage is reported in Refs 643, 771, 772 and 773.

4.2.2 Notched specimen

Results of damage propagation investigations on monotonic loading of notched specimens with holes cannot be found in the literature.

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* A detailed explanation of this phenomenon is contained in Refs 141 and 219.

Radiographs of tensile stressed CFC specimens with central slot show in Refs 167 and 731 the damage occurring at the ends of the slot in $[\pm 45]_{3s}$, $[0, \pm 45]_{2s}$ and $[0, \pm 45, 90]_{2s}$ composites. The radiographs were taken at different load stages during tensile stress. They show distinct differences in development of damage for the three composites. A common characteristic is:

Composite	Damage
$[\pm 45]_{3s}$	Formation of long matrix cracks along the fibres.
$[0, \pm 45]_{2s}$	Formation of shorter matrix cracks than in $[\pm 45]_{3s}$, slight delamination along ±45 and 0^o fibres.
$[0, \pm 45, 90]_2$	Formation of: - matrix cracks along 90^o fibres over the width of the specimen, - matrix cracks along ±45 and 0^o fibres, and - greater delamination than in $[0, \pm 45]_{2s}$.

For the last composite the maximum length of matrix cracks along the 90^o, $\pm 45^o$ and 0^o fibre were measured at different stages of the test and plotted against load as shown in Fig 4.1. The matrix cracks emanating from the notch root grow most rapidly in the 90^o layer.

4.3 Development of damage under repeated stressing

4.3.1 Unnotched specimen

4.3.1.1 Effect of fibre material and laminate structure

Uni-directional composites

Under axial strain-controlled loading in the varying tensile range development of damage in uni-directional GFC flat specimens was observed with an optical microscope in Ref 141. According to this three areas of different damage development can be defined on the established Woehler line, see Fig 4.2. While in Ranges I ($N < 100$) and II ($10^2 < N < 10^6$) faults which lead to failure are generated at the first increase in load but such faults are not generated in Range III ($N > 10^6$) until after numerous load cycles. Damage at the first increase in load develops in Range I as in the static tensile test. In Range II microcracks form at the surface of the matrix, from which damage to failure is propagated. This development of damage is shown schematically in Fig 4.3

In Ref 166 uni-directional GFC and CFC bars were loaded with alternating torsion under strain control up to critical losses in stiffness and critical increase in damping capacity. The matrix cracks formed could be observed satisfactorily in the illuminated GFC, but in CFC only if they were on the surface. First signs of matrix faults which lead to long cracks along the fibres under continued loading were whitening in the resin below the surface of GFC, and a fine white layer of ground resin along the line of the crack in CFC.

0°, 90° composites

In orthotropic composites delamination cracks emanating from the edge of the specimen were observed before failure in Refs 209 and 224. In a GFC with epoxide resin the predominant delamination of outer layers was greater under variable tensile range loading than variable compression range loading[209]. In GFC $[0, 90]_{8s}$ matrix cracks developed parallel to the fibres in 90° and 0° layers before the appearance of delamination[224].

Angle ply composites

The authors of Ref 254 observed the progress of damage in quasi-isotropic GFC $[0, \pm 45, 90]_s$. Depending on the loading level, they found the following under variable tensile load:

at

(a) Maximum stress > strength of the 90° layer ($\sigma_0 > \sigma_B$, 90°)

- (i) matrix cracks in 90° layer parallel with the fibres passing through layer thickness and specimen width,
- (ii) cracks in $\pm 45^{\circ}$ layers along the fibres, and
- (iii) finally delamination, and

at

(b) $\sigma_0 < \sigma_B$, 90°

- (i) cracks in the matrix when $N > 10^3$, and
- (ii) delaminations following later.

The damage observed in cases (a) and (b) and the viscoelasticity of the resin caused heating of the specimens.

After compilation of several visual, radiographic and ultrasonic observations of damage development in composites with $[(0, \pm 45, 90)_s]_2$ laminate structure Sendeckyi (AFFDL) found in Ref 523 that matrix cracks and the resultant delaminations (see also Ref 623) are the dominant mechanisms in damage development in CFC. He postulates that on loading in the variable tensile range, edge delamination cracks,

(a) occur in the first load cycle if its maximum stress (σ_0) is greater than stress (σ_D) at which the start of delamination was observed in the static test ($\sigma_0 > \sigma_D$), or

(b) occur during cyclic loading if $\sigma_0 < \sigma_D$, provided σ_0 is above the stress level (σ_{SD}) at which delamination cracks are just generated on repeated loading.

This stress σ_{SD} is presumed to be near the level at which first failure of a layer occurs, since delaminations in $[(0, \pm45, 90)_S]_n$ composites* are always preceded by the tearing of the 90^o layer, as shown by many observations. Once the weakest layer is split delamination cracks at the edge of the specimen first grow most rapidly in the direction of load and then grow more quickly across the load direction towards the longitudinal axis of the specimen. If the crack has split the specimen its parts are usually asymmetrical in structure which produces warping. The related bending stresses increase the load on the external 0^o layers and initiate fracture.

Delaminations have also been observed[626] in angle composites with lesser interlaminar stresses than in the $[0, 90, \pm45]_s$ structure. However, these occurred in the plane of symmetry between 45^o layers and at the edge of the gripping area. The reasons have not yet been discovered.

4.3.1.2 Effect of type of stressing

In a CFC $[(0, +45, 90, -45)_s]_2$ more extensive delamination and more distinct buckling of the delaminated outer layers were observed[728,729] under alternating load with a constant negative minimum stress σ_u than under variable tensile load. Fracture always occurred under compressive load, shortly after delamination cracks had been found, *ie* delamination occurred when 90% or more of fracture life had been exceeded. In contrast, significant delamination did not always occur under variable tensile stress and when it did the subsequent load cycle to fracture could be large or small[729]. The authors therefore suggest defining failure as the occurrence of delaminations under alternating load and as the fracture of the specimen under load in the variable tensile range.

* Index n = 2, 3 ...

Under loads with negative minimum stress the observed delamination cracks occurred primarily at the edge of the specimen on the plane of symmetry between the -45^{o} layers of the composite and were greater at low stress levels.

4.3.2 Notched specimens

4.3.2.1 Effect of shape of notch

The notch most frequently examined is a drilled hole situated in the centre of the specimen. The damage processes observed in holes on repeated cyclic load are reported in Refs 170, 258, 523, 532, 625, 717 and 732. Results of damage development investigations in CFC specimens with slots at the edges and in the centre of the specimen across the load direction are dealt with in Refs 730, 732 and 517 (surface crack). The following differences in the development of damage were observed:

In composites $[(0, \pm 45, 0)_s]$ the damage (matrix cracks under $\pm 45^{o}$ and delamination cracks) with symmetric edge slots grows directly into the nett cross-section if a tensile-compressive load is applied; in contrast, for central notches (slot and hole) and, under variable tensile load, also for edge slots, it grows primarily in areas above and below the notches; longitudinal cracks (matrix cracks along the fibres of the 0^{o} layer) first emanate from the notches, between which further damage is generated[517,730,732,532]. In composites with additional 90^{o} layers, on the other hand, the damage grows directly into the nett cross-section even with central slots, while in the case of holes damage first extends in the same way as in composites without 90^{o} layers above and below the hole, although here too 90^{o} matrix cracks appear very early which run from the 0^{o} matrix cracks to the edge of the specimen.

4.3.2.2 Effect of fibre material and laminate structure

Observations in the literature show that development of damage in notched specimens with holes depend on the fibre material. For comparisons between GFC and CFC with the same laminate structure of 0, ±45 and 90^{o} layers there are radiographs in Ref 170 which were taken during the first four load cycles. Differences refer to 0^{o} matrix cracks and delamination cracks at the edge of the hole. In CFC the composite delaminates between the 0^{o} matrix cracks which were generated previously along the fibres at a tangent to the hole. In GFC many 0^{o} matrix cracks grow in the centre of the specimen emanating from the edge of the hole, and layers delaminate uniformly around the hole.

Observations of damage development during cyclic tensile load are available for comparison between BFC and CFC in Refs 258, 717, 523 and 732 for high numbers of load cycles. In Fig 4.4 the damage processes compiled from the above for BFC are compared with those for CFC angle ply composites. Comparison of the damage processes shows in the main:

- in BFC delamination cracks are restricted to relatively small areas,

while

- in CFC relatively large delaminated areas can occur.

In composites with 90° layers the damage emanating from the edge of the hole extends

- in BFC across the load direction (directly into the nett cross-section), and
- CFC primarily in the direction of load.

4.3.2.3 Effect of type of stressing

So far damage processes have been described in notched specimens subjected to cyclic stress in the tensile range. The literature[532,732] also reports on development of damage in notched specimens under alternating compressive stress. It has been observed that in CFC with 0° and ±45° layers delamination cracks first occur, as under variable tensile stress, between the outermost 0° layers and neighbouring layers. As in unnotched specimens, however, further damage is more intensive, *ie* delamination occurs between more layers than under variable tensile stress. Radiographs of 0, ±45° fibres in the delaminated area are described in Ref 732. Fracture of the laminate was initiated by the buckling of delaminated layers as soon as their extension in the direction of load had become critical[532]. A pronounced drop in stiffness under compressive loading, greater than under tensile loading, was found by the authors of Ref 732 after repeated loading with negative minimum stress in CFC specimens $[(0, \pm45, 0)_{2s}]$ with edge slots. Refs 517 and 730 also report on the development of damage under flight-by-flight loading of notched CFC. Radiographs were taken during the test after various loading periods. On these the maximum spread of visible damage was measured[730] and plotted against number of cycles. This presentation shows major differences in the growth of damage between $[\pm45]_{3s}$ on the one hand and $[0, \pm45, 90]_{2s}$ and $[0, \pm45]_{2s}$ on the other, see Fig 4.5. While in ±45° laminates matrix cracks along the fibres grew to 10 mm in length before fracture

LT ...5

occurred, in $[0, \pm 45]_{2s}$ and $[0, \pm 45, 90]_{2s}$, *which had 1.5 times the fracture life* of the $[\pm 45]_{3s}$, the maximum extent of damage (matrix cracks along 0^o fibres) was less than 3.5 mm. Flight-by-flight tests in Ref 517 were performed on $[(0, \pm 45, 0)_s]_3$ composites with a surface notch at elevated temperatures. Radiographs and ultrasonic scans (C-scan) show that the extent of the damage increases with the test temperature.

The extent of the delaminations shown in Fig 4.6 serve as an example.

4.4 Summary

To summarise, the following can be established:

- Methods of *damage observation* used successfully are:

  (a) visual observation,

  (b) observation with optical microscope on illuminated CFC,

  (c) X-rays with Tetrabromethane (TBE) contrast medium,

  (d) ultrasonic test with C-scan equipment,

  (e) X-rays with soft radiation and micrographs under the electron scan microscope.

- *Damage* recognised by these methods of observation is

  with (a) matrix cracks on the surface, delamination at edges, buckling of delaminated external layers,

  with (b) matrix cracks, matrix crazing, fibre breaks and delamination in GFC,

  with (c) spread of matrix cracks along fibres and extension of delamination in CFC,

  with (d) extension of delaminations in CFC,

  with (e) fibre breaks (X-ray) and matrix cracks (electron scan microscope).

- Of the methods primarily used for CFC ultrasonic testing (C-scan) is easier than radiography with a contrast medium,

- radiographic and ultrasonic methods show damage in two planes only, but damage development in fibre materials is three-dimensional. Additional visual observations of the narrow sides of hole and outside edges of the specimens and fracture and micrographic analyses can give information on the progress of damage through the thickness,

- the formation of matrix cracks (across the layer plane) and the generation of delaminations from matrix cracks are the dominant processes of damage accumulation under repeated stressing,
- delaminations begin primarily at the outside edges of specimens and edges of notches because interlaminar stresses are high there. Once delamination has started the damage process is accelerated until total fracture occurs,
- in notched specimens damage always starts at the notch root and in unnotched specimens always at the edge of the specimen,
- development of damage emanating from notches is affected by the fibre material,
- matrix cracks at the notch root which run along the fibres in the 0° layer, blunt the notch,
- in unnotched specimens with 0, ±45 and 90° layers development of damage under monotonic load is similar to that under repeated load. In notched specimens damage at the notch root is propagated along the 0° fibres under repeated load. Under monotonic load propagation of damage along the $\pm 45^{\circ}$ or along the 90° fibres (if present) is dominant,
- in angle ply composites with a central hole notch *development of damage to fracture* is different in composites with 90° layers from that in composites without 90° layers,
- compressive stresses under monotonic and repeated load cause more extensive damage than tensile stresses; buckling of delaminated layers can initiate the fracture of the laminate.

4.5 References

In this section the references evaluated in the preceding sections are classified according to contents. Papers mentioned in the text are underlined.

Section 4.1

Review of types of damage, indications of damage and methods of observation:

141, 166, 209, 224, 254, 258, 517, 523, 532, 623, 625, 626, 717, 728, 730, 775.

Chapter 4.2.1

Monotonic load: unnotched specimen:

24, 141, 191, 219, 503, 523, 547, 626, 643, 728, 729, 771, 772, 773.

Section 4.2.2

Monotonic load: notched specimen:

141, 167, 219, 517, 730, 731.

Section 4.3.1

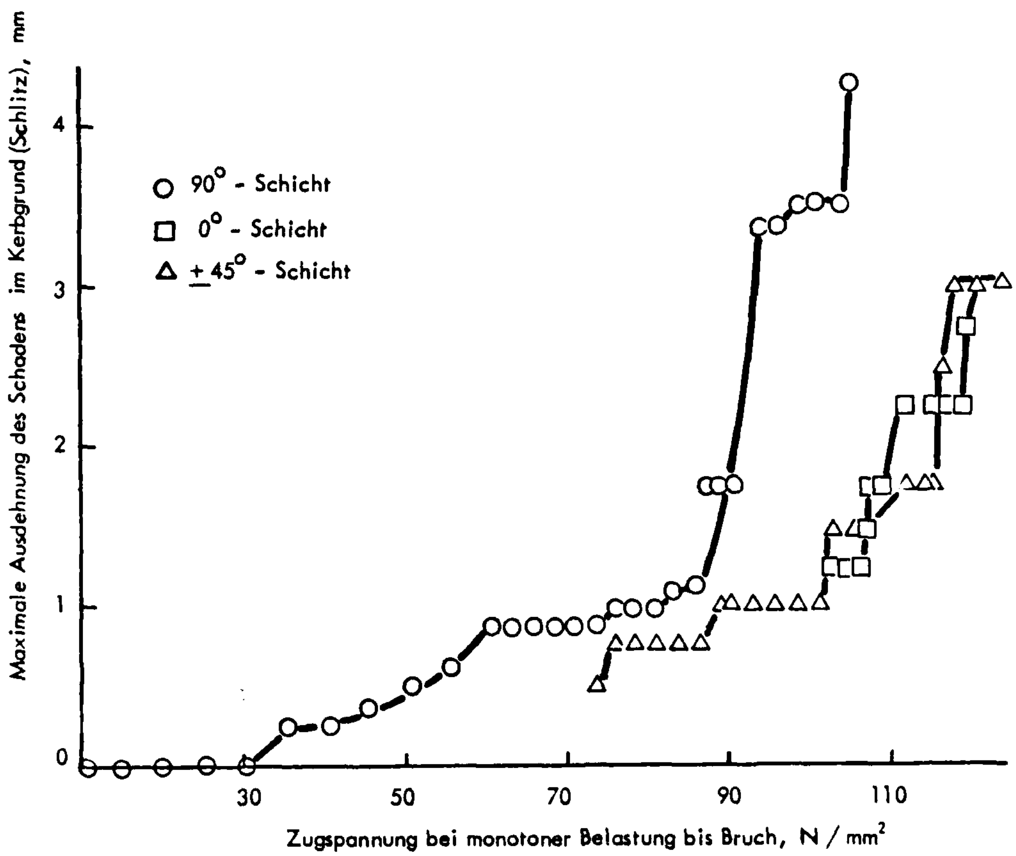
Repeated load: unnotched specimen:

49, 95, 141, 166, 168, 201, 208, 209, 210, 216, 224, 226, 254, 523, 538, 623, 626, 728, 729.

Section 4.3.2

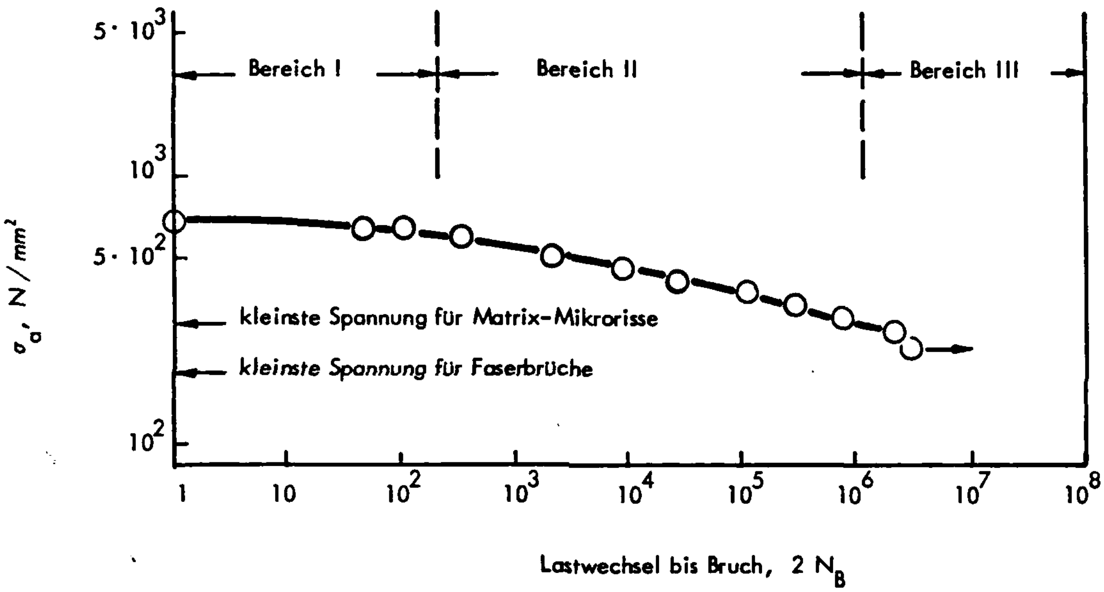
Repeated load: notched specimen:

170, 220, 233, 258, 517, 523, 532, 625, 717, 730, 732, 775.



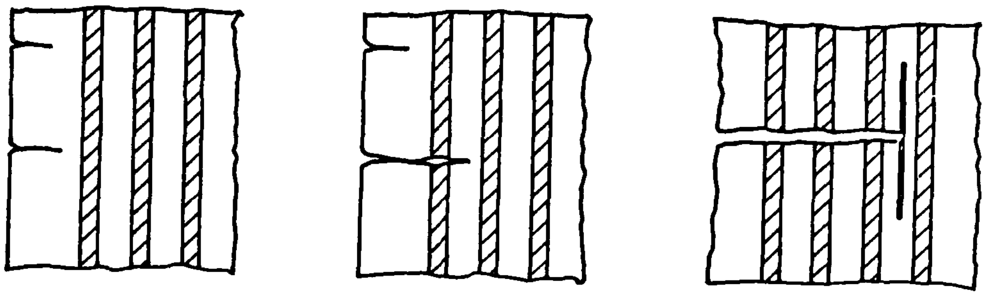
Key:
Maximale Ausdehnung des Schadens im Kerbgrund (Schlitz) = maximum extent of damage in notch root (slot)
Zugspannung bei monotoner Belastung bis Bruch = tensile stress in monotonic load to fracture

Fig 4.1 Progress of damage in 90°, 0° and 45° layers of a $[0/\pm 45/90]_{2S}$ CFC (Ref 730)



Key:
Kleinste Spannung für Matrix-Mikrorisse = lowest stress for matrix micro cracks
Kleinste Spannung für Faserbrüche = lowest stress for fibre fractures
Lastwechsel bis Bruch = load cycles to fracture

Fig 4.2 Areas of differing damage development in unidirectional CFC (Ref 141)



Key:
Aus Oberflächenschäden wachsende Matrix-Mikrorisse = matrix micro cracks generated by surface damage
Bevorzugter Faserbruch = preferred fibre fracture
Zwischenfaser-Schubbruch = inter-fibre shear fracture

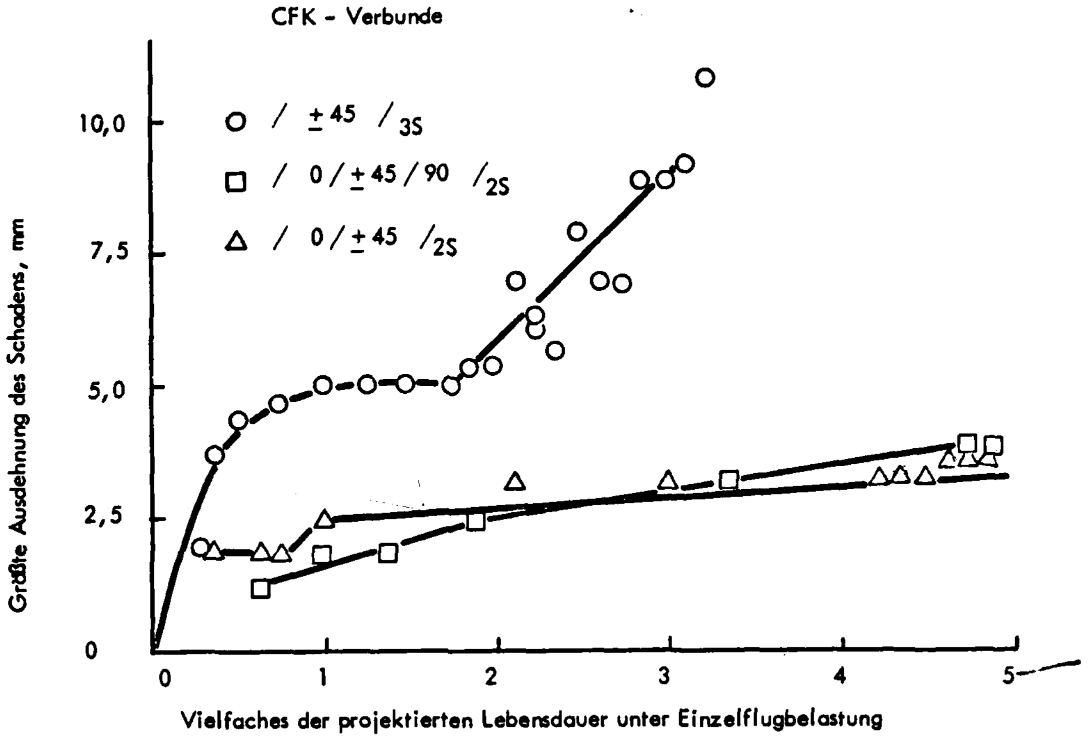
Fig 4.3 Damage development in Range II of Fig 4.2 for unidirectional GFC (Ref 141)

Fig 4.4 (Table 1)

Comparison of development of damage in BFC and CFC angle ply composites; cyclically stressed specimens (T-T) with central hole

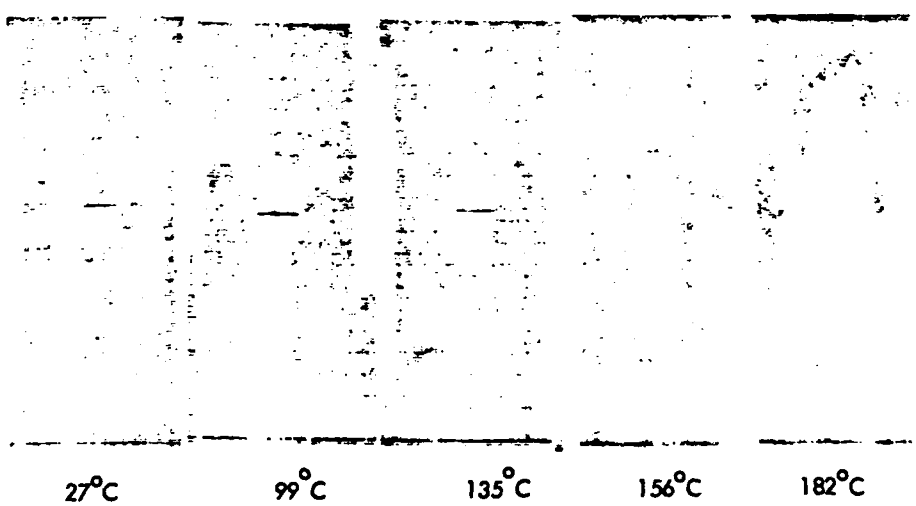
Composites with	BFC	CFC
0^{o}, $\pm 45^{o}$ layers	1. Intralaminar cracks (interface cracks) occur at edge of hole. 2. Longitudinal cracks and local intralaminar cracks in $\pm 45^{o}$ layers occur in narrow bands parallel to load direction and bordering the hole. 3. In this area the fibres in the $\pm 45^{o}$ which were not intersected by the hole broke.	1. Matrix cracks occur at hole edge which run across the layer plane along $\pm 45^{o}$ fibres and along 0^{o} fibres bordering the hole. 2. Interlaminar cracks (delamination) occur at the edge of the hole between 0, +45 and -45^{o} layers. 3. 0^{o} matrix cracks which border the hole (longitudinal cracks) grow and 45^{o} cracks grow at the same time into the gripping area above and below the hole.
0^{o}, $\pm 45^{o}$, 90^{o} layers	1. Intralaminar cracks (interface cracks) occur at hole edge. 2. Intralaminar cracks in $\pm 45^{o}$ layers grow from edge of hole into the nett cross-section along the lateral axis of the specimen. 3. Near the hole 0^{o} and $\pm 45^{o}$ fibres broke where intralaminar cracks had occurred in $\pm 45^{o}$ layers. 4. Further away from the hole (in nett cross-section) only $\pm 45^{o}$ fibres broke where intralaminar cracks had occurred in $\pm 45^{o}$ layers.	1. Matrix cracks occur at edge of hole and run across the layers along the 90^{o} and $\pm 45^{o}$ fibres and along the 0^{o} fibres bordering the hole. 2. 0^{o} matrix cracks (longitudinal cracks) grow and 90^{o} matrix cracks extending over the whole width of the specimen increase in number. They are particularly dense in the hole area and grow up to the specimen edge. Interlaminar cracks (delamination) occur at edge of hole between 0^{o}, $\pm 45^{o}$ and 90^{o} layers and at specimen edge between 90^{o} and neighbouring layers. 3. Delamination at the hole extends primarily between the propagated longitudinal cracks. $\pm 45^{o}$ matrix cracks increase and grow from the edge of the hole across the nett cross-section. Number of 90^{o} cracks has increased further. 4. Delaminations and $\pm 45^{o}$ cracks grow between the longitudinal cracks and with them into the gripping area. At the same time the delaminations emanating from the specimen edge have grown more rapidly. They finally penetrate the entire width of the specimen.

Fig 4.4



Key:
Grösste Ausdehnung des Schadens = maximum extent of damage
Vielfaches der projektierten Lebensdauer unter Einzelflugbelastung = multiple of projected life span under flight-by-flight stressing

Fig 4.5 Effect of laminate on growth of damage (Ref 730)



Key:
Ultraschall-Bilder, etc = Ultrasonic images for specimens with surface notches after flight-by-flight stress (twofold life) at different temperatures

Fig 4.6 Effect of temperature on the extent of fatigue damage (delamination) (Ref 5[illegible])

5 USE OF FRACTURE MECHANICS FOR FIBRE COMPOSITE MATERIALS

H. Huth

Contents

5.1 Introduction

In aircraft construction the use of fracture mechanics methods in proving the fatigue life and safety of metal structures is a matter of course. As a result of the success of the technique the regulations for providing evidence of adequate operational strength (*eg* MIL 83444) have been adapted entirely to the use of fracture mechanics. However, structures of fibre composite materials were expressly excluded. For civil aircraft construction AC20-107 of 10 July 1978, makes recommendations for demonstrating the operational strength of fibre composite structures. In principle, the same procedure is suggested as for metal structures, *ie* adequate damage tolerance for fail safe structures is to be demonstrated by damage propagation and residual strength investigations. In the absence of proven analytical methods (fracture mechanics) the emphasis should be on experimental evidence.

The reason must be sought in the imperfect transferability of the experimentally established behaviour of specimen bars to the behaviour of the component when using the concepts of 'classical fracture mechanics'. Knowing the damage mechanisms taking place in fibre composite materials, and the anisotropy and properties which are very different from metals, the application of fracture mechanics without certain supplementary features is not to be expected.

The aim of this literature evaluation is to provide a survey of the present state of knowledge. The different approaches for prediction of damage propagation under static and cyclic stress based on fracture mechanics principles are compared. Purely empirical and analytical methods will be dealt with. Mathematical formulations will be largely ignored. A concluding assessment will point to conclusions for practical application and likely research.

5.2 Use of fracture mechanics for metals

To further a better understanding of the difficulties which can occur in the prediction of damage propagation and residual strength of fibre composite materials, the usual method for metal materials will first be explained.

The procedure for determining crack propagation life (the term residual life is also common) of a component will be described with the aid of the flow chart in Fig 5.1:

First the critical cross-section of the component and the (nominal) stress distribution arising from the external loads have to be determined. After that assumptions can be made on possible forms of cracking, based on practical experience if available. Now an often very difficult intermediate step can be taken - namely establishment of the stress intensity factor. The possibilities for inspection determined by the structural design of the component and operating conditions (accessibility) yield the magnitude of the initial crack lengths necessary for life estimation which can be safely detected. (In place of this quantity the crack length of damage discovered in operation can also be used.) For the component material the fracture mechanics characteristics relevant to the existing material condition and the operating conditions (temperature, environment, loading speed, etc) must be known or determined experimentally. From load assumptions (for new designs) or long-term measurements, data must be provided on the load-time sequence and the loading spectrum. The determining factor for further calculation is the nominal stress spectrum in the critical cross-section, as needed for proving fatigue strength. Further, the extent of the maximum load possible in operation must be known.

The size of this maximum load governs the critical crack length established from the breaking toughness with the aid of the equation for the stress intensity factor. The residual life available is the time (or number of cycles) needed for a fatigue crack to grow from the above mentioned initial crack length to the critical crack length. This crack propagation calculation presupposes knowledge of the crack propagation behaviour of the material as well as the stress-time sequence (spectrum). The length of inspection intervals to be laid down can be derived from the mathematical crack propagation life which is now known, insofar as inspections can be carried out at all during operation. Otherwise a crack propagation calculation can be carried out as evidence of uncritical crack growth behaviour during the entire proposed useful life of the component (safe-crack-growth-structure).

To summarise, it may be said that successful use of fracture mechanics methods is assured only when the following points are known:

(i) the equation of the stress intensity factor for the assumed or actual crack form,

(ii) applicable, statistically secured fracture mechanics material data (breaking toughness and crack growth behaviour),

(iii) reliable data on operating conditions, especially on the time sequence of loads,

(iv) reliable crack growth calculation method,

(v) precise stress analysis for the cross-section in question,

(vi) trustworthy evidence on the precision of the inspection method used.

In practice this is only so in exceptional cases even for metal structures. Generally speaking, it is left to the experience of the user to make suitable assumptions in place of missing information and to judge the accuracy of the results thus achieved.

In the attempt to judge the applicability of fracture mechanics to fibre composite materials it is not necessary to go into point (iii) of the above conditions. The same applies to point (v), since this is dealt with in section 6. Point (iv) must be modified for purposes of comparison since even with isotropic materials there are uncertainties in crack growth prediction for load-time sequences with variable amplitude.

5.3 Use of fracture mechanics for fibre composites

5.3.1 Special characteristics as compared with use for metals

The question arises in the case of fibre composites as to whether the basic condition for the applicability of linear elastic fracture mechanics, namely the validity of the equation: $G_c = f(a_{i,j})K_c^2$ can be satisfied at all. In this basic equation G_c is the critical energy release rate, K_c the critical stress intensity factor and $a_{i,j}$ elastic constants. Due to the heterogeneity and anisotropy of fibre composite materials this equation is in general not valid. A large number of investigations are therefore concerned with this fundamental question (see section 5.5.1). If the composite material can be assumed to be, *eg* homogeneous and orthotropic it is possible to relate G_c and K_c, as shown by Sih[115], Dahran[665] and Kanninen[752].

For this special case an anisotropy factor can then be determined for establishment of the stress intensity factor. This factor calculated by Konish[742] is shown in Fig 5.2 for a specimen with centre crack.

As a further criterion for the use of fracture mechanics many authors quote the condition that the local zone of damage at the tip of the crack must be small compared with the specimen (component) and that this zone does not have a major adverse effect on the elastic stress distribution. This, according to, *eg* Zweben[114] cannot be met if delaminations occur as a type of failure.

Altogether the differing failure mechanisms seem to be the main obstacle to the use of fracture mechanics concepts. Since these were described in detail in section 4, the individual types of failure will not be dealt with further here.

Many authors deal with the mathematical establishment of stress intensity factors, partly by means of finite element calculations, partly analytically (see section 5.5.2). In the cases treated so many assumptions had to be made regarding type of crack, laminate structure, material properties, etc, that it is not yet possible to see the practical value of these results. An example of such mathematical results is shown in Fig 5.3.

5.3.2 Fracture mechanics concepts for residual strength prediction

Some concepts being followed at present will be introduced below in order in the first place to document the state of the art and, in the second place, to explain the difficulties which occur in attempting to apply linear elastic fracture mechanics to fibre composite materials.

The investigations of Waddoups *et al* described in Ref 770 of CFC laminates with holes or slits showed that residual strength could be satisfactorily predicted for different cracks lengths if an equivalent crack length describing the size of the damage zone was defined. But it was not possible to confirm fracture toughness values derived therefrom for specimens with holes of different diameters.

For residual strength prediction Whitney and Nuismer[591,620] also took account of the stress concentration factor and stress distribution at a notch or crack tip. The idea behind their 'average stress' theory is that fracture occurs when the average value of stress in the load direction (σ_u) over a certain length (a_0) has reached the tensile strength (σ_0) of the unnotched material. An example of their results is shown in Fig 5.4. Their test results with two different CFC laminates agree very well with predictions using

$$K_Q = \sigma_0 \sqrt{\pi \ell (1 - (\ell/\ell + a_0)^2)}$$

True, the constants σ_0 and a_0 for the material and laminate structure must be established experimentally.

In Ref 753 Morris and Hahn treated, in a similar manner, cracks in two CFC laminates which sloped across the load direction. They quote for instance for a 0°, $\pm 45^\circ$ laminate the size of the damage zone a_0 at which predictions most closely agreed with the experiments as a_0 = 1.25 to 2.87 mm.

The theories mentioned so far, as well as that of Wu[735], all require two or more parameters to be established experimentally for each laminate. From the start they do not raise any expectation of transferability to other laminates. This is more likely to be expected of the following experiments which concern themselves in more detail with stress conditions at the crack tip.

Wang and co-workers[739,752] used hybrid elements in order to carry out a finite element stress analysis in layers. They found that outside a zone of damage at the tip of the main crack the classicial $1/\sqrt{r}$ stress distribution obtained in the individual layers if they assumed that each displayed a homogeneous linear elastic and anisotropic behaviour. Within the damage zone, comparable with the plastic zone in metals, heterogeneous behaviour and the formation of subsidiary cracks was admitted.

Mandell and co-workers tested experimentally in Refs 392 and 740 the effect of subsidiary cracks (for example cracks parallel with the 0° fibres) and delaminations on the fracture toughness of GFC, CFC, BFC and Kevlar laminates. The observed effects of specimen thickness and specimen width were attributed also to the different interlaminar shear stresses.

Wu[735] as well as Sih and Chen[751] used the strain energy - density concept in order to determine the conditions for crack propagation and the direction of crack extension in a material which is orthotropic outside the damage zone. So far, however, it has only been possible to apply these micromechanical concepts to uni-directional laminates.

One means of avoiding the difficulties arising with more complicated laminate structure is contained in the model introduced by Kanninen[752]. He defines a local heterogeneous zone surrounded by a linear orthotropic continuum (see Fig 5.5). Here the material properties of the fibres, the resin and the interface are required to describe the properties of the local zone. Thus the various failure mechanisms can be simulated locally. Comprehensive experimental confirmation of the useability of this model is not yet available.

5.3.3 Fracture mechanics concepts for prediction of the spread of damage under cyclic load ('crack propagation')

The difficulties described in 5.3.1 and 5.3.2 in the determination or definition of a stress intensity factor for fibre composites affected by cracks or damage suggest little likelihood at present of universally applicable fracture mechanics treatment of fatigue damage propagation. This is demonstrated inter alia by the extremely small number of references to this subject.

Proof that, like metals, the measured crack propagation rates can be presented as a function of the range of stress of the stress intensity factor, in the form of

$$\frac{d\ell}{dn} = f(\Delta K)$$

has succeeded in few cases only. Investigations of GFC laminates (unidirectional, 0^o, 90^o) by Mandell[190] and Mandell and Meier[133] showed that crack propagation in such materials takes place stepwise, that is from fibre bundle to fibre bundle (lying in direction of load). They therefore developed an approximation solution for prediction of crack propagation which takes account of fracture toughness, (unnotched) fatigue strength and the local stress field. Good consistency was shown between theory and experiment for all laminate types. In a 'global' observation of crack propagation an exponential dependence on the stress intensity factor was found.

Owen and Bishop[417] achieved good results in describing crack propagation behaviour of various GFC laminates by means of a fracture mechanics crack propagation equation (Paris equation). However, they had to define an equivalent crack length via compliance calibration curves, since the partially extended damage zones at the crack tips did not permit crack length measurement in a true sense. The specimens tested were central crack specimens (with saw cut), and the crack opening in the centre of the specimen was measured with a clip-on-gauge to indicate compliance.

A similar procedure was selected by Sturgeon[537] when investigating a CFC laminate. Residual life was determined by him too by the compliance change (measured between grips).

Campbell and Cherry[747] examined especially the processes at the crack tip under cyclic load in various CFC laminates. They found different damage propagation behaviour as compared with monotonic loading, which they attributed to adiabatic heating at the crack tip. Crack propagation equations could not be derived from their test results. This frequency-dependent behaviour was studied intensively by Williams[768] on BFC and Boron-Al.

Crack propagation tests of composite materials with metal matrix are described in Refs 734, 736 and 737. Of these special emphasis should be given to Kendall's investigation[736] for he demonstrated on specimens with different laminate structure that the crack length visible on the surface of the specimen conforms well to the dimension of the broken 0^o fibres. As in the case of most metal matrix fibre composites crack propagation here too was collinear and transverse to the direction of loading. For this reason it was possible to present the test results in the form of a crack propagation equation. Not tested were questions of transferability, effects of laminate structure, mean stress, etc.

5.3.4 Application of fracture mechanics to fibre composite components

Except for two examples described by Dharan in Ref 665 and the investigation by Porter[737] which applies the test results to the laying down of acceptance tests of fibre composite components (pressure tanks), the state of the art is a long way from achieving successful residual life prediction by fracture mechanics means. Even under favourable circumstances, such as available test results of crack propagation and residual strength investigations on the same laminate, the following obstacles prevent straightforward prediction for the component:

- effect of size (edge effect),
- form of initial damage in operation,
- residual thermal stresses,
- effect of ageing and corrosion (environment).

This means that on the one hand there is no evidence of transferability of results established with small specimen bars, and on the other hand damage propagation is affected by a multiplicity of parameters not so far examined.

In the selection of material and optimisation of laminate structure fracture mechanics tests have proved helpful, as for instance in the layout of hybrid laminates[476,493] and in the assessment or improvement of resistance to impact damage[771-774].

5.4 Conclusions

The application of fracture mechanics methods to fibre composites is extremely complex since, in contrast to isotropic materials, failure depends on fibre orientation and the mechanical properties of fibres, matrix and interface. Symmetrical initial damage can therefore very well lie in an asymmetrical stress field, which may result in damage propagation which is not transverse to the load. Moreover this is not reproducible and can contain different types of failure. Thus the application of concepts of linear elastic fracture mechanics derived from isotropic homogeneous materials is excluded from the start.

For those fibre composites which exhibit reproducible crack propagation characteristics (one dominating failure mechanism) and which can be regarded as homogeneous and orthotropic, linear elastic fracture mechanics can be used. In this case equations can be set up which link the crack energy release rate with the stress intensity factor. Several fracture mechanics theories have been developed to predict the critical stress produced by damage propagation (residual strength). So far none of these theories is universally applicable, each having been developed only for a definite laminate structure of one material and only sparsely, if at all, confirmed by experiment.

The main conclusion must therefore be that a very great deal of intensive research (experimental and theoretical) is needed to put the application of fracture mechanics to fibre composite materials on the same level as for isotropic materials.

5.5 References

The references evaluated in the preceding sections are listed by subject. The papers mentioned in the text are underlined.

Section 5.5.1

On applicability of fracture mechanics:

<u>114</u>, <u>115</u>, 465, 622, <u>665</u>, 728, 735, 743, 751, <u>752</u>, 755, 758, 764, 765, 767, 768.

Section 5.5.2

Stress analysis and stress intensity factors:

Finite elements:

114, 115, 417, <u>739</u>, <u>740</u>, 741, 748, 750, 756, 759.

Analytical:

36, 116, 279, 591, 620, 624, 738, <u>742</u>, 746, 765.

Section 5.5.3

Residual strength tests (K_{Ic}, R curves, compliance measurements),

(a) unidirectional:

GFC: 113, <u>115</u>, <u>392</u>, 743.

CFC: 113, <u>115</u>, 116, 136, 622, 624, 743, 769.

B-Al: 58, 97, 186, 243, 465, 481, 505, 744, 754, 755, 760, 764, 765, 767.

BFC: 113, 767.

Borsic-Ti: 765.

(b) angle ply:

GFC: 113, 133, 190, 235, 392, 621, 743, 757, 762.

CFC: 113, 116, 136, 185, 236, 537, 622, 737, 745, 743, 746, 748, 749, 753, 758, 763, 769, 770, 786.

Boron-Al: 243, 465, 736, 748, 754, 760, 768.

BFC: 113, 768.

Hybrid: 476, 493.

Section 5.5.4

Crack propagation tests:

GFC: 133, 190, 417, 747.

CFC: 136, 537, 745, 737.

Boron-Al: 734, 744, 736, 768.

BFC: 768, 717.

Be-Al: 734.

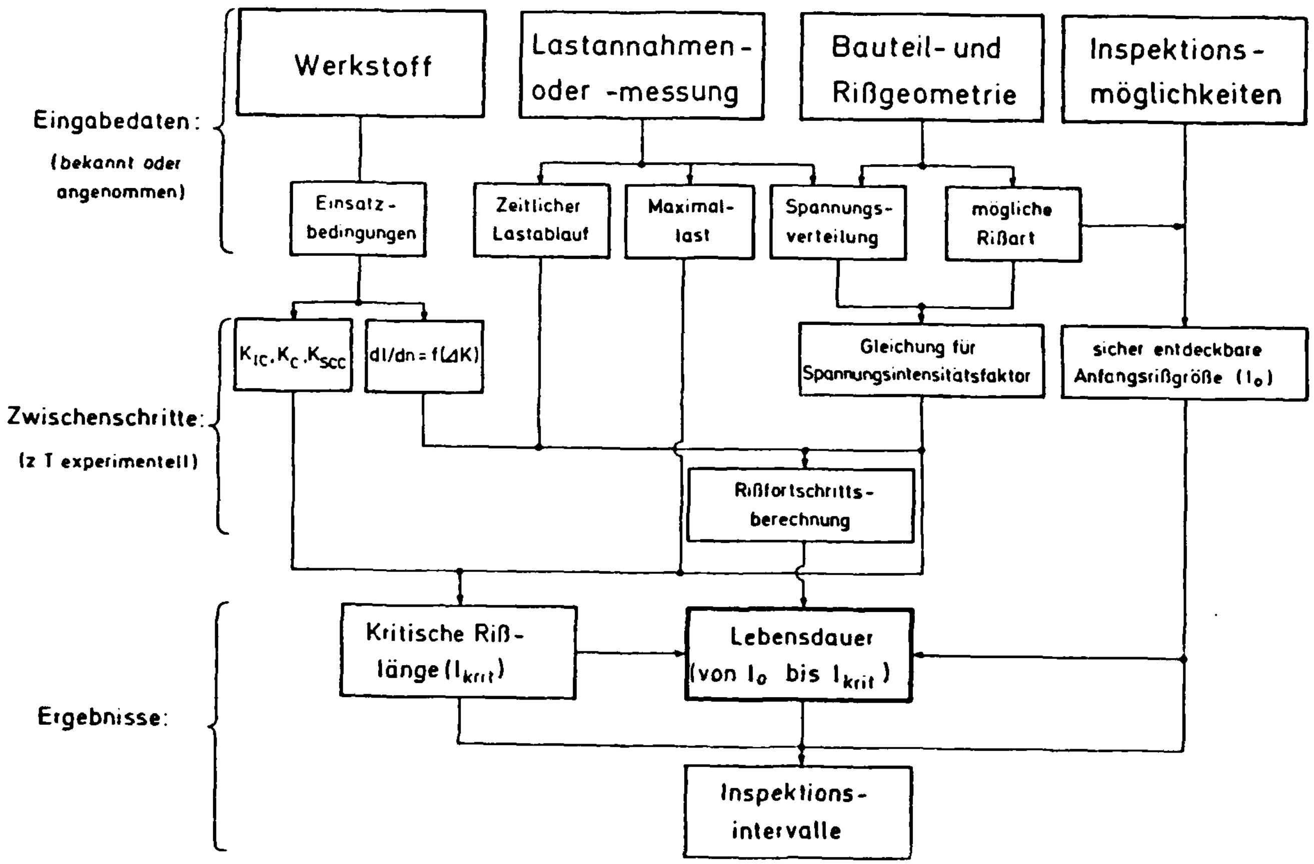


Fig 5.1 Life prediction for cracked components with the aid of fracture mechanics

Key:

Eingabedaten (bekannt oder angenommen)	= input data (known or presumed)
Zwischenschritte (z. T. experimentell)	= intermediate steps (partly experimental)
Ergebnisse	= results
Werkstoff	= material
Lastannahmen-oder-messung	= assumed or measured load
Bauteil-und Rissgeometrie	= component and crack geometry
Inspektions-möglichkeiten	= inspection possibilities
Einsatzbedingungen	= operational conditions
Zeitlicher Lastablauf	= time load sequence
Maximallast	= maximum load
Spannungsverteilung	= stress distribution
Mögliche Rissart	= possible type of crack
Gleichung für Spannungs-intensitätsfaktor	= equation for stress intensity factor
Sicher entdeckbare Anfangsrissgrösse	= size of initial crack which can be detected safely
Rissfortschrittsberechnung	= crack propagation calculation
Kritische Risslänge	= critical crack length
Lebensdauer	= fatigue life
Inspektionsintervalle	= inspection intervals

Fig 5.1 (continued)

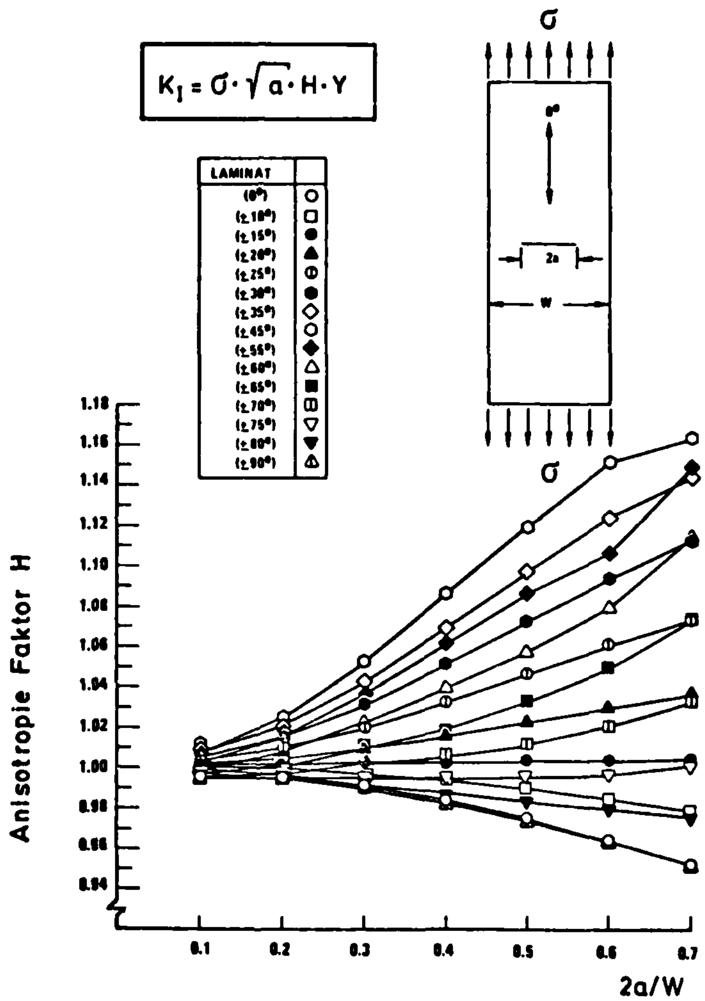


Fig 5.2 Effect of anisotropy on stress intensity factor (Ref 742)

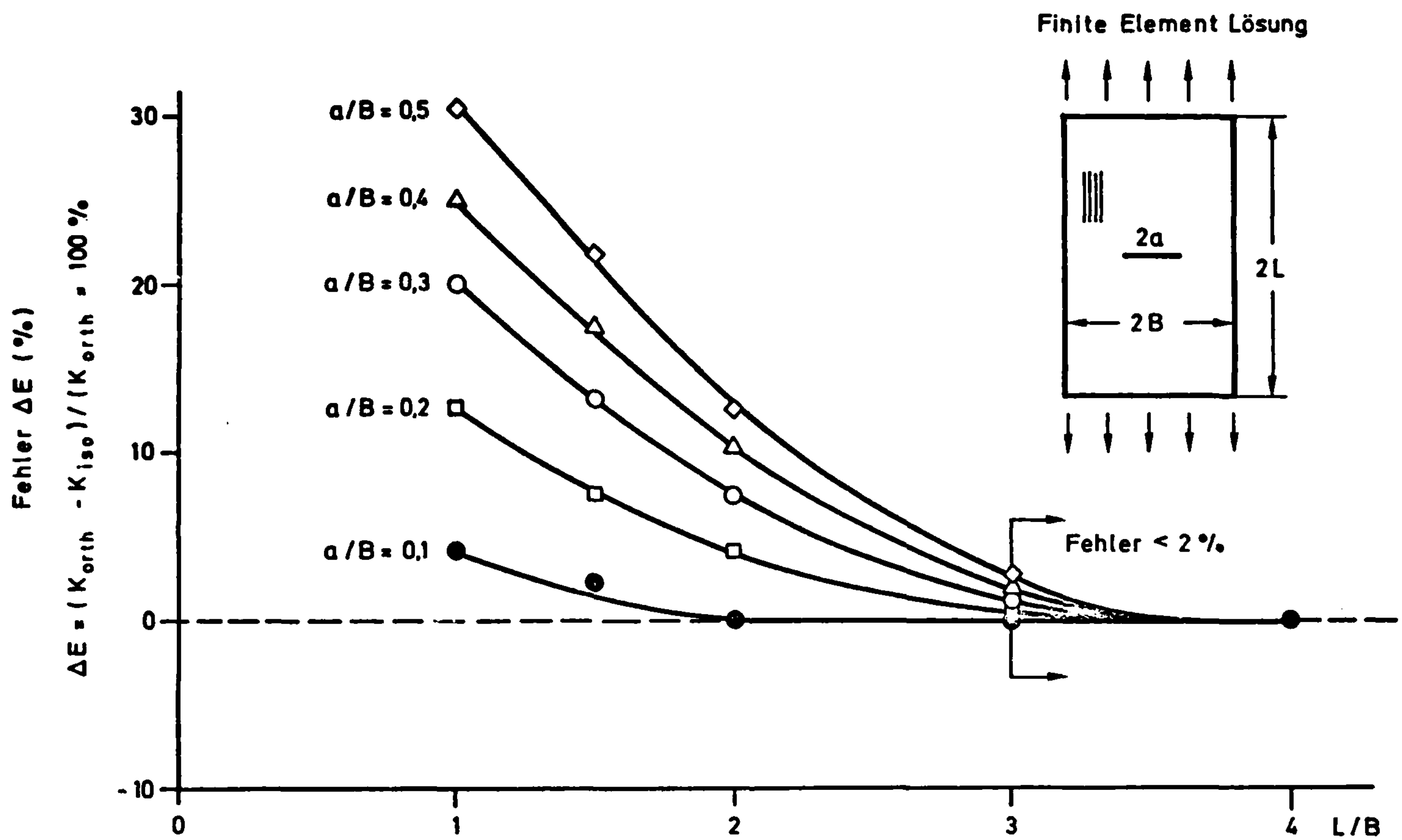
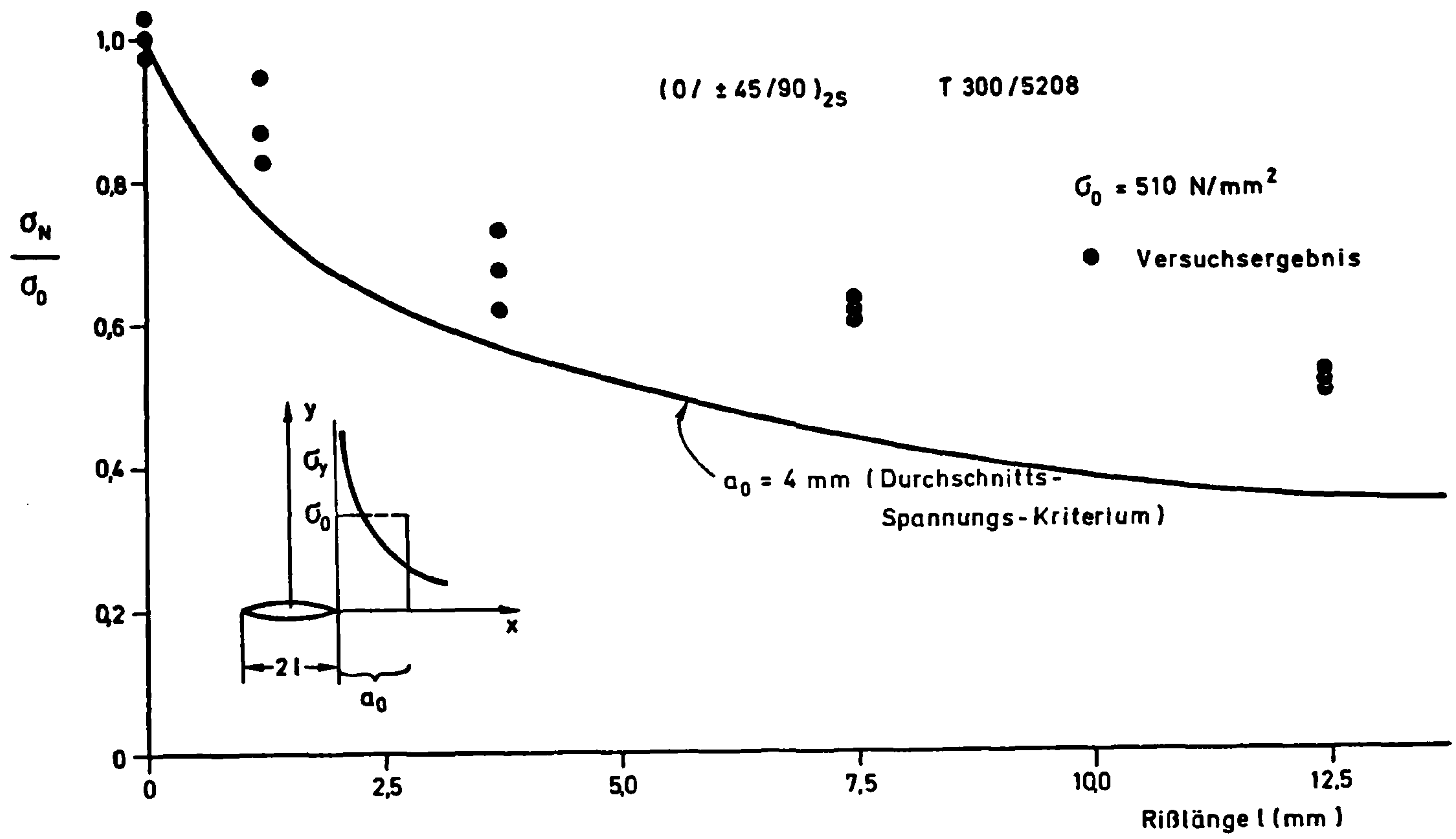


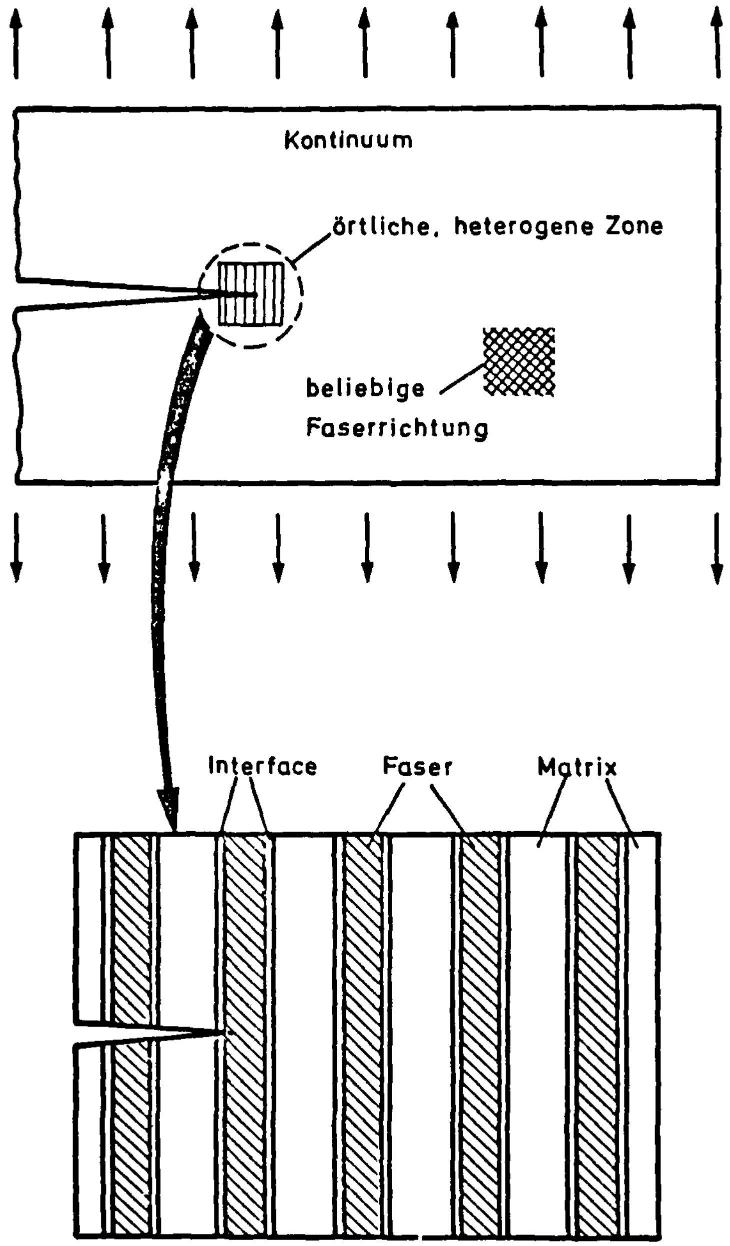
Fig 5.3 Approximation error in use of the isotropic stress intensity factor for uni-directional CFC plates with central crack (Ref 748)



Key:

Versuchsergebnis	= test result
Durchschnitts-Spannungs-Kriterium	= mean stress criterion
Risslänge	= crack length

Fig 5.4 Comparison of predicted and measured residual strength values for a CFC laminate (Ref 620)



Key:
Örtliche, heterogene zone = local heterogeneous zone
Beliebige Faserrichtung = optional fibre direction

Fig 5.5 Kanninen's model (Ref 624)

6 ARITHMETICAL METHODS OF DETERMINING THE MECHANICAL BEHAVIOUR OF FIBRE COMPOSITES

J. Franz

Contents

6.1 Introduction

In the manufacture of fibre composite materials the fibres (normally glass, carbon, boron or Kevlar) are generally embedded in a plastic matrix (*eg* epoxide or polyamide resin). The materials thus produced are fundamentally different in properties and behaviour from the conventional, especially in the following respects:

- A composite of fibres and matrix generally behaves anisotropically.
- No pronounced plastic area is usually discernible in the $\sigma - \varepsilon$ behaviour (since the fibres are frequently very brittle).
- The $\sigma - \varepsilon$ line can be bent over its entire length (either progressively or regressively), *ie* there is not necessarily a linear area.
- The scatter of static strength and the elasticity constants is greater as a rule than in metals.
- Static strength can be different under tensile and compressive loading.
- The damage mechanisms occurring are quite different from those in conventional materials, for instance:
    - fibre fracture,
    - fracture in the matrix,
    - delamination (severing of the bond between individual layers),
    - debonding (detaching of fibres from the matrix).

It is not possible, therefore, to simply transfer to fibre composites the well known methods of determining the stresses and deformations occurring in structural components. They must either be modified or quite new methods must be developed.

The following literature evaluation is aimed at providing a first general survey to the reader who is going into the special problems of calculation methods for fibre composite materials. A particular effort has been made to present clearly the phenomena which distinguish these materials from the conventional. Extensive mathematical formulations have been purposely avoided, they can be gleaned from the literature mentioned. Furthermore, only those physical phenomena are discussed which can affect the fatigue strength of a laminate (thus not, *eg* bending or bulging problems in fibre composite components).

6.2 Establishing the elasticity constants of laminates

6.2.1 Survey

The aim of a mathematical investigation is as a rule to predict the failure of a component. For a conventional isotropic metal structure one calculates the stresses on the most heavily loaded part of the component with the experimentally determined elastic constants E and ν. If there is a multiaxial stress condition a fracture hypothesis is used to determine a uniaxial stress which by comparison with a static tensile test enables one to state whether fracture will occur at the given state of stress or not.

Procedure for fibre composite materials is similar. In addition, however, means exist of optimizing the elastic behaviour during the manufacture of the laminate (by selecting fibres and resin, fibre content and orientation of the individual layers). A parameter study to be carried out for this purpose must comprise the following steps:

- establishment of the elasticity constants of the uni-directional individual layer (assuming a uniform stress condition, at least E_1, E_2, G_{12}, ν_{12}, ν_{21}, from the elastic properties of fibre and matrix),
- transformation of these constants in the direction of the external load unless the load direction coincides with an orthotropic axis of the layer in question, and
- establishment of the elasticity constants of the laminate by superimposition of the elastic properties of the individual layers.

The following paragraphs go more fully into the necessary procedures and the results thus achieved.

6.2.2 Uni-directional laminate or individual layers

Several papers deal with the calculation of the elastic characteristics of uni-directional laminates or individual layers. Depending on the assumptions required for derivation, the equations used in the individual methods differ. They all have in common, however, that the laminate properties depend on the fibre and matrix properties as well as fibre and matrix content. Empirical constants can also play a part.

A detailed presentation of the problem including a parameter study and comparisons with test results is found in Ref 339. The dependencies on the influencing variables resulting from this parameter study are shown in Figs 6.1 to 6.3. The results of the study are:

- E_1 depends essentially on the modulus of elasticity of the fibres.
- E_2 and G_{12} depend essentially on the modulus of elasticity of the matrix.
- The transverse strain factors of fibre and matrix do not affect E_1, E_2 and G_{12} significantly.
- The matrix fraction affects E_1, E_2 and G_{12} significantly.
- Misalignment of the fibres reduces E_1 (not shown in Figs 6.1 to 6.3).

Comparisons with test results in Ref 339 show that the theoretical predictions hold true with satisfactory accuracy, provided the empirical variables are chosen skilfully.

The following can be observed in general:

- In the development of a uni-directional laminate or individual layer a prior parameter study is necessary to achieve certain mechanical properties; the literature contains suitable methods of calculation, *eg* Ref 339.
- Since linear material behaviour of the individual components (fibres and matrix) is assumed as a rule in such parameter studies (this means the $\sigma - \varepsilon$ line runs straight over the whole length, which is not precisely so in practice), elasticity constants independent of load also arise for the layer.
- If the laminate or the layer is to hand it is advisable to check the mathematical prediction by additional experimental determination of the elasticity constants in the interests of accuracy of the theoretical investigations based on them.

If, as happens in many cases in practice, the individual layers are supplied by the manufacturer in the form of prepregs a parameter study to optimize the fibre fraction is superfluous.

6.2.3 Multi-directional laminate

Since components of fibre composite materials are usually available for practical purposes as real structures, most authors regard the conditions for a level stress state as fulfilled. A detailed presentation in Ref 600 shows how the general, three-dimensional law of elasticity

$$\varepsilon_i = S_{ij}\sigma_i \tag{1}$$

or

$$\sigma_i = Q_{ij}\varepsilon_j \tag{2}$$

with

$$i, j = 1, 2, 3, 4, 5, 6$$

is converted into the much simpler law of elasticity of the biaxial stress condition by omitting individual coefficients which can be regarded as negligible:

$$\begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} = Q_{ij} \begin{bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{bmatrix} \tag{3}$$

For the uni-directional individual layer the components Q_{ij} are computed from the moduli of elasticity and transverse strain factors determined in 6.2.1 (cp, *eg* Ref 600). If the load direction is at an angle to the 0^o axis, there are equations with which the transformed rigidity elements $\overline{Q}_{ij}$ can be determined in the load direction.

Taking account of the definition of force and moment usual for plates and dishes, (3) yields the following material law for a plane load-bearing structure (which applies both for an individual layer and a uni-directional or multi-directional laminate):

$$\left[\begin{array}{c} N_1 \\ N_2 \\ N_{12} \\ \hline M_1 \\ M_2 \\ M_{12} \end{array}\right] = \left[\begin{array}{ccc|ccc} A_{12} & A_{12} & A_{16} & B_{11} & B_{12} & B_{16} \\ A_{12} & A_{22} & A_{26} & B_{12} & B_{22} & B_{26} \\ A_{16} & A_{26} & A_{66} & B_{16} & B_{26} & B_{66} \\ \hline B_{11} & B_{12} & B_{16} & D_{11} & D_{12} & D_{16} \\ B_{12} & B_{22} & B_{26} & D_{12} & D_{22} & D_{26} \\ B_{16} & B_{26} & B_{66} & D_{16} & D_{26} & D_{66} \end{array}\right] \left[\begin{array}{c} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \\ \hline \kappa_1 \\ \kappa_2 \\ \kappa_{12} \end{array}\right] \tag{4}$$

where N_1, N_2 : normal forces — N_{12} : shear

ε_1, ε_2 : strains in neutral plane — γ_{12} : slip

M_1, M_2 : bending moments — M_{12} : torsional moment

κ_1, κ_2 : curvatures — κ_{12} : twist

$$\left.\begin{array}{l}[A] : \text{strain stiffness matrix}\\ [B] : \text{bending strain coupling matrix}\\ [D] : \text{bending resistance matrix}\end{array}\right\}\ \text{terms as Ref 597}$$

The equations for determination of the components of the sub-matrices A, B and D can be found in the literature[600,339,597]. It is a matter of equations for superimposition (parallel connection) of several layers, taking account of the differing elasticity behaviour (*ie* differing coefficients Q_{ij} or $\overline{Q}_{ij}$) and different layer thicknesses.

The unusual indexing is to indicate that in the transition from spatial to plane stress condition parts of the components have been dropped (*eg* A_{13}, A_{14}, A_{15}, B_{13}, etc).

Coverage of the stiffness matrix occurring in material law (4) gives information on the deformation behaviour of a laminate. Some possible cases are shown in Fig 6.4. Some explanations of the terms used there:

A material is:

- isotropic if it behaves the same in all directions (normal assumption for a metal),
- orthotropic when there are three symmetry planes standing on top of one another (*eg* a uni-directional laminate),
- anisotropic in all other cases.

For fibre composite materials the following additional terms are used to characterise the laminate structure:

- 'balanced' if for every layer of the laminate with orientation $+\theta$ there is a layer of the same thickness with orientation $-\theta$ (as Ref 502), and
- symmetrical if the median plane of the laminate is also the plane of symmetry.

A laminate which is symmetrical and balanced is called a 'balanced angle composite' according to Ref 609.

Fig 6.4 after Ref 546 shows the deformation behaviour of laminates under tensile and bending stress, where the validity of material law (4) is assumed. In principle the following deformations can occur:

- strain in two directions (tensile deformation) } disc deformations
- slip (shear deformation) }

- curvature round two axes (bending deformation) and } plate deformations
- twisting (torsion deformation) }

In a general laminate structure as Fig 6.4 all these deformations arise both under tensile and bending load (complete coupling of deformation and force values). In symmetrical laminates the disc values (membrane values) are uncoupled from the plate values (bending values), *ie* the bending-strain coupling matrix B becomes zero. This also applies to the laminates used in practice. As a rule, however, it is not a case of true orthotropic laminates but pseudc orthotropic:

For if a laminate contains $\pm\theta$ layers it is no longer symmetrical to the planes lying vertically to the median plane (since the $\pm\theta$ layers do not lie in one plane); thus the condition for orthotropy is not met. In deformation behaviour this is expressed by a coupling between bending and torsion (cp Fig 6.4), the effect of which according to Ref 600, however, decreases with increasing number of layers (because the asymmetry of individual layers then plays a subordinate role).

Apart from the laminates shown in Fig 6.4, there are the so-called quasi-isotropic composites. Here the fibres are orientated in n directions (minimum 3) inclined at an angle $180^{\circ}/n$. If also the structure about the median plane is symmetrical and if all layers are of the same thickness, then such a laminate behaves like an isotropic material under load with membrane forces, *ie*

$$\left.\begin{aligned} E_1 &= E_2 = E \\ \nu_{12} &= \nu_{21} = \nu \\ G &= \frac{E}{2(1+\nu)} \end{aligned}\right\} \quad \text{compare Ref 597}$$

applies.

A very detailed explanation on completing the stiffness matrix as a function of certain symmetry properties over and above the cases shown in Fig 6.4 will be found in Ref 603. This also deals with determination of the stiffness components.

Equations to determine the elasticity constants for a triaxial stress condition are contained in Ref 600. A method which determines the elasticity constants by the use of finite elements and applies it to short fibre laminates is described in Ref 85.

In conclusion it can be stated:

- As a rule plane type load carrying structures of fibre composite materials are based on a level stress condition.
- The elasticity constants of these load carrying structures are obtained by parallel connection of the individual layers (corresponding to parallel connection of springs).
- Determination of the elasticity constants for a triaxial stress condition is more complicated, partly because additional mechanical characteristics of the individual layer must be known (E_3, G_{13}, G_{23}, ν_{13}, ν_{23}).

6.3 Stress analysis

6.3.1 Stress analysis for unnotched laminates

Stress analysis of a fibre composite component can be considerably more demanding than an isotropic material because of the inhomogeneous structure, depending on the findings one expects. In certain circumstances it may be necessary to detect local micromechanical stress peaks (which occur, *eg* between fibre and matrix) since unlike metals these cannot be removed by flow processes in the material.

In the stress analysis of plane load-carrying structures the assumptions on which the standard methods of the plate and dish theory are based are frequently applied to fibre composites too, namely that

- a uniform stress condition and linear-elastic material behaviour (validity of the law of materials (4)) are present,
- there is no elongation in direction of thickness ($\varepsilon_3 = 0$), and
- the cross-sections remain level (linear displacement distribution through the thickness).

As a rule the following are also assumed:

- shear stiffness (vertical position of the cross-sections on the median plane),
- thin walls, and
- small deformations (theory first order).

How far all these assumptions apply to fibre composite materials does not appear to have been clarified satisfactorily. If one assumes they are valid, however, stress analysis can be carried out for discs, plates or dishes with the standard methods for plane load-carrying structures, when the necessary elasticity constants of the laminates in question must either be measured or determined theoretically, *eg* by the methods introduced in 6.2.

If stress distribution is to be determined in geometrically complicated load-carrying structures or in the vicinity of discontinuities (cp 6.3.2), the use of finite element methods (FEM) is usually unavoidable. But that this may be necessary even in simpler cases is clear from various investigations, compare for examples Refs 52, 479 and 558. Fig 6.5 shows the stress and displacement distributions in bending specimens of aluminium and uni-directional carbon fibre composite respectively, established with the use of a two-dimensional finite element method. While the linear stress distribution over the cross-section assumed in the classic elasticity theory of the transverse beam does actually take place in the aluminium specimen, there is an irregular stress distribution as well as displacement of the neutral plane into the tensile zone in the carbon fibre laminate. Furthermore the cross-sections no long remain level. In Ref 558 the difference in behaviour of the laminate is explained by:

- the high E/G ratio (E/G is almost 40), and
- the fact that the conditions for the use of the elasticity theory and the limiting conditions assumed are not met.

It becomes clear from this example that the assumptions valid for isotropic materials cannot simply be transferred to fibre composites. In certain cases it is even necessary to work on the basis of a triaxial stress condition in real load-carrying structures.

The definition of the stresses occurring in a laminate in a general triaxial stress condition is shown in Fig 6.6. It is a question of:

- intralaminar stresses which are transmitted by fibres and matrix and can produce failure in the layer plane, and
- interlaminar stresses which are only transmitted by the matrix and can cause the layers to separate.

In an analysis based on a uniform stress condition intralaminar stresses only are taken into account. No information is obtained on the effect of interlaminar stresses which play a part at least at loading points, holes and free edges ('edge effects'). If one departs from a uniform stress condition the interlaminar stresses can also be determined. In the literature evaluated this calculation is carried out for instance for a flexibly mounted plate with two free edges under axial[335,526,562] or bending load[250,577].

In Ref 335 two analytical methods of determining interlaminar stresses are compared with a difference equation, using an axially loaded $[\pm 45]_s$ laminate as an example.

The following applies to the three methods:

Method A : triaxial stress condition, analytical solution.
Method B : modified biaxial stress condition, analytical solution.
Method C : triaxial stress condition, solution with finite differences as Ref 526.

A comparison of the numerical results is shown in Fig 6.7 (the stresses not mentioned are negligible at the layer boundary examined). It should be noted that with method B mean stresses are determined and shown, with methods A and C local stresses and their maxima. But it is clear in all cases that interlaminar shear stress τ_{13} can become very great at the edge of the specimen. Possible separation of the layers will thus emanate from there. With method C the axial displacement distribution over the thickness of the laminate was also established in Ref 526. The result is shown in Fig 6.8a. It follows from this that the assumption that the cross-sections will remain level to the edge of the specimen is progressively less fulfilled.

Fig 6.8b, also from Ref 526, shows the dependence of the interlaminar shear stress τ_{13} on the fibre orientation angle θ. One notices that at $\theta \approx 60^\circ$ the direction of this stress is reversed. For $\theta \approx 60^\circ$ τ_{13} becomes zero, as for $\theta = 0^\circ$ or $\theta = 90^\circ$.

The investigations carried out on interlaminar stresses lead to the following conclusions:

in Ref 562

- the size of the interlaminar stresses depends on fibre orientation and the sequence of the individual layers ('stacking sequence'),

and in Ref 526

- interlaminar stresses occur at free edges of laminates and fade towards the centre,
- depending on the fibre orientation relatively high interlaminar shear stresses may be necessary to enable shear transmission between individual layers to take place,
- it can be expected that these high stresses in the vicinity of the free edge will cause delaminations, especially under cyclic load.

These results show that even for relatively simple load-bearing structures of fibre composite materials it may be necessary to use extensive methods of calculation (*eg* FEM) to determine stress. The use of simpler procedures (such as the classic elasticity theory) frequently fails because the edge conditions cannot be met satisfactorily. The importance of FEM is even greater for stress analysis in the vicinity of changes in the cross-section (*eg* notches).

6.3.2 Stress analysis around notches

In isotropic materials the stress concentration factor α_k is defined as the ratio of maximum tangential stress at the notch to nett stress in the notched cross-section. The introduction of such a stress concentration factor becomes very problematical for multi-directional laminates, since both the nett stress and the stress distribution around the notch are very different in the individual layers. Furthermore because of the non-isotropic behaviour of these materials in contrast to metals the material constants (of fibres and matrix) also influence stress around the notch. This effect can be recognised in Fig 6.9. The increase in stress at the side of the hole rises in proportion to the stiffness of the fibres in comparison with the matrix, *ie* the greater the ratio E_1/E_2 or the deviation from isotropic behaviour (for a carbon-epoxide laminate the stress is greater by a factor of approximately three than for an isotropic material). Apart from this, the rise in stress depends on the angle between direction of fibre and load. This connection is shown in Fig 6.10.

It will be seen that the stress at the lateral hole edge is greatest if the direction of load is parallel to the fibres. Fig 6.10 also shows the points at the edge of the hole where the first damage is expected. (This prediction is made by using a fracture hypothesis for uniaxially loaded notched laminates deduced in Ref 62.) According to this failure in most cases does not arise at the location of maximum concentration of tensile stress, but is governed by an unfavourable combination of longitudinal, transverse and shear stresses at the edge of the hole.

For multi-directional laminates satisfactory theoretical determination of stress distribution at notches is possible only with relatively onerous methods. This emerges, *eg* from an investigation in Ref 253 in which different methods are used and compared to determine the circumferential stress at the edge of the hole of a $(-45, +45)_s$ laminate. These are:

- a two-dimensional isotropic calculating method,
- a finite element method with three-dimensional isotropic elements, and
- a finite element method with three-dimensional anisotropic elements.

The result of the comparison is shown in Fig 6.11. This shows that, judged by the anisotropic method, both isotropic methods are equally unsuitable for description of the anisotropic material behaviour.

In multi-directional laminates interlaminar stresses occur at the edge of the hole as well as at free edges. An investigation will be found in, *eg* Ref 258. Fig 6.12 shows the τ_{23} distributions at the edge of the hole of a $[0, \pm 45, 0]_s$ laminate established theoretically in this paper. The shape and the absolute size of these distributions differ in each layer and between the individual layers. The maxima of τ_{23} in the case examined are about 10% of the normal stress in the gross cross-section.

To summarise:

- A most accurate determination of both the intralaminar and interlaminar stresses at the edges of holes or cutouts in multi-layer fibre composite components demands three-dimensional analysis, *eg* with anisotropic finite elements.

- The interlaminar stresses occurring at these edges may assume great importance in respect of the life of a cyclically stressed component. They should therefore be taken into account when using a failure hypothesis.

6.3.3 Stresses caused by environment

(a) Preliminary remarks

In conventional structures thermal stresses can occur due to changes in the ambient temperature if expansion of a component is prevented. The same effects arise in multi-directional fibre composite materials due to the differing thermal expansion behaviour of the individual layers. Further, internal stresses can occur if the moisture concentration in the laminate changes. The combined effect of a change in moisture concentration and a change in temperature can be defined as follows:

The elasticity law (2) quoted in 6.2.3:

$$\sigma_i = Q_{ij}\varepsilon_j \qquad (2)$$

when taking account of temperature and moisture concentration as in Ref 566 is converted into:

$$\sigma_i = Q_{ij}[\varepsilon_j - \alpha_j T(z,t) - \beta_j M(z,t)] \qquad (5)$$

where α_j : temperature expansion coefficient

β_j : moisture expansion coefficient

$T(z,t)$: temperature distribution

$M(z,t)$: moisture concentration distribution.

Formal detection of changes in moisture is therefore similar to detection of temperature changes when determining the stresses in a laminate.

(b) Example: Moisture absorption by a laminate

Fig 6.13 shows the moisture as well as longitudinal and transverse stress distribution in individual layers following moisture absorption from the environment for a $[0^{o}, \pm 45^{o}]_s$ laminate. The longer the specimen is exposed to the moist environment, the more uniform the distribution of moisture becomes in the laminate and also the (theoretical) stress distribution in the individual layers. When a constant moisture content is finally reached over the laminate cross-section, however, the stresses do not become zero (because the differently orientated layers expand differently in the various directions due to their anisotropic behaviour). When using fibre composite components penetration of moisture should therefore be prevented by suitable surface treatment, especially as it must also be expected to detract from the fatigue strength of the laminate.

(c) Thermal stresses during cooling after curing

A frequent and decisive case for the residual stresses present in the laminate in which thermal stresses can occur in fibre composite materials is the cooling of the laminate after curing. A method for theoretical determination of the strains and stresses arising is described in Ref 579. In view of the relatively wide temperature differences, it is necessary to take account of the dependence on temperature of the elasticity constants and the temperature expansion coefficients in the longitudinal and transverse directions. In determining these relationships one usually has to resort to measurements.

Fig 6.14a compares theoretically predicted thermal expansions with experimentally established ones. This yields very satisfactory agreement. In Fig 6.14b after Ref 569 the transverse and shear stresses, calculated after curing, for cooling of a carbon/polyimide laminate are given, when it transpires that the transverse breaking strength can be exceeded, although actual measurements yield lower stress values. The calculation is based on the supposition that the laminate is stress-free at the curing temperature (an assumption which is probably rarely justified).

(d) Conclusions

In conclusion the following may be asserted on the 'hygrothermal' behaviour (behaviour under moisture and heat) of fibre composite materials:

- A change in the moisture content or temperature of fibre composites causes distortions and stresses in the laminate.
- Particularly in the cooling process after curing these may reach a considerable size (because the temperature drop is relatively large).
- If the occurrence of these stresses cannot be prevented (for instance by protecting the surface from moisture), one should at least be clear as to their presence and size and take due account of them in stress analysis or damage prediction.

6.4 Failure prediction

6.4.1 Failure prediction for uniaxially loaded uni-directional laminates

Failure can occur in a uniaxially loaded uni-directional laminate due to exceeding:

- tensile strength in longitudinal or transverse direction,

- compressive strength in longitudinal or transverse direction, or
- intralaminar shear strength.

In order to obtain evidence on the failure of such a laminate these strengths, therefore, have to be determined either theoretically or experimentally (if the laminate is already available) and compared with the stresses arising from the external load.

Theoretical formulae for calculation of the strengths are given in Refs 26 and 424. The following parameters are relevant:

- volume fraction, size, number and distribution of fibres and voids,
- elastic properties (E, G, ν) and strength properties (breaking strength, breaking strain) of fibres and matrix,
- and adhesive power of the fibres in the matrix and residual stresses in the laminate.

The reduction in strength by voids rests on the stress concentrations produced in the matrix as well as the loss of cross-sectional area. An example of the size of the stress peaks caused by fibres and voids is shown in Fig 6.15a. Fig 6.15b shows the dependence of transverse tensile strength of a unidirectional laminate on fibre and void volume fractions. It emerges that the strength across the fibres diminishes with increasing fibre fraction (and increasing void fraction).

A case dealt with very frequently in the literature is failure of a unidirectional laminate under compressive load. Here as a rule the buckling of the fibres in the layer plane (micro-bulging) is investigated in addition to pure compressive failure of the fibres or the matrix. According to Fig 6.16a two different cases may occur, namely failure of the matrix due to transverse tension (opposing fibre buckling) or due to shear (synonymous with fibre buckling). Various methods of calculation are compared in Ref 120 for the theoretical establishment of the failure stress for these two cases, the individual authors starting from different preconditions. One, for example, makes the following assumptions:

- The fibre volume fraction is so small that there is no reciprocal influence of the fibres and the surrounding matrix can be regarded as infinitely large.

- The fibre diameter is so small compared with its length that the fibre can be regarded as infinitely long.
- Fibres and matrix behave homogeneously and isotropically; therefore the classic linear elasticity theory applies.
- No torsion moment is transmitted to the fibre by the matrix.

Another author regards the fibres as beams on an elastic bed. In both cases the stresses transmitted from the matrix to the fibre are determined by the three-dimensional elasticity theory. Other authors use two-dimensional formulae to determine the failure stresses. As indicated in Ref 120, these methods can only be regarded as first approximations to the practical circumstances.

A comparison between measured and calculated results by a two-dimensional theory from Ref 77 is shown in Fig 6.16b. This reveals that quantitative prediction of compressive strength is possible, but that there is also a relatively wide scatter of test results. Furthermore, with increasing proportion of fibres discrepancies between calculation and experiment grow even wider.

The methods for theoretical prediction of static strength of uni-directional laminates only make sense under uniaxial load. If a uni-directional laminate is stressed multiaxially the use of a fracture hypothesis is essential as a criterion for possible failure, just as it is for isotropic materials or multi-directional laminates.

6.4.2 Failure prediction for multiaxially loaded uni-directional and multi-directional laminates using fracture hypotheses

In isotropic materials a stress condition in any rectangular coordinate system produces the same material stress as that stress condition which arises from transformation in the principal stress directions (in return the shear stresses equal zero). However, this statement does not apply to anisotropic and orthotropic, *ie* non-isotropic, materials. When using a fracture hypothesis for these, therefore, it is not possible to calculate with principal stress conditions and the normal and shear stresses parallel to the anisotropy principal axes must be used to determine the stress on the material (cp Ref 599).

Comprehensive presentations of the fracture hypotheses normally used for fibre composites are to be found, *eg* in Refs 14, 45, 248, 557 and 595. In Ref 557 these are broken down into fracture hypotheses based on:

- mean stresses in a laminate,

- mean stresses in an individual layer, and
- micromechanical stresses.

Fracture hypotheses based on mean stresses in a laminate are not founded on physics. They are merely expansions of known methods for isotropic materials or are based on empirical experience.

With them it is only possible to say whether the laminate as a whole will fail or not. If mean stresses in a layer are considered it can be predicted which layer will fail but not whether failure occurs in the fibres or the matrix.

The equations which provide the criterion for the failure of a laminate or layer according to these fracture hypotheses generally describe, if a uniform stress condition is assumed, ellipsoids or ellipsoidal figures, the so-called 'fracture curves'. These are shown in Fig 6.17. Even for a uniform stress condition these are spatial and not plane curves as for isotropic materials, since due to the anisotropy the 'interaction' of the three stresses σ_1, σ_2 and τ_{12} has to be considered and transformation to principal stresses is not possible.

As indicated by Fig 6.17, the failure of a multi-layer composite (laminate) can either be predicted by:

- starting with the mean stresses in the laminate, determining experimentally the free parameters of the equation describing the fracture hypothesis and thus determining the fracture curve (Fig 6.17a); this then applies only to the particular laminate structure in question, or
- arriving in the same way at the fracture curve for an individual layer and obtaining the fracture curve for any required laminate structure by superimposing individual layers (Fig 6.17b).

If there are differing tensile and compressive strengths in a material, it may be possible to cover this, *eg* by shifting the fracture curve out of the coordinate origin (cp Ref 557). Incidentally, all fracture curves can also be presented as bands of plane curves with τ_{12} as parameter (cp, *eg* Ref 599).

If it is intended with the aid of the fracture hypothesis used to produce information not only as to whether failure or damage will occur, but also on the type of damage and the physical processes on which it is based, then the local stresses, *eg* between fibre and matrix (micromechanical stresses) must be considered. Investigations pointing in this direction are made in Ref 607, where the criteria are quoted for:

- fibre fracture,
- failure of matrix, and
- failure of interface fibre/matrix

} inter-fibre fractures

Similar expressions are found in Refs 557 and 608 where it is assumed that stress parallel to the fibres causes fibre fracture and a combination of the normal stresses and the shear stress the inter-fibre fractures.

A difficult problem in the use of every fracture hypothesis is the definition of the damage in itself. Thus it is possible to imagine a case where a layer lying across the direction of loading fails completely but the load-carrying behaviour of the laminate does not decrease in practice (for example failure of the 90° layer of an axially loaded $[0^{\circ}, 90^{\circ}]_s$ laminate). On the other hand, the shear carrying capacity of a layer or parts of a layer may equal zero due to matrix failure while the fibres transmit the normal tensile stresses satisfactorily. In a case like this it is very difficult to judge the remaining load-carrying capacity of a component reliably.

If the use of a fracture hypothesis yields first damage (*eg* the failure of one layer) there are two possible means of proceeding, namely:

- one can either take the view that there is total failure of the laminate (under static load which is further increased this will in fact frequently be the case, since further damage will occur after a short time and the static fracture of the entire laminate will follow very rapidly),

- or one can regard the failure of a layer as 'first damage', re-determine the elasticity constants of the layer in question and the whole laminate, calculate the new stress distribution and determine the next damage occurring by renewed use of the fracture hypothesis. Theoretically this procedure can be repeated up to prediction of total failure. Although in practice it is likely to prove very complicated to describe such a damage mechanism realistically with the aid of mathematical models, particularly if, for example, the symmetry of a laminate is lost on failure of a layer, which causes the material law to change considerably and thus bending stresses and deformations to occur in a purely axially loaded laminate (cp Fig 6.4).

In conclusion the following can be asserted on the use of fracture hypotheses for fibre composite materials:

- The fracture hypotheses used most frequently in the literature are based on mean stresses in a laminate. They are in part expansions of the fracture hypotheses applicable to isotropic materials.
- The evaluation of the individual fracture hypotheses is very difficult and associated with a large number of experiments. Comparative investigations are only incidentally available in the literature evaluated.
- In order to obtain more information on the physical processes in the failure of a laminate there is a tendency to regard first the damage in a single layer (failure of the fibres or matrix) and draw from this conclusions on the laminate.
- To determine the constants in the equation describing the fracture hypothesis a greater or lesser number of experiments with different combinations of the stresses σ_1, σ_2 and τ_{12} is necessary in every case. The aim should be to find a fracture hypothesis applicable to any required fibre/matrix combination if the relevant constants are entered in the equation.
- Since laminates as a rule are treated as plane load-carrying structures in stress analysis, the use of fracture hypotheses has hitherto also been restricted to plane stress conditions.
- So far there is no fracture hypothesis in existence which applies to failure under cyclic loading. Taking over fracture hypotheses for static loading does not appear to be justified. Thus in experimental investigations, for instance, a rise (and no drop) in static residual strength of notched specimens has been observed, an indication of different damage processes under static and cyclic load.

6.5 Conclusions

As already mentioned in the introduction, the experimentally established elasticity constants and breaking strengths are subject to greater scatter in fibre composites than in isotropic materials (according to Ref 558 a scatter of ±15% must be expected). This is probably due primarily to:

- different type and frequency of faults (pores, air bubbles, etc), and
- different fibre and matrix content

of the individual specimens.

In mathematical calculations it is usual to assume ideal material properties. The assumptions necessary as a rule have been extracted from a table in Ref 339:

- The fibres are parallel and of infinite length.
- There is an ideal bond between fibres and matrix.
- The laminate contains no air bubbles or inclusions of foreign matter.
- Fibres and matrix consist of homogeneous, isotropic and linear-elastic materials.
- The uni-directional laminate or the individual layer is quasi-homogeneous *ie* no stresses occur between fibres and matrix (microstresses).
- The bonding of the individual layers of a laminate is ideal.
- As a rule a uniform stress condition is assumed.

Differences between theoretical predictions and test results must in many cases be ascribed to these assumptions. Departure from these ideal states, even if it is possible at all, is very costly (change to FEM with three-dimensional elements, etc). The phenomena which cannot be covered theoretically by the assumptions quoted (*eg* inclusions, air bubbles, non-adhering fibres, interlaminar stresses, etc) in many cases, however, determine the fatigue strength of the laminate.

For this reason the following aspects which can be decisive for fatigue strength do not appear to have been investigated adequately:

- The literature evaluated deals with relatively simple laminates with few layers only as a rule. For many effects (*eg* bending deformations on failure of a layer, interlaminar stresses, thermal stresses, etc) it is therefore not known whether they have the same significance for the behaviour and failure of the laminate in composites used in practice.
- So far there are no findings on the effect of micromechanical stresses (*eg* between fibre and matrix and in the vicinity of defects) on the fatigue strength of a laminate.
- The dependence of the fatigue strength of a laminate on the interlaminar stresses present is not known.

- Triaxial stress conditions have hardly been examined, either in stress analysis (exception Ref 519, preliminary work in Refs 246 and 589) or in the development and testing of fracture hypotheses.
- There is no fracture hypothesis describing the failure of a laminate under cyclic stress.

In the interests of optimum dimensioning of a laminate from the point of view of fatigue strength it appears logical to extend future theoretical and experimental research to these problems in increased measure.

6.6 References

The literature evaluated in the preceding sections is classified according to contents. There is no order of priorities, but the papers mentioned in the text have been underlined. References with finite element methods are shown in round brackets.

Section 6.1

General survey of arithmetical methods for fibre composite materials:

339, 600, 606.

Section 6.2.2

Determination of elasticity constants of an individual layer or a unidirectional laminate from the fibre and matrix properties:

79, 109, 171, 262, 337, 339, 376, 401, 555, 557, 580, 600, 603, 606.

Section 6.2.3

Determination of elasticity constants of a multi-directional laminate:

46, 85, 122, 225, 262, 339, 502, 546, 557, 597, 600, 603.

Section 6.3.1

Stress analysis for uniform cross-sections:

General presentation:		345, 559, 594, 596, 602, (546, 558, 561).
Transverse beams	:	78, 571, 583, 596, (52).
Discs, plates:	:	65, 82, 98, 558, 596, 597, (563).
Dishes	:	80, 109, 123, 262, 400, 519, 578, 598, (479).

Interlaminar stresses:

118, 250, 262, 335, 519, 526, 560, 562, 577, 587, 590, 638, (564).

Section 6.3.2

Stress analysis in the vicinity of notches:

62, 96, 117, 257, 259, 260, 262, 420, (63, 117, 253, 258, 263).

Section 6.3.3

Hygrothermal behaviour:

171, 337, 401, 556, 564, 565, 566, 567, 569, 572, 573, 579, 593, 601, 638.

Section 6.4.1

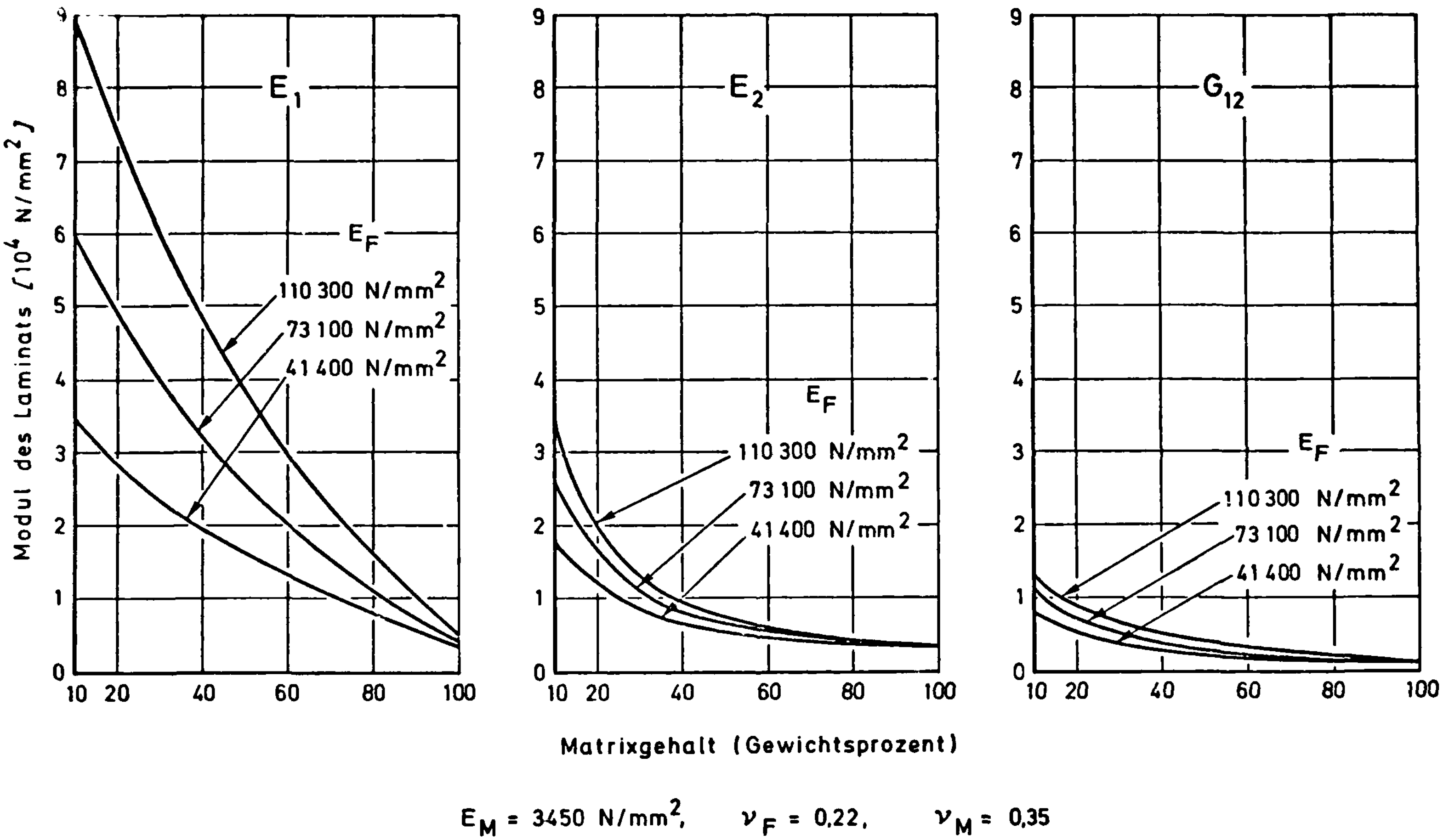
Failure prediction for uni-directional laminates:

22, 26, 54, 74, 77, 107, 120, 176, 180, 211, 336, 401, 424, 575.

Section 6.4.2

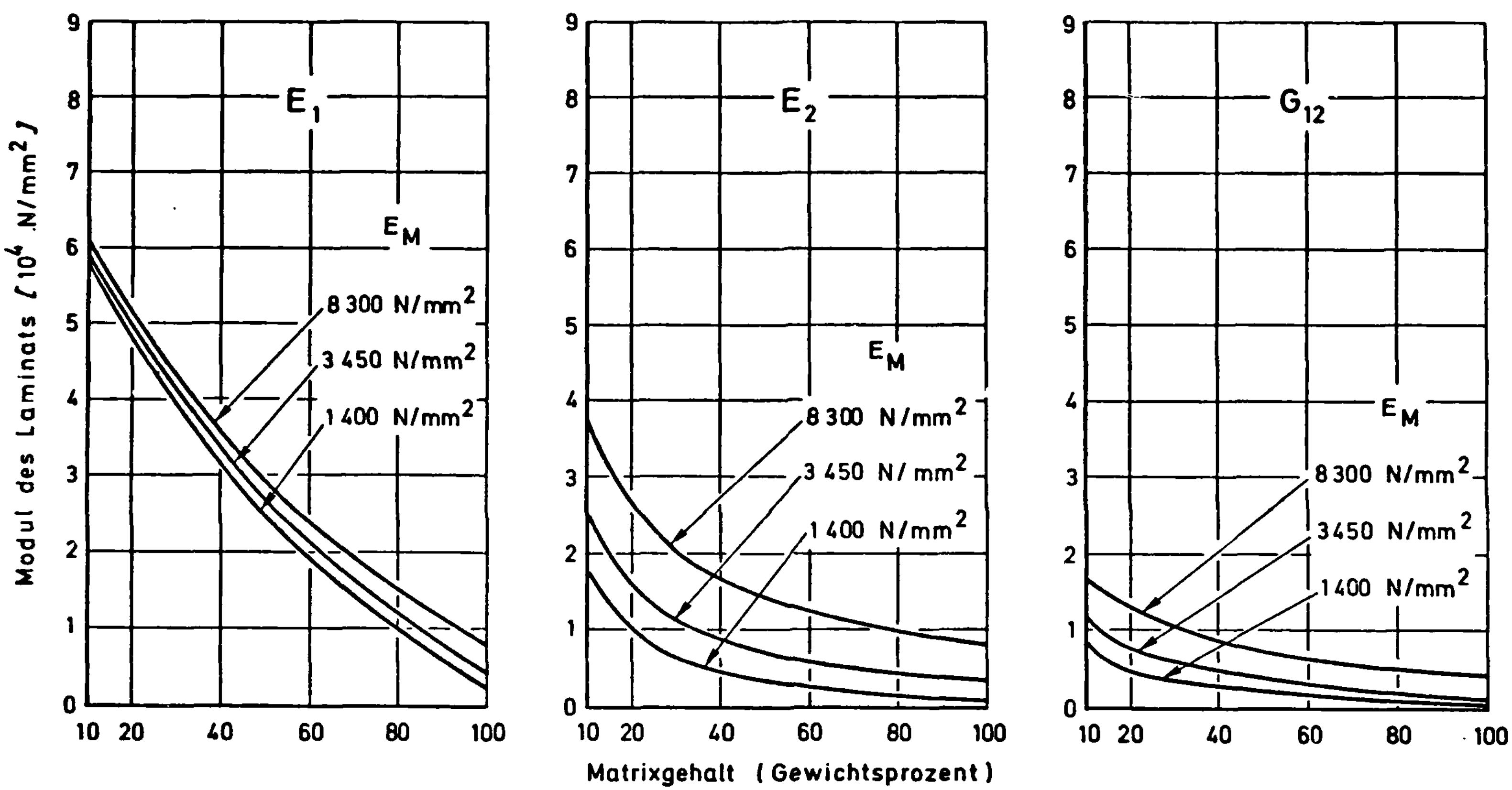
Fracture hypotheses:

14, 24, 26, 45, 62, 86, 109, 173, 248, 262, 264, 275, 336, 337, 344, 352, 418, 419, 424, 430, 503, 524, 557, 558, 595, 599, 607, 608.



Key:
Matrixgehalt (Gewichtsprozent) = matrix content (percentage weight)

Fig 6.1 Effect of fibre modulus of elasticity on the moduli of a unidirectional laminate (parameter study as Ref 339)



$E_F = 73100$ N/mm^2, $\nu_F = 0,22$, $\nu_M = 0,35$

Fig 6.2 Effect of the matrix modulus of elasticity on the moduli of a unidirectional laminate (parameter study as Ref 339)

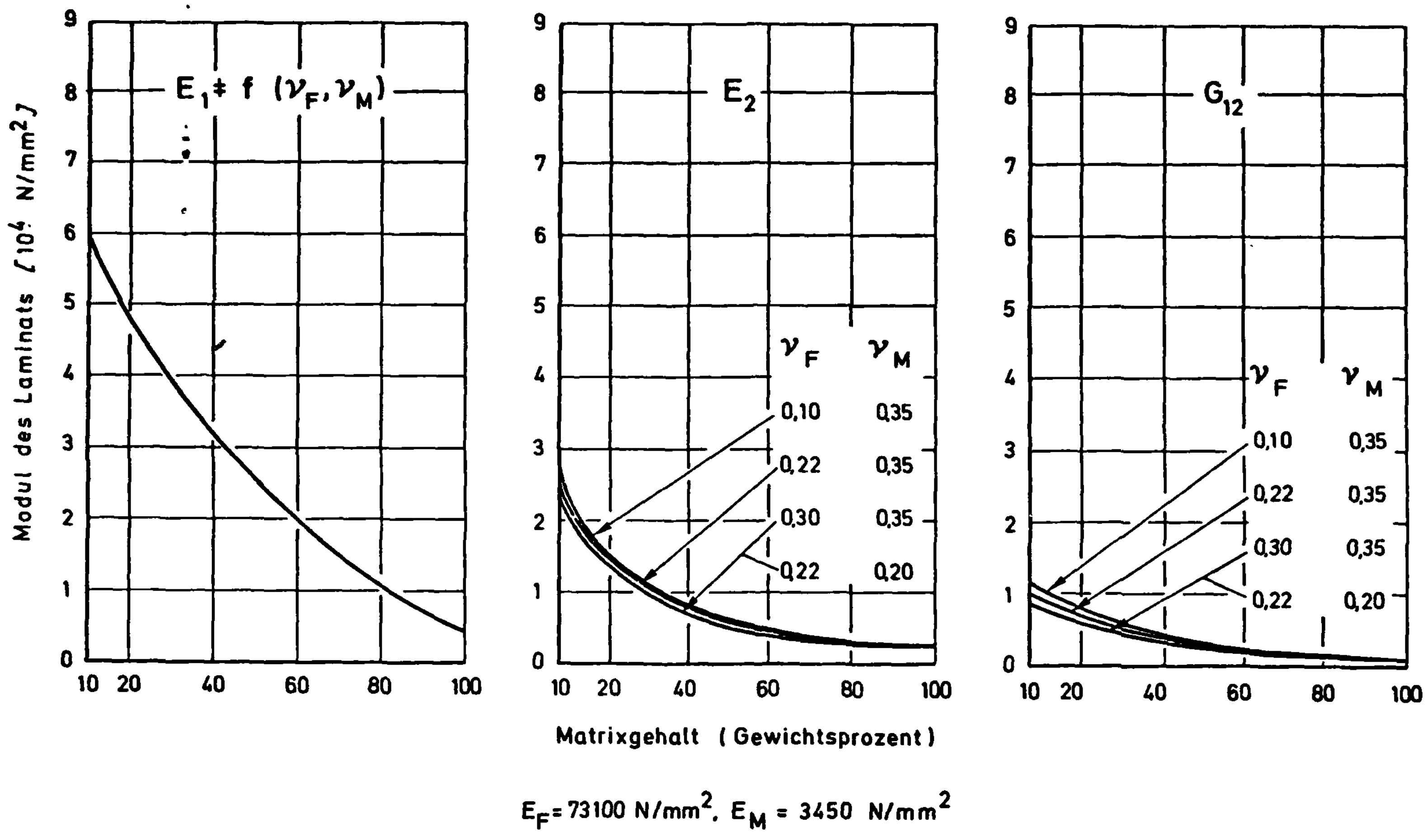
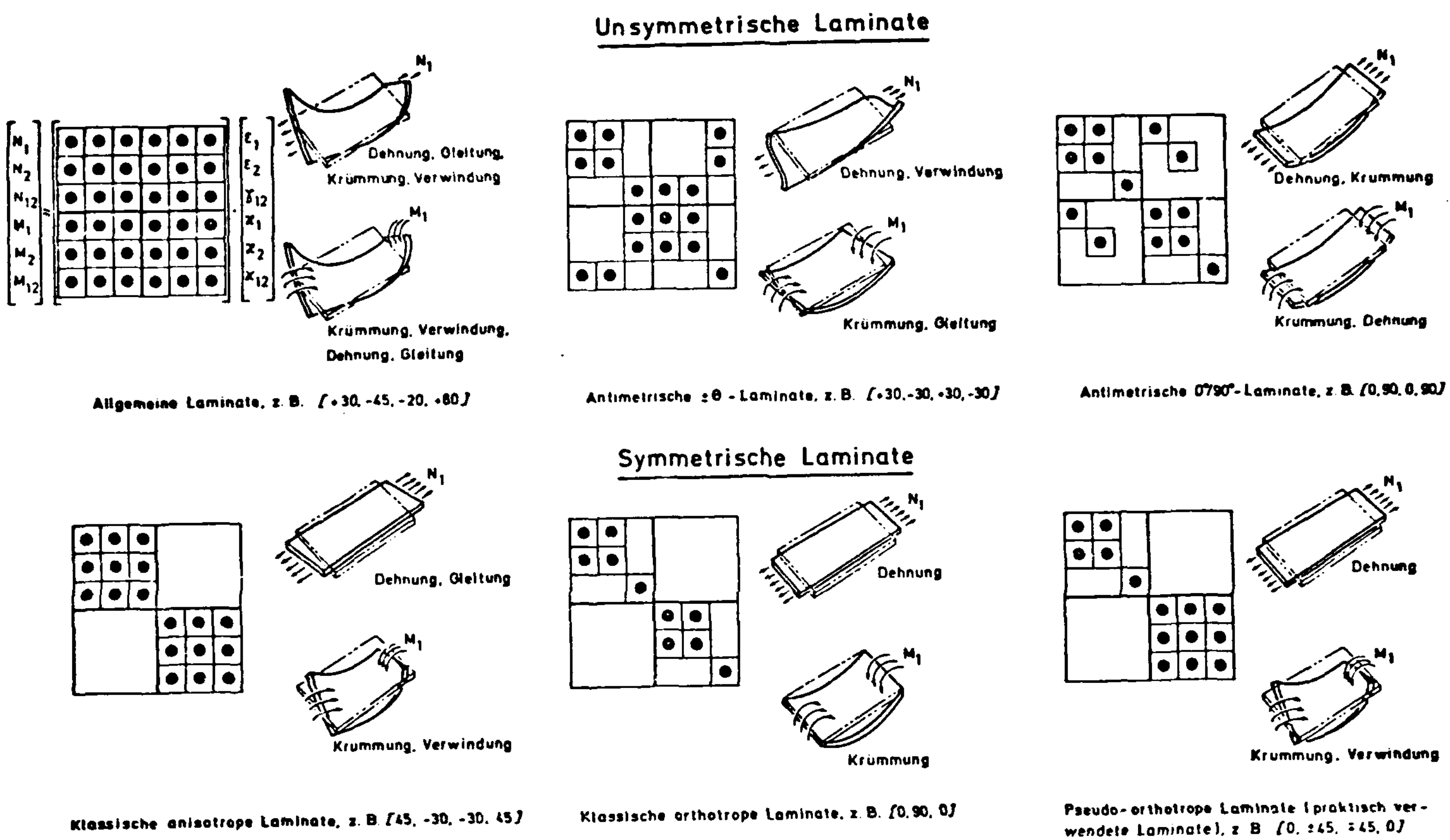


Fig 6.3 Effect of ν_{fibre} and ν_{matrix} on the moduli of a unidirectional laminate (parameter study as Ref 339)



Key:
Dehnung = extension
Gleitung = slip
Krümmung = bending
Verwindung = twist
Praktisch verwendete Laminate = laminates in use in practice

Fig 6.4 Material law for deformation behaviour under tensile and bending load for differently constructed laminates as Ref 546

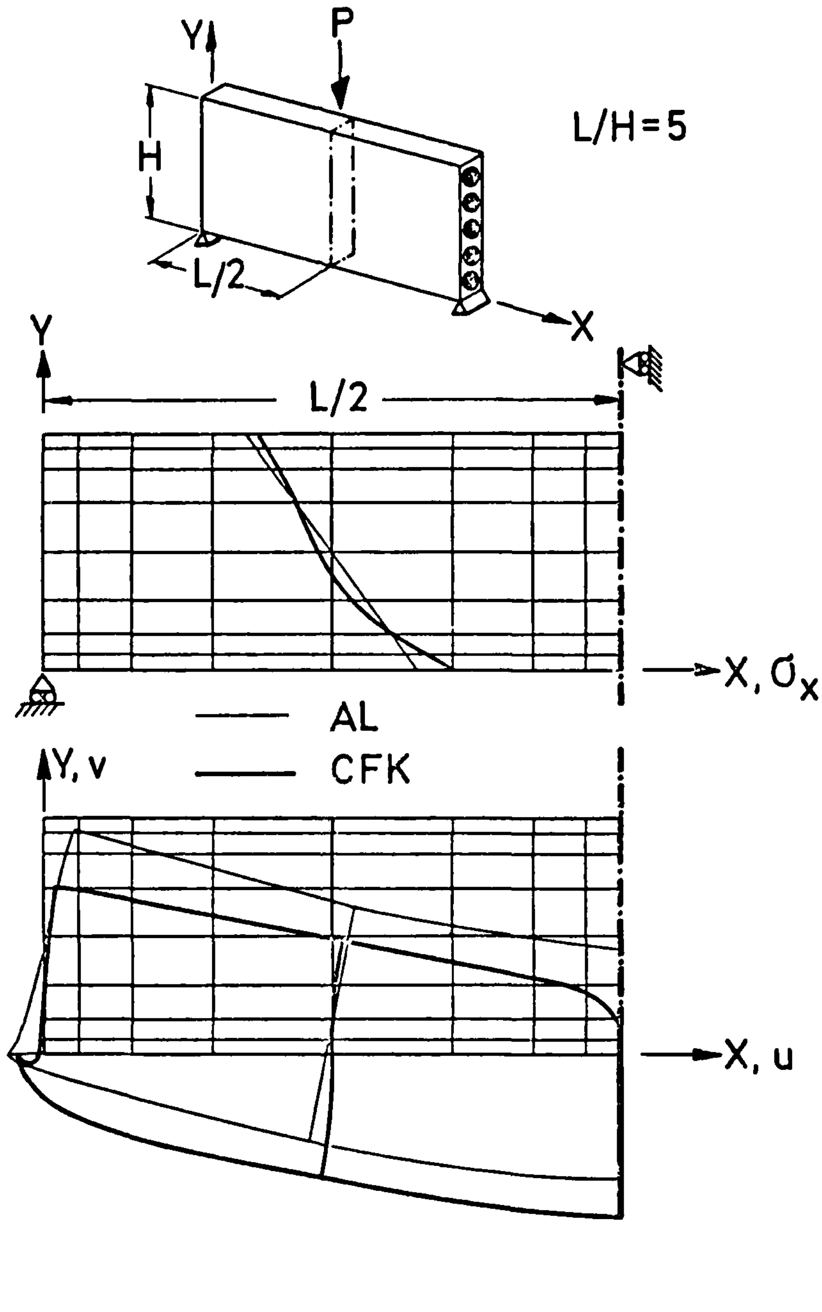
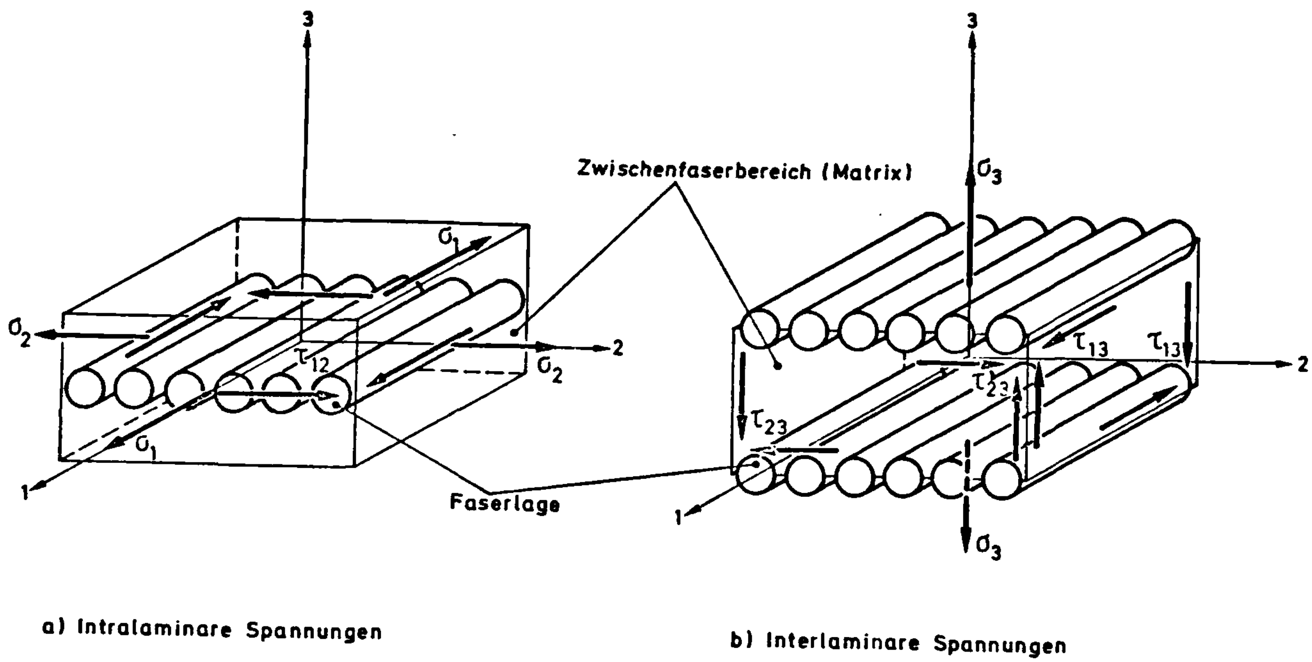
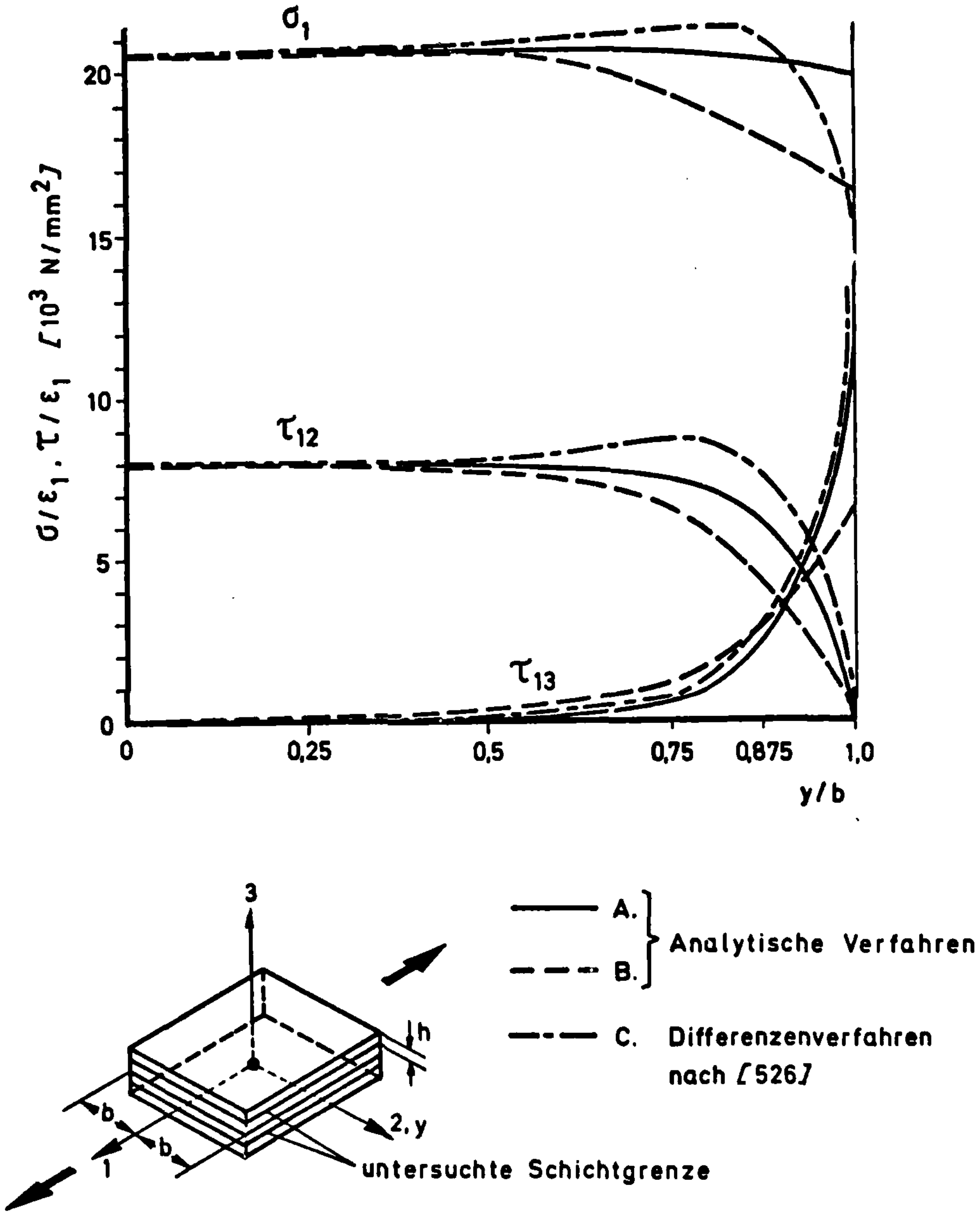


Fig 6.5 Normal stress and deformations in a bending specimen of aluminium/carbon fibre composite as Ref 558



Key:
Zwischenfaserbereich = inter-fibre area
Faserlage = fibre position

Fig 6.6 Definition of normal and shear stresses in the laminate. (Shear stresses are marked in the plane in which they produce distortion)



Key:
Untersuchte Schichtgrenze = layer boundary investigated

Fig 6.7 Stress distribution over the width of a $[\pm 45]_s$ laminate as Ref 335

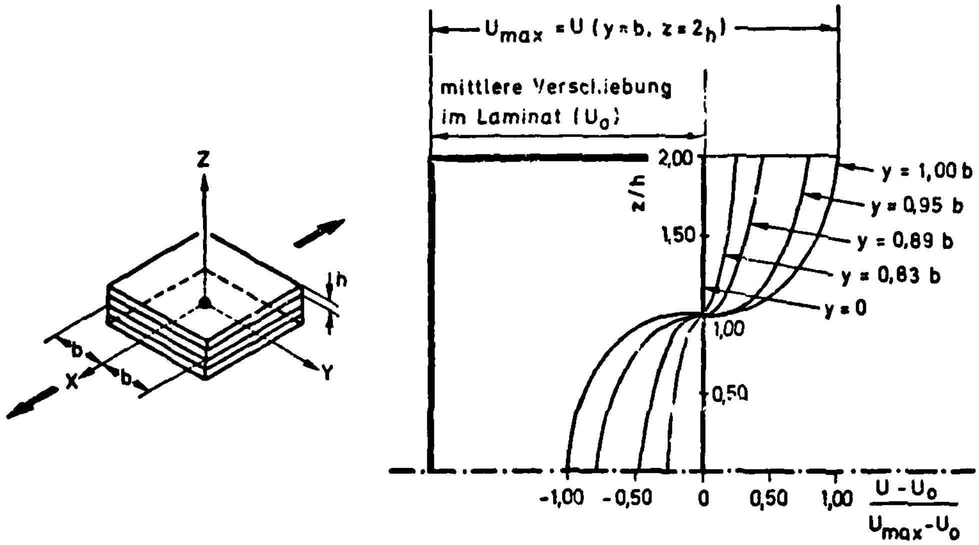


Fig 6.8(a) Axial displacement distribution over the laminate thickness of a $[\pm 45]_s$ laminate as Ref 526

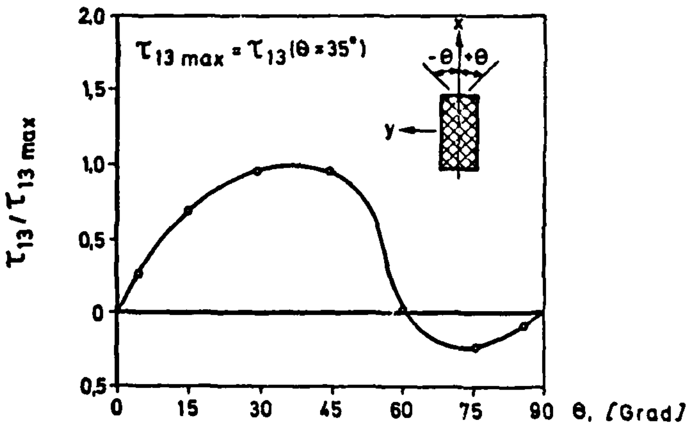
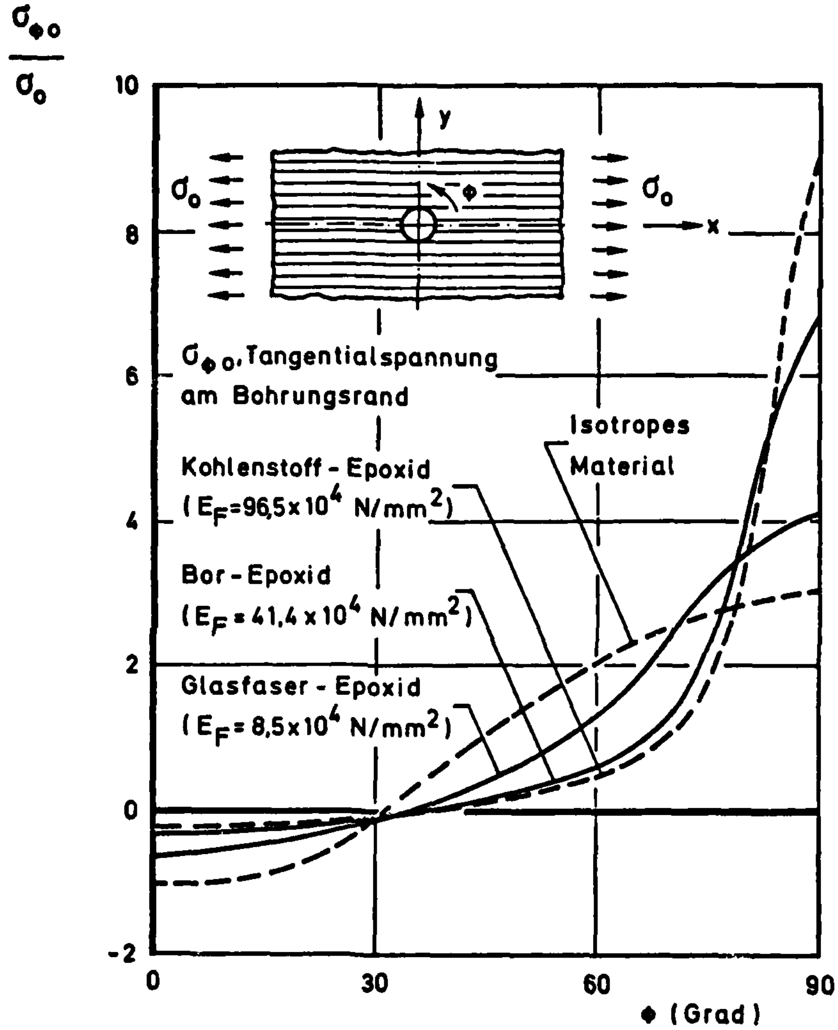


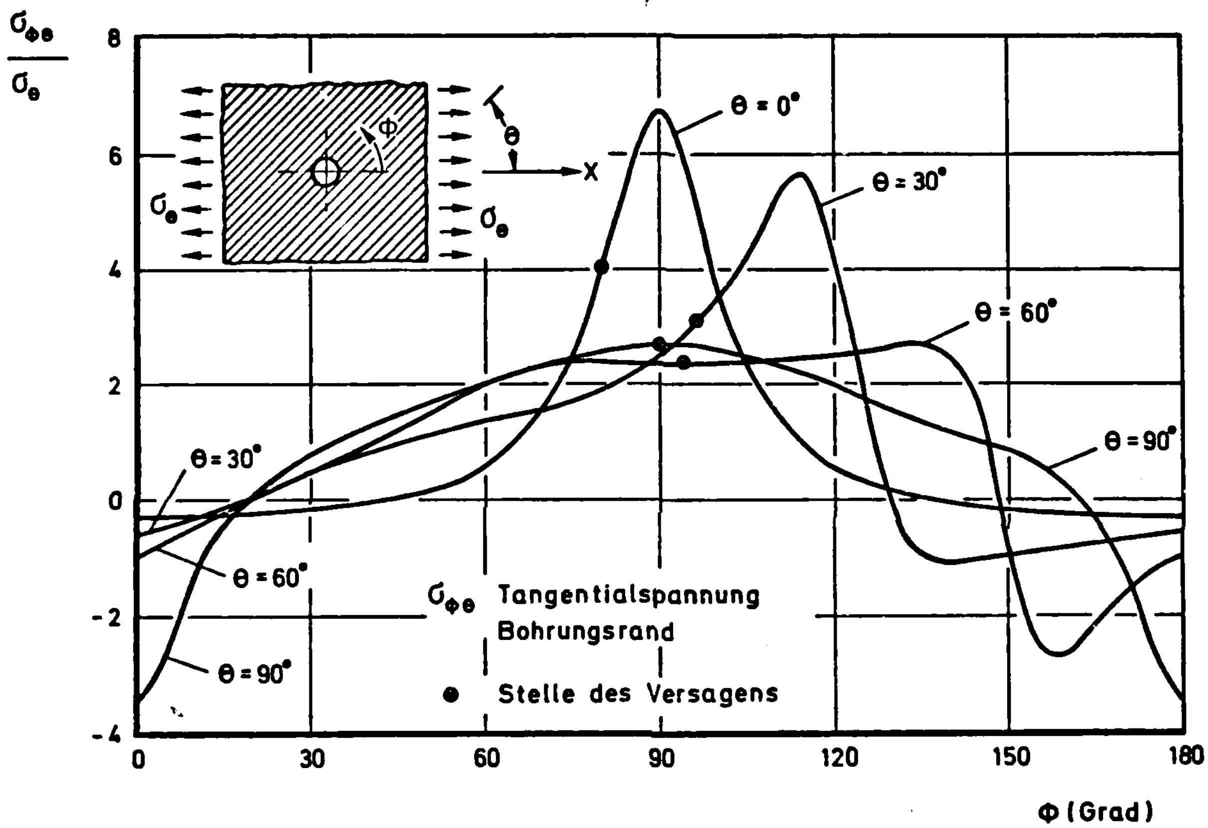
Fig 6.8(b) Interlaminar shear stress τ_{13} as a function of fibre orientation of a $[\pm\theta]$ laminate under axial loading as Ref 526



Key:

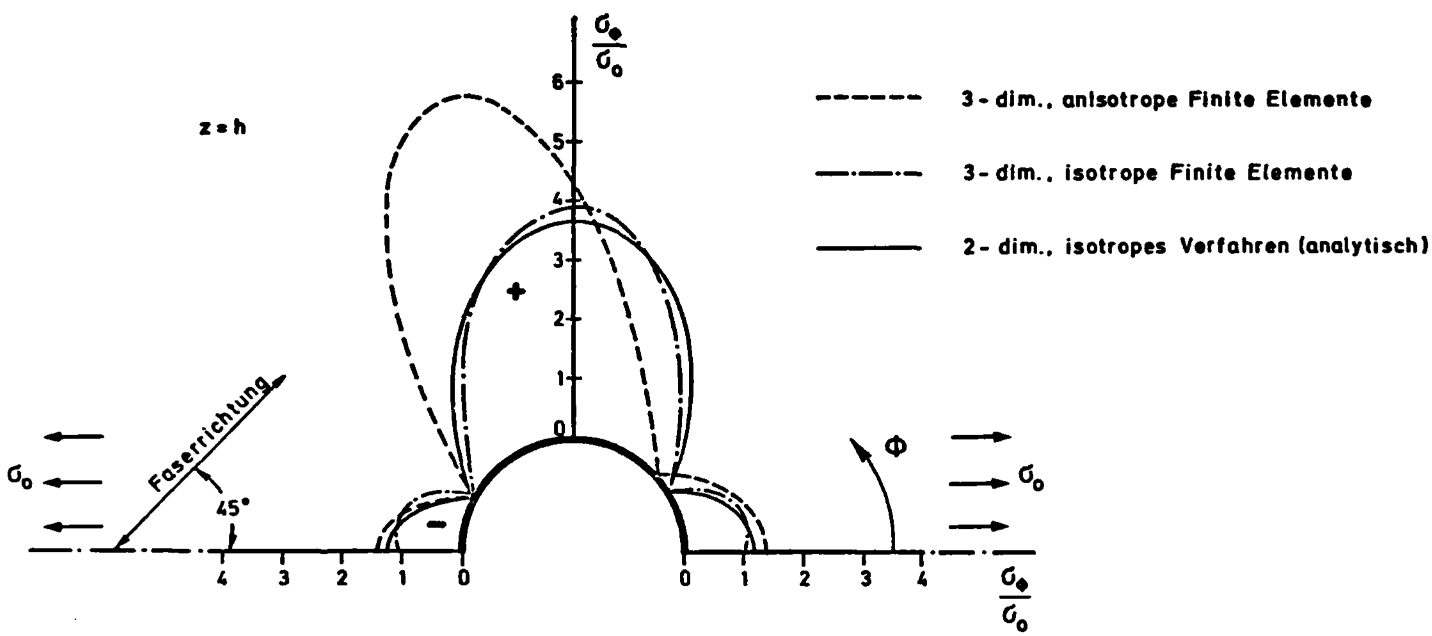
Tangentialspannung am Bohrungsrand	= tangential stress at edge of hole
Kohlenstoff	= carbon
Bor	= boron
Glasfaser	= glass fibre

Fig 6.9 Effect of fibre material on the stress distribution around a circular hole in a unidirectional laminate as Ref 62



Key:
Stelle des Versagens = location of failure

Fig 6.10 Stress distribution as a function of fibre orientation in a unidirectional laminate (boron epoxide) as Ref 62



Key:
Faserrichtung = direction of fibre

Fig 6.11 Tangential stresses at edge of hole in a $[\pm 45]_s$ boron epoxide laminate as Ref 253

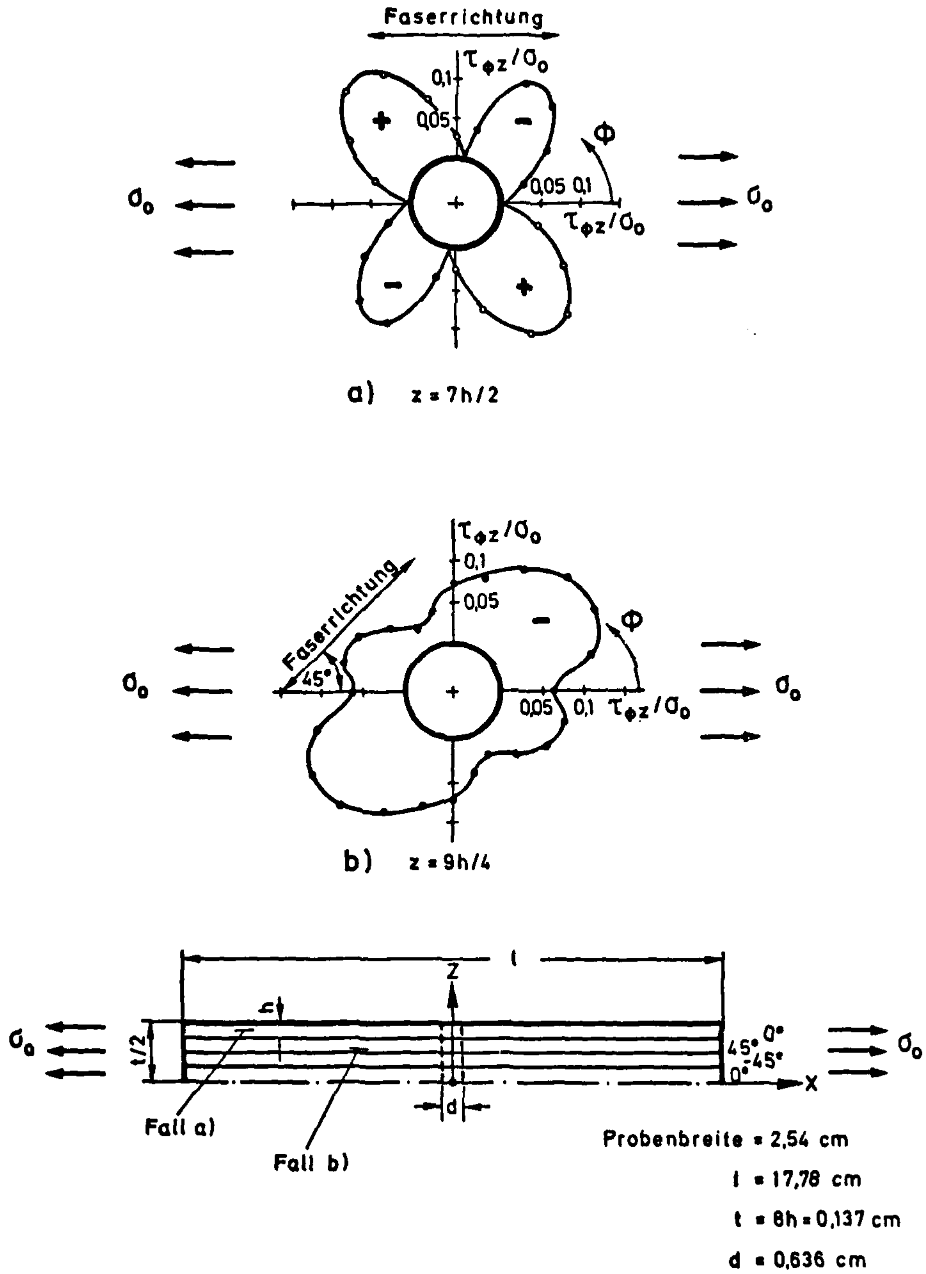
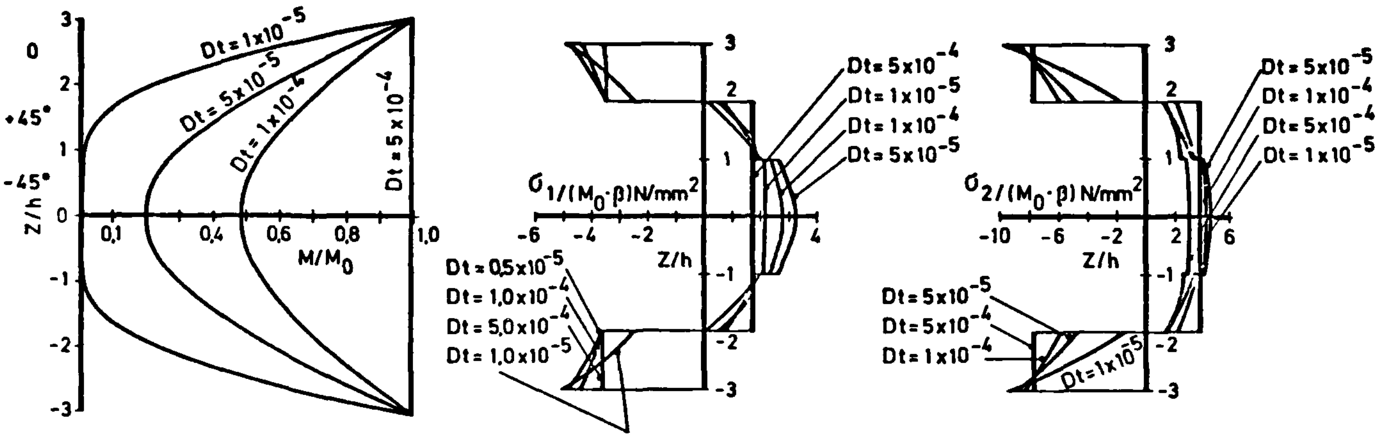


Fig 6.12 Variation of interlaminar shear stress $\tau_{\phi z}$ at edge of hole in a $[0/\pm45/0]_s$ laminate as Ref 258



M : **Feuchtigkeitskonzentration in %**

M_0: **Feuchtigkeitskonzentration der Umgebung in %**

β : **Ausdehnungskoeffizient infolge Feuchtigkeitsaufnahme ($\beta = 6{,}67 \cdot 10^{-3}$/% Änderung der Feuchtigkeitskonzentration**

D : **Diffusionskoeffizient**

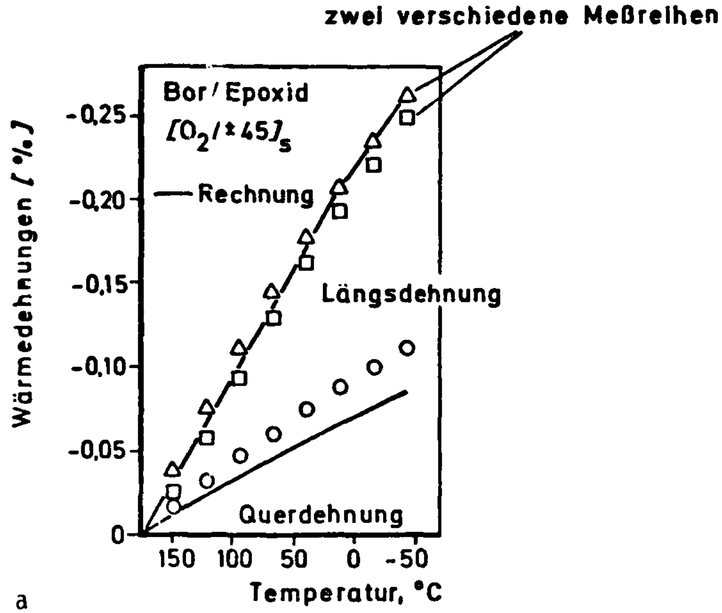
t : **Zeit**

$\left.\begin{array}{l}D\\t\end{array}\right\}$ **Dt: "dimensionslose" Zeit**

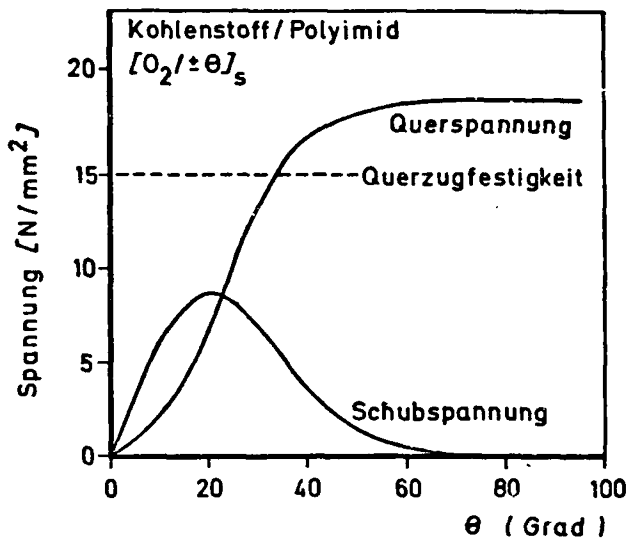
Key:

Feuchtigkeitskonzentration der Umgebung = moisture concentration of environment
Ausdehnungskoeffizient infolge Feuchtigkeitsaufnahme = coefficient of expansion due to moisture absorption
Änderung = alteration
Dimensionslose Zeit = dimensionless time

Fig 6.13 Moisture and stress distribution in a $[0^{o}/\pm45^{o}]_s$ laminate (carbon epoxide) during moisture absorption as Ref 566



a



b

Key:
Zwei verschiedene Messreihen = two different test series
Längsdehnung = longitudinal expansion
Querdehnung = transverse expansion
Querspannung = transverse stress
Querzugfestigkeit = transverse tensile strength
Schubspannung = shear stress

Fig 6.14(a) Thermal expansions on cooling of the laminate after curing as Ref 579

Fig 6.14(b) Theoretically determined stresses in the 0^o layers in the cooling process after curing as Ref 569

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LITERATURE RESEARCH ON THE MECHANICAL PROPERTIES OF FIBRE COMPO--ETC(U)
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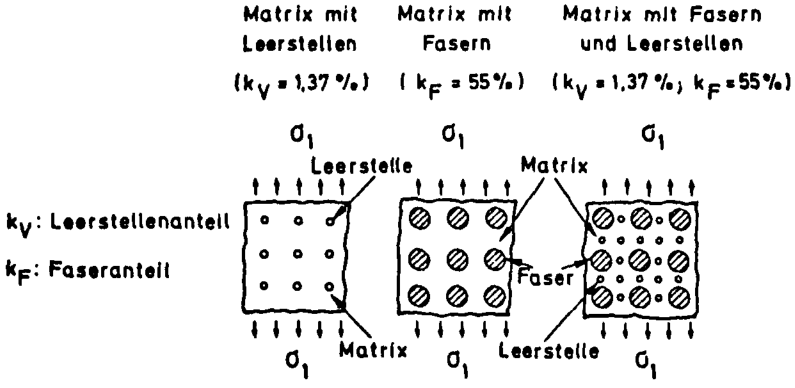
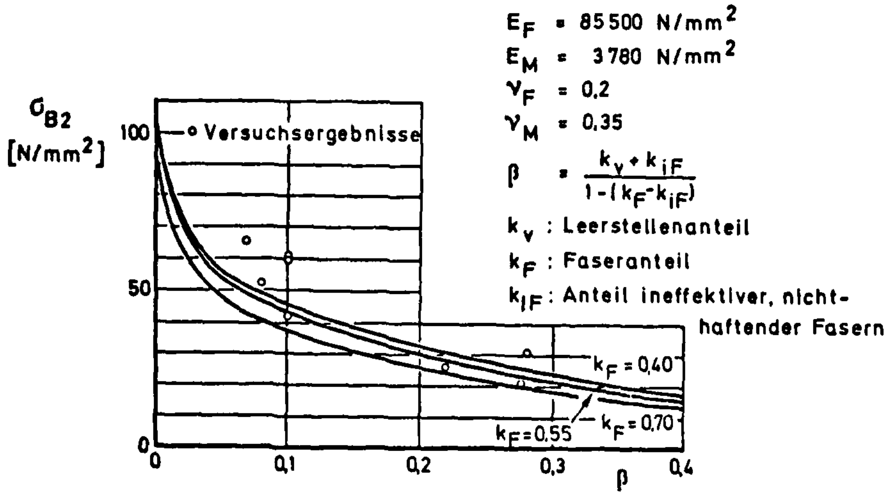


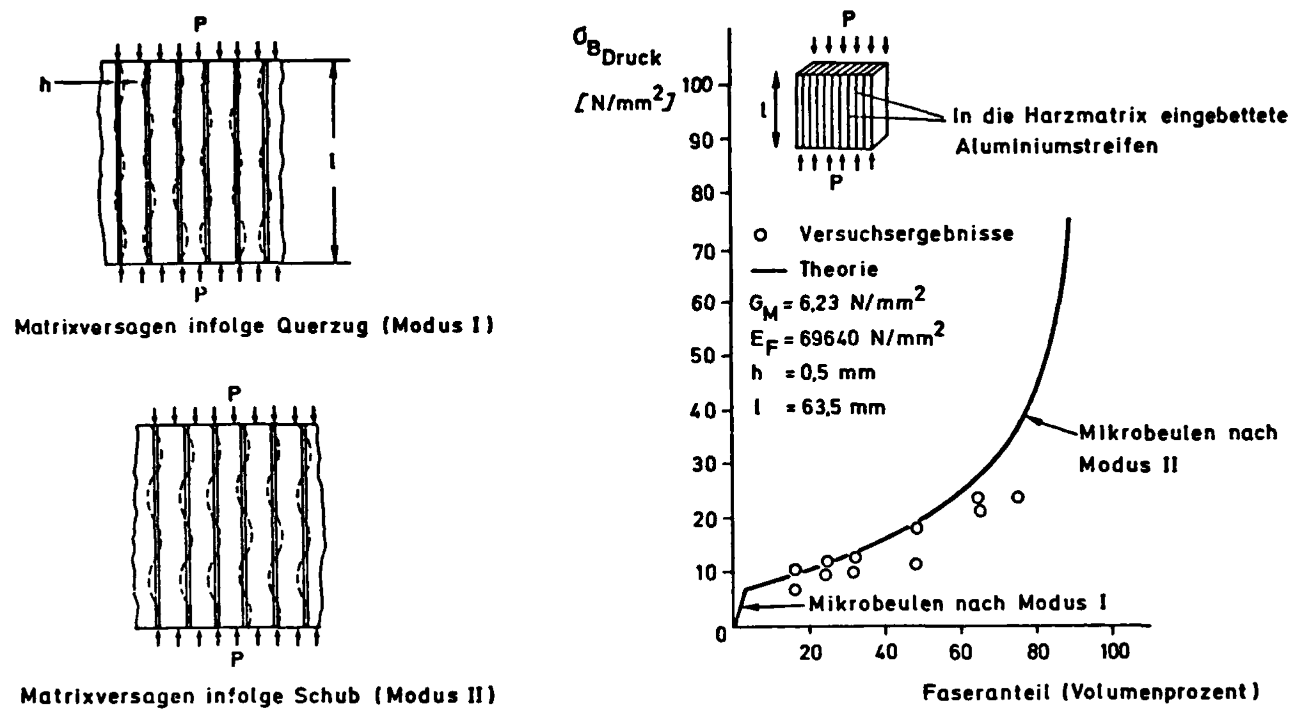
Versuch σ_{max}/σ_1 =	3,0	1,62	3,98
Theorie σ_{max}/σ_1 =	3,03	1,696	4,11

Fig 6.15(a) Stress concentration in the matrix because of fibres and voids as Ref 424



Key:
Leerstelle = void
Faser = fibre
Anteil ineffektiver, nicht-haftender Fasern = proportion of ineffective non-adhering fibres

Fig 6.15(b) Transverse tensile strength σ_{B2} as a function of fibre and void content, unidirectional laminate (S glass fibre/epoxide) as Ref 424

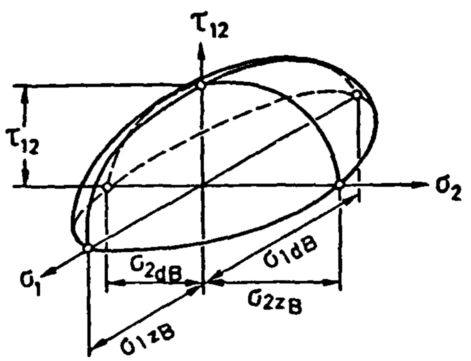


(a) Types of failure

(b) Comparison measurement-calculation

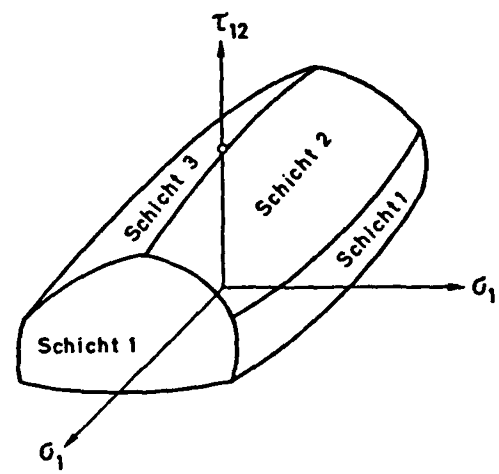
Key:
Matrixversagen infolge Querzug = matrix failure due to transverse tension
Matrixversagen infolge Schub = matrix failure due to shear
In die Harzmatrix eingebettete Aluminiumstreifen = aluminium strips embedded in the resin matrix

Fig 6.16 Microbulges due to compressive loading as Ref 77



(a) Fracture curve for mean laminate stresses or mean stresses in a single layer

1,2: principal orthotropy axes
d: compression
z: tension
B: fracture



(b) Fracture curve for mean layer stresses in a multi-layer composite (from superimposing fracture curves for the individual layers)

Fig 6.17 Fracture curves for plane stress conditions as Ref 557

7 DESIGN OF FIBRE COMPOSITE STRUCTURES

J. Franz

Contents

7.1 Introduction

Very comprehensive knowledge of the behaviour of a material under the stresses occurring in operation and the ruling environmental conditions is required for the design of structures.

For conventional metal materials there are very many empirical data available. On going over to new materials and methods of construction the question must be posed in every case as to how far the use of conventional procedures is permissible or logical. The use of fibre composites involves, apart from problems of stress determination, many other questions regarding damage tolerance and fail-safe behaviour which play a decisive part in designing to the requirements of fatigue strength. For this purpose the following questions inter alia need clarification for fibre composites:

- in principle the course of damage development, damage propagation and final failure,
- the effect of material composition, laminate structure, manufacture and environment on fatigue behaviour,
- the determination of comparative fatigue strength data for different materials,
- the formulation of methods for predicting permissible stresses and the performance of the components as well as
- the simulation and estimation of the effect of practical load sequences.

The knowledge gained theoretically and above all experimentally of these relationships determines the design process and the stressing of fibre composite structures in view of certain design aims, such as:

- optimum static strength,
- optimum stiffness,
- optimum fatigue strength, etc.

The concepts for composites on which these considerations are based will be discussed further below.

7.2 Aspects of static design

Apart from achieving the necessary level of strength, the static design of a structure must in many cases ensure adequate stiffness (*eg* in the wing or tail units of an airplane where aerodynamic conditions would change to an inadmissible extent under excessive distortion). In spite of this, the investigations considered within the framework of the present literature evaluation are as a rule based on the failure criterion 'static fracture'. A great deal of research is concerned with statistical methods to determine the probability with which static fracture will occur under a certain load. A review of some models and procedures for the statistical treatment of the strength behaviour of fibres, fibre bundles and uni-directional fibre-matrix combinations will be found in Ref 83. In a concluding summary the author states that the methods described are at an early stage and that extensive theoretical and experimental research remains to be done in this area.

The determination of failure probability of redundant structures is treated in a very clear manner in Ref 157. Fig 7.1a&b shows a comparison of a series system of load-carrying members with a parallel redundant system. In case (a) the entire structure breaks down on the failure of one member (weakest link theory, *ie* for failure prediction only the failure of the weakest member need be considered); in case (b) the parallel members assume the task of the failed component. In case (c) Fig 7.1c it is assumed that one member only is in use while two others are ready as replacements in case of failure ('standby' redundant system), which applies, for instance, to the electronic equipment of an aircraft.

Considerations similar to those applying to structures composed of individual components can also be employed for the redundancy behaviour of a composite material itself which is built up of a very large number of parallel fibres. Case (b) presents itself for determination of the failure probability: on failure of one or several fibres (or fibre bundles) the fibres running in parallel take over the free load. In consequence of the now increased load the probability of failure of further fibres has become greater. By this method the probability of the total failure of a redundant material structure can be predicted as a function of the load, while the relatively greatly simplified models, such as the fibre bundle theory, are primarily applicable only to uni-directional laminates. Here and particularly for more complicated material (*eg* multi-directional laminates) a decisive part is played by the sequence of faults occurring or 'failure path', as shown in Fig 7.1d. The structure illustrated collapses after the failure of four diagonal struts if the failure process runs in the sequence A, B, C, D, but after failure of two struts if

B and D fail consecutively. This case can also be transferred to a laminate. If, for example, a certain number of neighbouring fibres fail the failure probability of the whole laminate is greater than if the same number of fibres fail but they are evenly distributed over the laminate (because the load on each neighbouring fibre is much less).

The authors of Refs 419 and 558 treat the problem of static design from a physically demonstrable and not a statistical point of view. They use the term 'reserve factor' where

$$\text{failure stress} = \text{stress condition} \times \text{reserve factor}$$

applies to the failure stress of each individual layer in a laminate.

The reserve factor is therefore a safety factor against fracture (for each individual layer). A design is then considered optimum when under the operational load occurring the reserve factors are the same for all layers, that is all layers are designed with the same safety factor against static failure. That this is difficult to achieve in many cases emerges from Fig 7.2. The laminate considered here, constructed from equal numbers of 0° and 90° layers is far from fully utilised if a reserve factor of 1.5 is provided for the 90° layers. The 0° layers are then loaded to only a fraction of their capacity. Better utilisation of the laminate can be achieved by raising the number of 90° layers or adding layers with different orientation angles (*eg* $\pm 45^{\circ}$) since then the 0° layers take a higher proportion of the load because of their greater stiffness (*ie* the breaking stress of the laminate drops, as can be seen in Fig 7.2). If the failure of a laminate is assumed to be the failure of the weakest individual layer (weakest link), this limits the admissible strain for this laminate. On the other hand, in many cases the component is still in a position to fulfil its purpose without noticeable adverse effects after failure of individual constituents because of its redundant structure. More far-reaching considerations of the problems raised here will be found inter alia in Ref 152. This also contains various criteria for the static design of fibre composite components, taking special account of requirements for their use in military aircraft.

7.3 Design for fatigue strength

7.3.1 Damage accumulation

When a conventional structure is to be designed for a particular period of use extensive experiments are required in the present state of technology. In the ideal case there would be an operational load test on the actual component. In practice, however, conclusions are frequently drawn from available test results on specimen bars and with simple load sequences (*eg* single-state tests) as to the fatigue life to be expected in practical operation for reasons of time and cost (at least in the preliminary design phase). This conclusion is reached as a rule by using the Miner Rule (or relative Miner Rule if the available test results refer not to single-stage but to multi-stage or flight-by-flight tests with other than the expected operational load sequence). Although life predictions by these methods are relatively imprecise, the procedure has proved itself in practice since in the decades in which the Miner Rule has been applied to different types of materials, components and load sequences, extensive empirical knowledge has been gathered on the deviations between actual life and prediction.

For this reason it is a natural step to use the Miner Rule or the relative Miner Rule for fibre composites also, at least as long as there is no better method of predicting the life of a structure. On the other hand very little definite data on the validity of the Miner Rule for composites are available so far. Tests in LBF of unnotched CFC specimen bars[785] show that in this case prediction compared with test is on the unsafe side by about a factor of 4. A similar picture emerges from Ref 532 for notched bars. In the case of the bonded joints investigated in Ref 716 prediction for the lower load levels is also on the unsafe side, whereas it is on the safe side for high load levels.

No further results of Miner calculations for fibre composite materials are contained in the literature evaluated because insufficient fatigue strength investigations have been performed so far with flight-by-flight loading. It is clear from this that extensive research is still required to create the conditions for use of the Miner Rule in the design of a structure. In the present state of technology acceptable design for fatigue strength is only possible by carrying out operational load tests. In each case, if single-stage test data is available, a Miner calculation should be carried out in order to gather experience on the applicability of the method to fibre composite components.

7.3.2 Wear-out model

In the design of metal structures it is usually assumed that the strength of the material will not change during the period of use (fracture mechanics view points are an exception, where the static strength of a component decreases with the size of the crack). Experiments on composites have shown that this assumption is not valid. As a rule a drop is observed in residual strength depending on the number of cycles sustained. This drop can be explained by the generation and propagation of faults in the laminate. In the case of notched specimen bars, in contrast, there is at first frequently a distinct rise in residual strength after a low number of cycles, which is most probably due to the reduction of notch stresses at the edge of the hole. Only at higher numbers of cycles does the residual strength decline again. Relative test results are shown in Fig 7.3. It will be noted that the difference in static strengths between unnotched and notched specimens gradually disappears with increasing numbers of cycles.

These effects, particularly of course the drop in residual strength, must be taken into account in the design of fibre composite structures. Safety from static failure declines steadily with increasing length of service. This relationship can be indicated theoretically by the wear-out model described in, *eg* Ref 113 for which the following assumptions apply:

- failure of the material is caused by faults which were present from the start (*eg* through manufacture),
- the faults grow in characteristic fashion as determined by:
    - the properties of the material,
    - the type and size of the stresses in the vicinity of the fault,
    - the load sequence, and
    - the environmental conditions,
- the load causing failure of the structure is a declining function of the size of the damage.

With the statistical equations used in Ref 113 the decline in residual strength as a function of the number of cycles applied can be stated theoretically, and can include the effects of temperature, moisture, creep, etc. Failure of the component occurs when the load applied exceeds the remaining static strength.

Fig 7.4 shows an example of a bonded joint under multi-stage loading, comparing the theoretically predicted and experimentally determined drop in residual strength and the corresponding endurance line. This reveals that in the life range in which fracture finally occurs the scatter of static strength is very great, *ie* that the decline must take place very rapidly (within relatively few load cycles). Agreement between theory and test is very close.

It finally remains to be said that a decline in static strength in operation must always be anticipated when using fibre composite materials. Apart from the applied load cycles, this decline depends on other parameters (environmental conditions, laminate structure, load level, etc). All these effects must be taken into account in design, while in case of doubt the required endurance and residual strength should be proved experimentally.

7.3.3 Consideration of drop in stiffness

In addition to the drop in static strength, a drop in stiffness depending on the number of load cycles is also observed in composites. Fig 7.5 shows by the example of two unnotched GFC specimens loaded to a constant strain amplitude the measured stress and temperature sequences as a function of the number of cycles. This reveals that in both cases stiffness drops distinctly shortly before specimen failure, combined with a sharp rise in temperature. These two effects indicate the impending failure of the specimen (comparable with the drop in residual strength, cp 7.3.2).

Similar results of tests by LBF on carbon/epoxide laminates are shown in Fig 7.6. Here stiffness drops particularly sharply at the start and towards the end of the test. Individual measurements of the temperature sequence showed the same tendency as Fig 7.5. Consideration of the drop in stiffness as a function of the number of load cycles plays a decisive role particularly in the design of those components whose serviceability is impaired by excessive deformations. Furthermore the damping behaviour of a structure is largely determined by stiffness. Thus the permissible drop in stiffness for a particular structure, for instance, is specified by these boundary conditions. In order to be able to take suitable account of this in the design, it is useful to quote endurance curves not only for 'fracture' failure but also for 5% loss of stiffness, 10% loss of stiffness, etc. An example with test results from LBF is in Fig 7.7.

The designer must limit the stress level occurring in operation so that the required life is achieved while the permitted drop in stiffness is not exceeded. It must also be remembered that in operation undue deformations can occur as a consequence of creep processes which take place under constant static load and are therefore not simulated as a rule in fatigue strength tests. In certain circumstances, therefore, real time simulation may be necessary (especially if the effect of environmental cycles is to be investigated). Major deformations may also be expected in rivetted joints if under high load cycles the rivet holes are expanded or 'ovalised' due to bearing pressure.

7.4 Conclusions

The questions on design of fibre composite structures treated in the evaluated literature show that the whole problem is far from having been dealt with exhaustively. No uniform picture has emerged concerning many problems (*eg* applicability of the Miner Rule), and other problems have not yet been investigated (*eg* behaviour of composites under load in realistic temperature sequences). Many questions of means of inspection and methods to be adopted are also still open.

Since the necessary data for design procedures for fibre composite structures and the requisite material characteristics are very incomplete, the US Air Force produced a manual in 1976 ('Structural integrity roadmap'[551]) which indicates the existing gaps in knowledge and contains a framework for future research work.

7.5 References

The references evaluated in the preceding sections are classified by contents. The papers mentioned in the text are underlined.

Section 7.1

Design generally:

23, 49, 113, 140, 150, 152, 153, 157, 158, 161, 162, 228, 244, 248, 399, 401, 405, 406, 448, 491, 402, 551.

Specific applications:

44, 152, 153, 154, 318, 491, 553, 613, 614, 616.

Section 7.2

Static design:

24, 39, 83, 113, 152, 157, 419, 558.

Section 7.3.1

Damage accumulation:

50, 248, 318, 338, 518, 523, 532, 716, 785.

Section 7.3.2

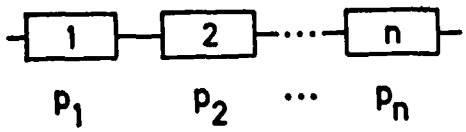
Wear-out model:

113, 162, 484.

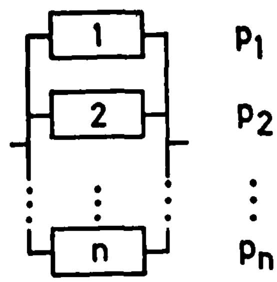
Section 7.3.3

Drop in stiffness:

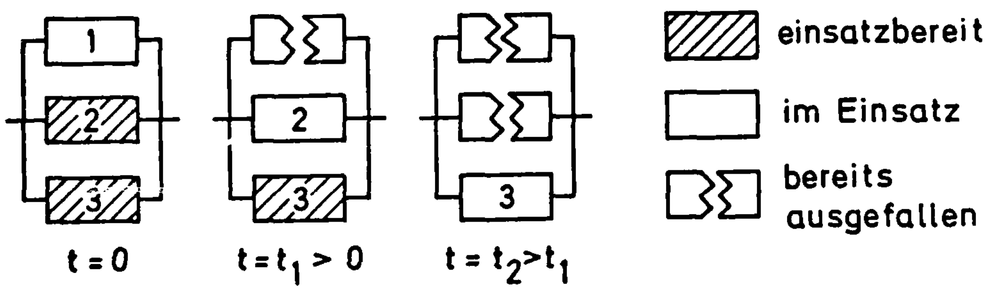
49, 245, 529.



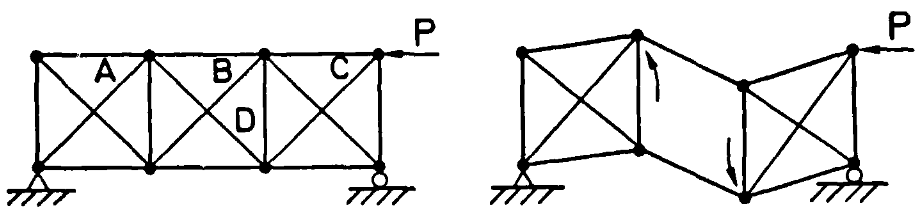
(a) Series or 'weak link' layout
(no redundancy)



(b) Parallel layout



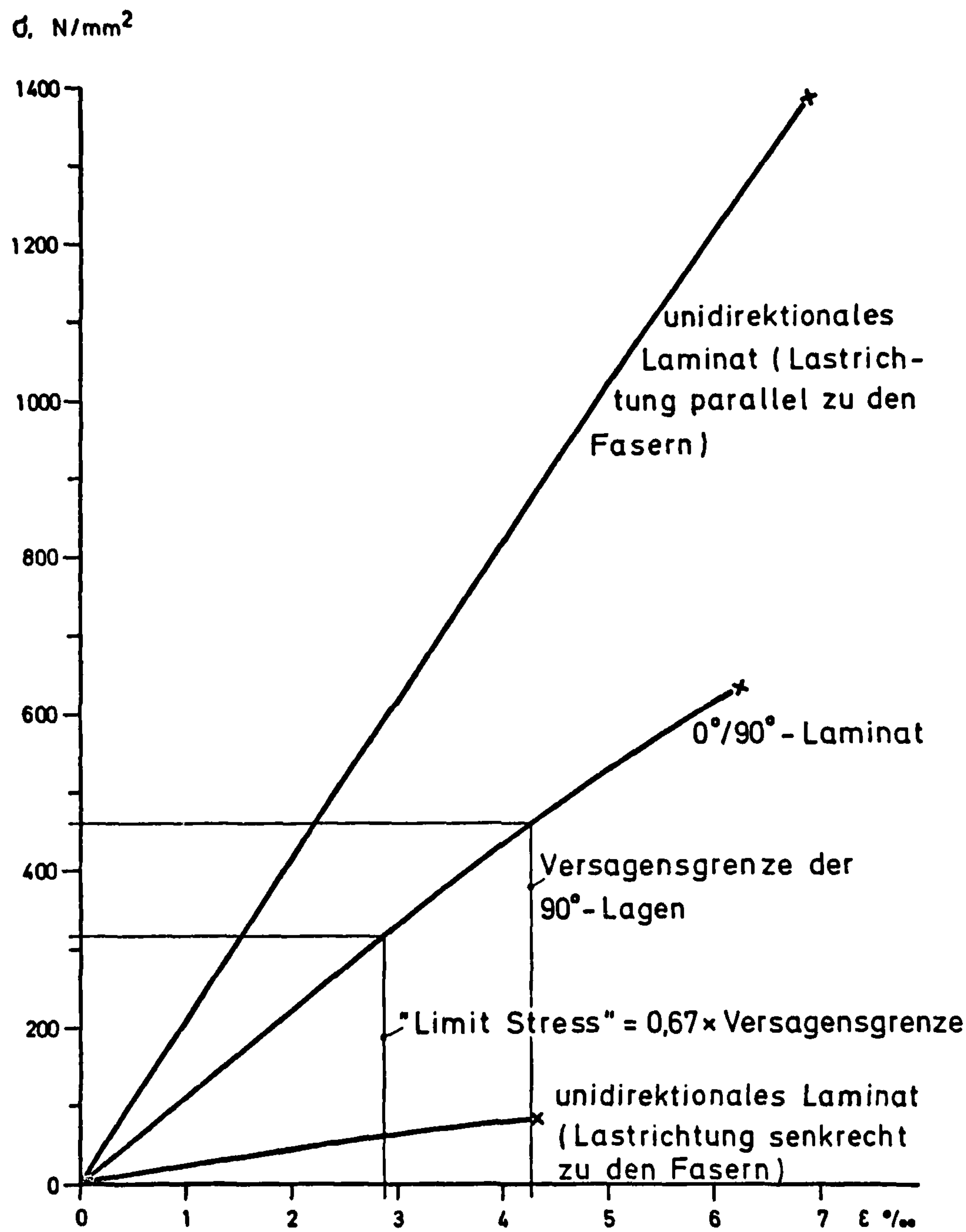
(c) Three component 'standby' layout



(d) Possible failure of a redundant structure

Key:
Einsatzbereit = ready for use
Im Einsatz = in use
Bereits ausgefallen = already failed

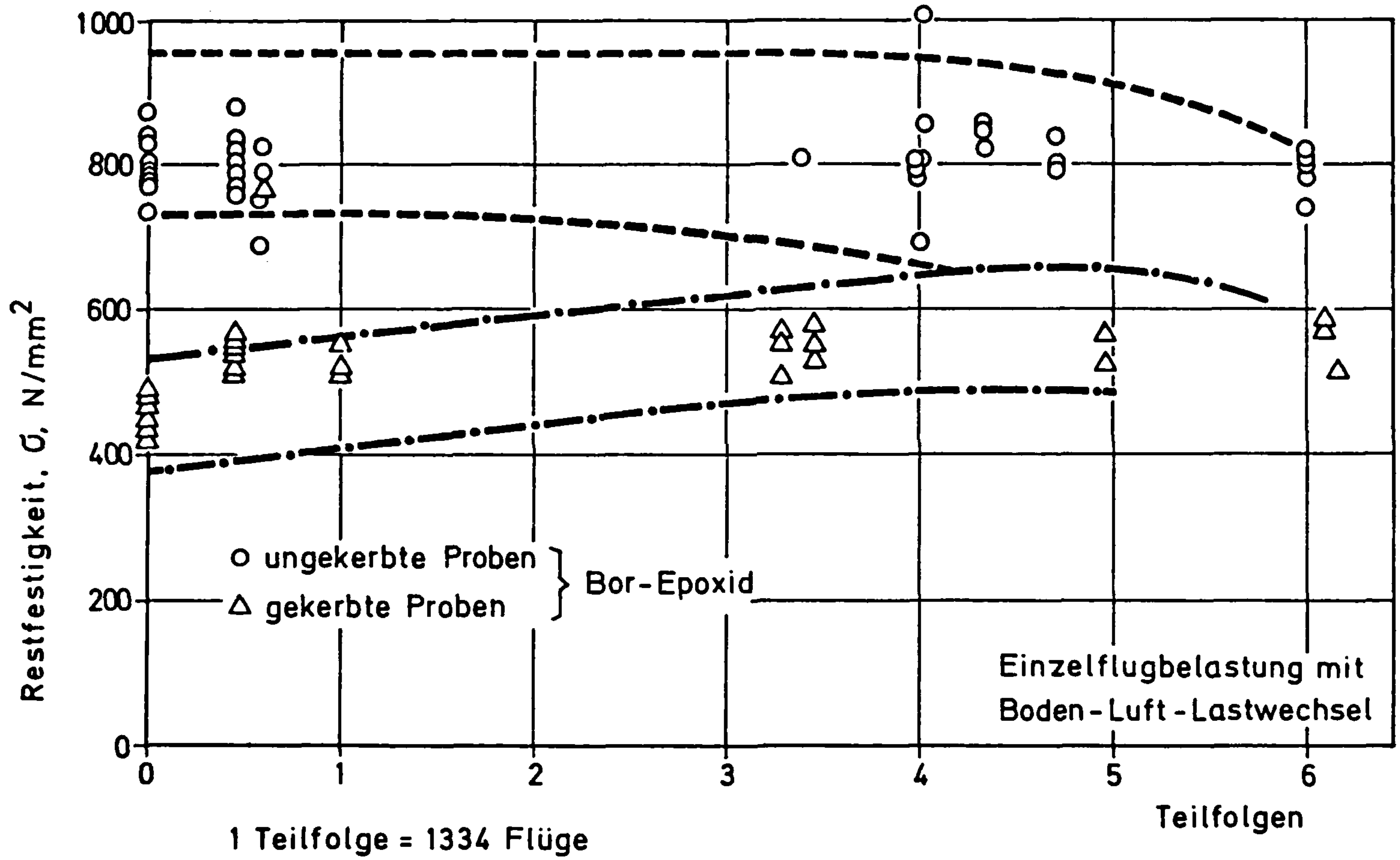
Fig 7.1 Models of redundant systems as Ref 157



Key:

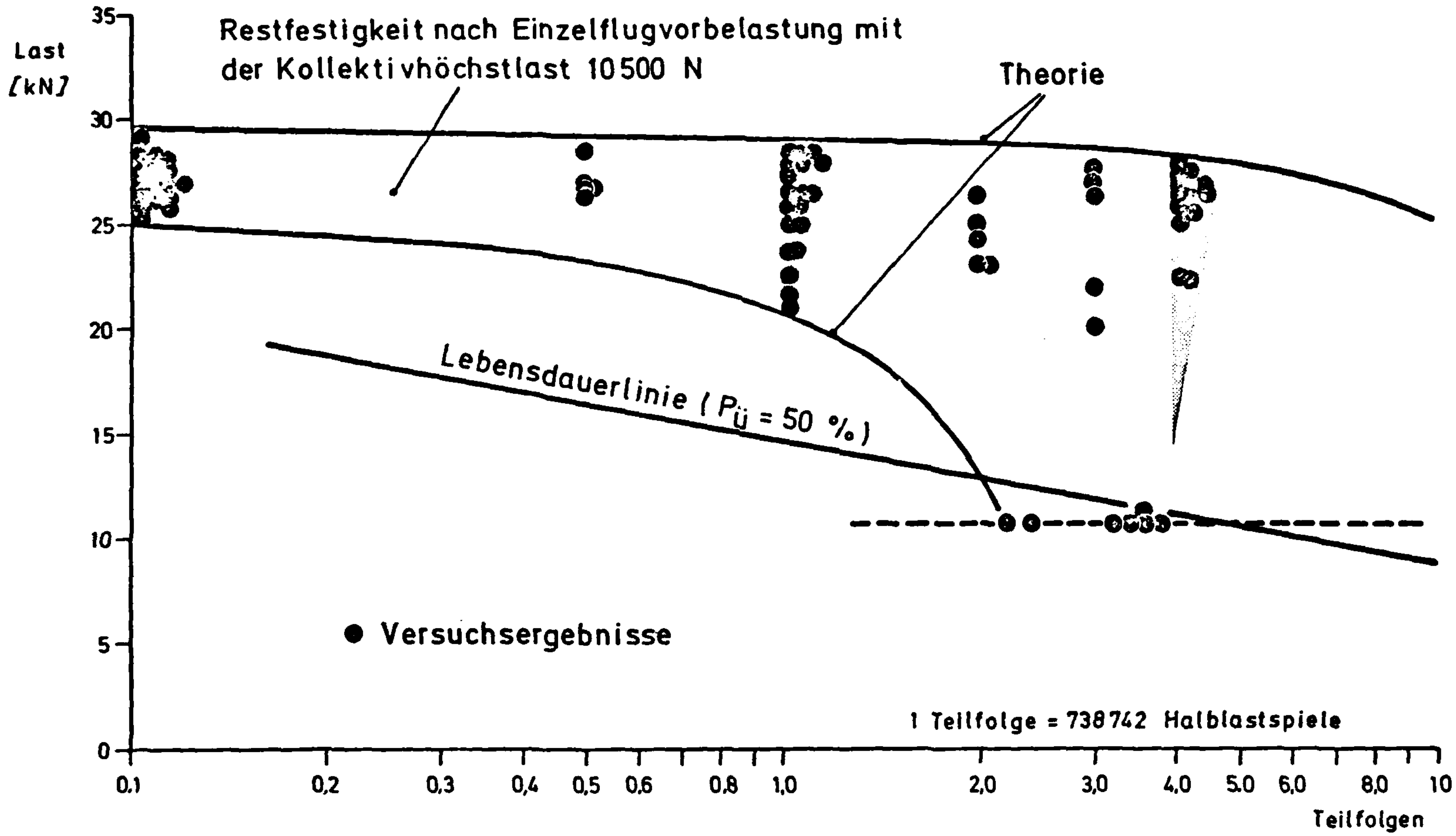
Lastrichtung parallel zu den Fasern	= direction of load parallel to fibres
Versagensgrenze	= failure limit
Lastrichtung senkrecht zu den Fasern	= direction of load transverse to fibres

Fig 7.2 Stress-strain diagram for boron/epoxide as Ref 152



Key:
Einzelflugbelastung mit Boden-Luft-Lastwechsel = flight-by-flight loading with ground-air load cycle
Teilfolgen = partial sequences

Fig 7.3 Residual strength after cyclic load as Ref 113



Key:

Restfestigkeit nach Einzelflug- vorbelastung mit der Kollektivhöchstlast = residual strength after flight-by-flight preliminary loading with an overall maximum load

Lebensdauerlinie = endurance line

Halblastspiele = half load cycles

Fig 7.4 Drop in residual strength for a bonded joint of CFC/titanium as Ref 113 (flight-by-flight tests) [redacted] [redacted] LT[redacted]5

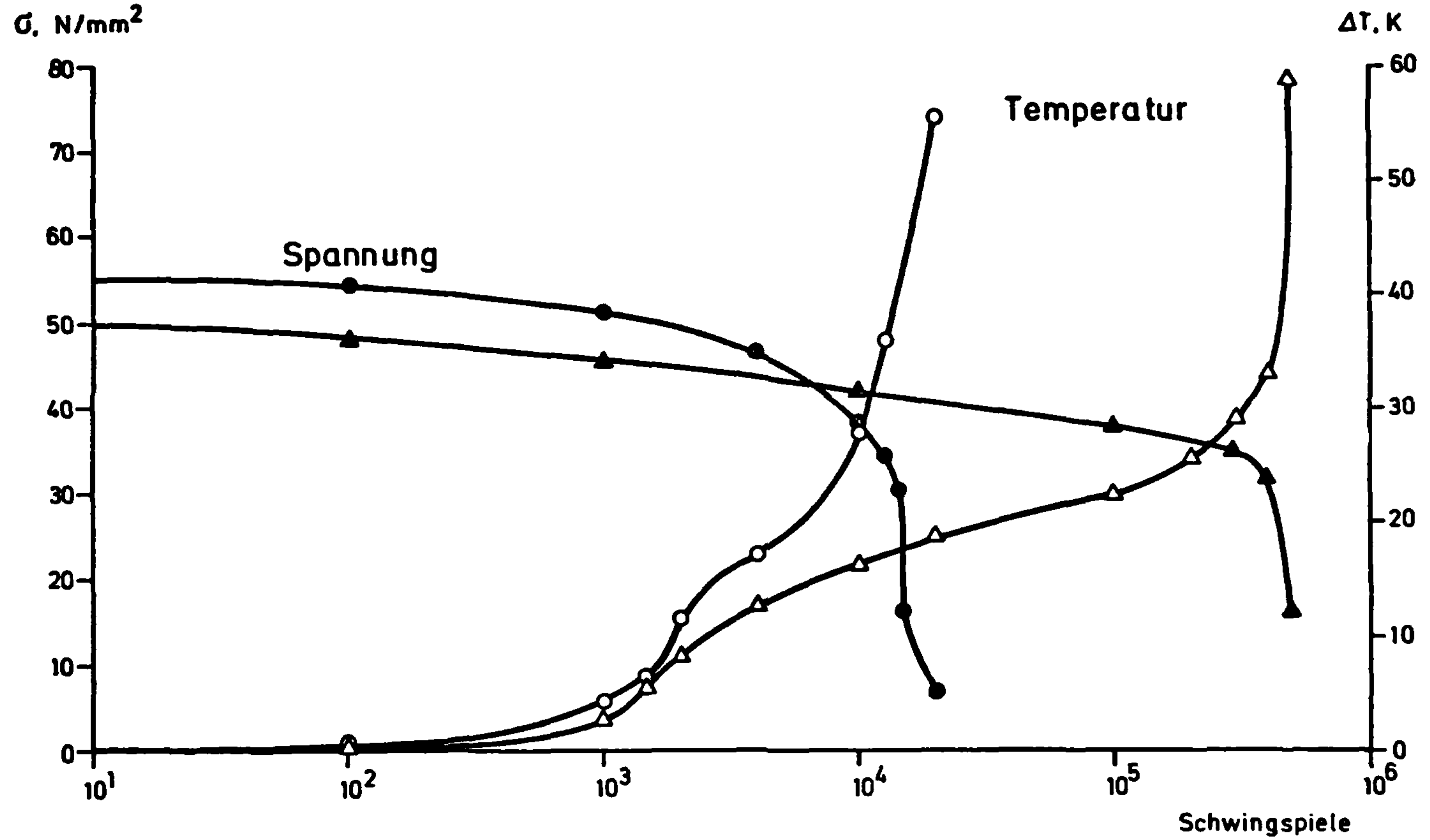
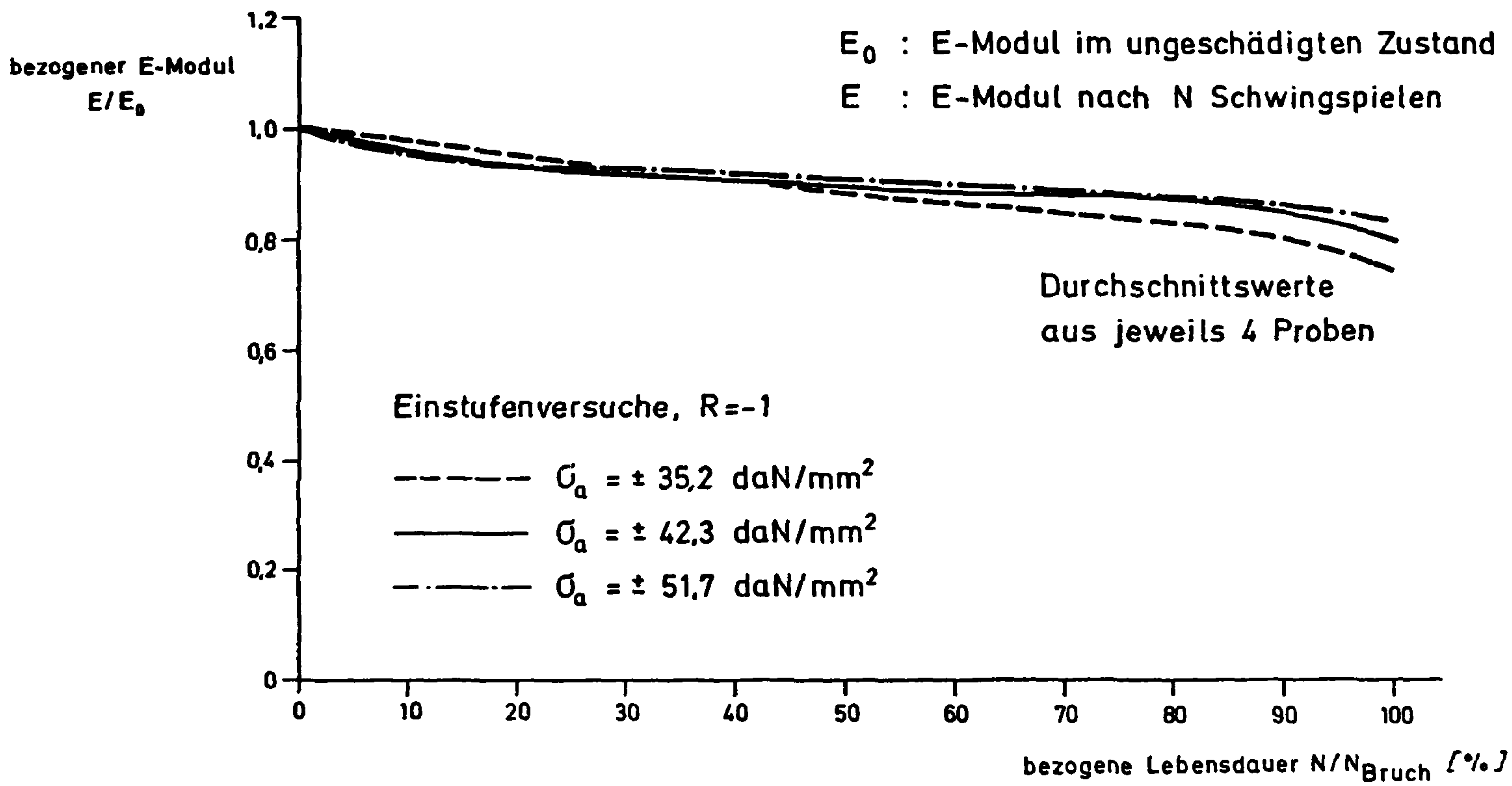


Fig 7.5 Variation of stress and temperature with number of cycles (strain controlled single-stage tests of CFC, as Ref 49



Key:
E-Modul im ungeschädigten Zustand = modulus of elasticity in undamaged condition
Nach N Schwingspielen = after N fatigue cycles
Durchschnittswerte aus jeweils 4 Proben = mean values of four specimens
Einstufenversuche = single-stage tests

Fig 7.6 Stiffness of CFC specimens as a function of number of cycles as Ref 785

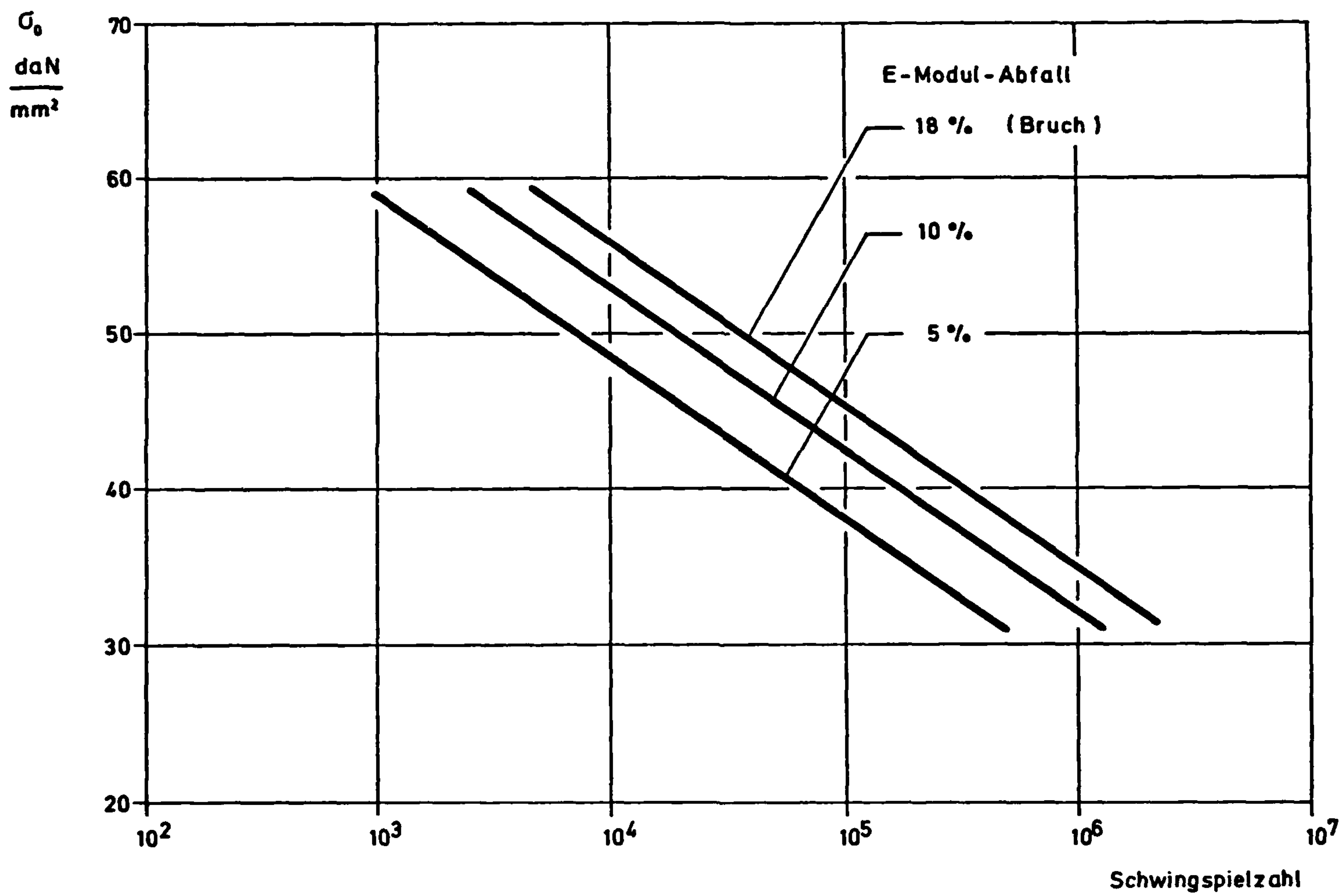


Fig 7.7 Wöhler lines for a given drop in modulus of elasticity, CFC unnotched
R = -1.0 (20 specimens)

8 INSPECTION PROCEDURES FOR FIBRE COMPOSITES

J.J. Gerharz

Contents

8.1 Introduction

Testing of materials for qualified applications, *eg* in aircraft construction is carried out firstly with the aim of ensuring production quality and, secondly, proving adequate safety during operation. To achieve both aims the methods normally used for metal materials had to be adapted to the properties of fibre composites or else new methods had to be introduced.

In general it can be said that inspections of fibre composites are more difficult and costly than metals, even though certain properties of these new materials permit inspection procedures hardly used heretofore. These are methods based on the change in stiffness, resonance and damping behaviour of components during the course of operation. In contrast to the behaviour of metal components, changes in these properties occur relatively early before catastrophic failure in fibre composites, thus assuring discovery of the damage.

- *Quality assurance* of fibre composite materials includes control of material as received and very comprehensive controls at the various production stages, which for particularly vital applications (production of primary structural components) may even include video records for the documentation of individual work stages.

  While destructive test methods are widely used for as received tests, the aim is to use non-destructive methods for controls of later production stages and the end product for economic reasons. In a few cases only, *eg* helicopter rotor blades, are components extracted from normal production at certain intervals for destructive testing. The sometimes destructive proof-load method used for some time for metal structures is also occasionally suggested for fibre composite structures[244,636,778,783,787].

- Inspection of the structure *during operation* serves as proof of adequate safety within the following interval to the next inspection. For economic reasons these intervals (inspection intervals) ought to be as long as possible and the methods as simple as possible and universally applicable. The special inspection procedures already mentioned with stiffness, resonance and damping measurements can also be used for these checks during operation.

8.2 Inspection procedures

In the following inspection procedures for fibre composite materials a distinction may be made between methods which require the component to be loaded during the test and methods which do not require loading.

The first methods include:

- stiffness measurements,
- measurement of damping properties and resonance frequency,
- temperature measurements,
- sound emission measurements,
- holographic interferometry.

The second methods include:

- ultrasonic pulse echo methods with or without C-scan recording,
- radiographic testing with soft radiation with or without contrast medium, and
- visual inspections[216,732].

Further information from the literature on the most important of the above methods are quoted briefly:

Measurements of drop in stiffness

This method is based on the change in stiffness properties produced by faults in a component, when the type of loading should be matched to the type of damage and the laminate structure in order to measure the most distant effects. In angle ply composites, for instance, axial stiffness drops most distinctly and in uni-directional composites torsional rigidity[202]. Careful attention must also be paid in these measurements to the effect of heating. Nevadunsky and co-workers[216] have, for instance, proved that the drop in stiffness is greater in a specimen heated during the test than that of the cold specimen.

Measurements of damping and resonant frequency

It was demonstrated in Refs 90 and 166 that changes in damping and resonant frequency indicate the presence of fatigue damage. The excitation of the component to high-frequency vibration necessary for this measurement is a drawback.

The possible heating up entailed may also cause changes in damping and resonant frequency[280].

Sound emission measurements

The measurement of sound emission appears to be a good method of indicating fibre breaks. Many applications under monotonic loading show pronounced sound emissions on the occurrence of individual faults[196,226,505,506,630,730,783].

According to investigations by Williams and Reifsnider on B/Al composite there is also close agreement between extent of damage and total number of emissions on vibration loading. They also find that delaminations generate disturbing shear and friction noise which masks the emissions from damage growth. An effect of the type of deformations in the laminate is found by the authors of Ref 630 for B/Al: according to this the number of sound impulses occurring per unit of time is greatest when the predominant deformations are in the direction of the fibres (tension), less when the deformation is primarily in the transverse direction (tension) and least on intralaminar shear deformation.

The results of all tests show relatively wide scatter[168].

Holographic interferometry

Tests with this technique have shown that irregularities such as local stress concentrations, weak zones and bond faults can be discovered easily with laser holography. Two methods have been used successfully[216], first the double exposure method with exposure with laser pulses of very short duration (30-50 μs) in the unloaded and loaded state, and secondly a method in which the specimen vibrates at high frequency (60 kHz), being exposed to a pulsating laser beam for approximately 300 ms. The high technical cost is a drawback of both methods.

The suitability of holographic interferometry as a non-destructive method for fibre composite components has been investigated particularly by the German aircraft industry among other methods (ultrasonics, sound emission, etc), see Ref 791, pp 111-116.

Of the non-destructive methods not requiring loading of the test object the furthest developed for fibre composite structures are radiographic and ultrasonic methods.

Radiographic methods

Radiographic methods have been used for damage analysis with and without contrast media. In composites with metal fibres radiation is very effective in detecting fibre breaks[104,226,717]. With contrast medium[162,167,170,517,730,732] delaminations, matrix cracks and fibre/matrix bond cracks (debonding) are particularly easily visible in CFC if the medium is able to penetrate these faults, *ie* if the cracks break the surfaces (see section 4).

Ultrasonic measurements

In *ultrasonics* a distinction must be made between two methods:

- the simpler most frequently used method with pulse echo and C-scan recording of the measured variables, and
- ultrasonic spectroscopy.

In the first method a weakening of the echo signal and in the second differences in the energy distribution over the frequency range of the echo are ascribed to damage or faults in the composite. In principle, only faults which lie normally to the direction of the acoustic irradiation can be detected by the ultrasonic method. Many applications of the C-scan method have shown the reliability of this test method in discovering delaminations and large voids in CFC and BFC[127,172,216,280,330,517]. Its use for fibre composite-metal bonds is problematical because of edge effects, which require rapid adjustment of receiver sensitivity in order to detect faults in the critical transitional areas with adequate reliability.

On comparing radiographic and ultrasonic methods it can be said that (see also section 4):

- ultrasonic C-scan is the simpler method,
- radiographic methods supply more detailed information.

Both methods are used predominantly in the aircraft industry nowadays, while ultrasonic sensors are used for components dipped in water or with water jet contact. The radiographic method was widely used by Messrs Grumman for the B-1 horizontal tail surfaces.

Visual inspections

Visual inspection is the most frequent and easiest method. The faults which can be detected by this means are resin 'crazing' (GFC), cracks and delaminations[732]. Production faults which can be discovered and which can initiate fatigue damage are scratches, indented areas, separation, porous areas, voids and delaminations. A proviso is, of course, that all the faults mentioned are visible on the surface, which is not always the case for voids and delaminations. Delaminations generated by cyclic loading, however, almost always emanate from the edges, as mentioned in section 4.

8.3 References

In this section the evaluated references are classified by content under the subjects covered in the preceding sections. The paper mentioned in the text are underlined.

Section 8.1

Introduction:

_244_, _636_, _778_, _783_, _787_.

Section 8.2

Methods of inspection:

General:

25, 49, 153, 154, 156, 167, 182, 202, _216_, 280, 330, 375, 517, 613, 614, _732_.

Measurements of drop in stiffness:

49, 98, 141, 144, 153, 158, 184, _202_, 203, 205, _216_, 218, 224, 227, 234, 245, 254, 280, 315, 316, 317, 338, 390, 474, 500, 529, 530, 531, 535, 625, 717, 729, 732.

Measurements of damping and resonant frequency:

_90_, _166_, _280_.

Measurement of heating up:

49, 138, 153, 166, 202, 206, 213, 216, 219, 220, 225, 240, 241, 254, 258, 280, 315, 324, 391, 531, 537, 787.

Sound emission measurements:

_168_, _196_, 218, 220, _226_, 244, _505_, _506_, _630_, _730_, _783_.

Holographic interferometry:

_104_, _162_, 164, _167_, _170_, 202, _216_, _226_, 422, 497, _517_, _717_, _730_, _732_, _791_.

Radiographic methods:

25, _104_, _162_, _167_, _170_, 216, _226_, 448, 450, _517_, 523, _717_, _730_, _732_.

Ultrasonic measurements:

25, _127_, 170, _172_, _216_, _280_, _330_, _517_, 737.

Visual inspection:

_732_.

9 TEST TECHNIQUES AND METHODS

J.J. Gerharz

Contents

9.1 Introduction

When new construction materials, which generally also require or permit new methods of construction, are introduced, extensive investigations on three levels are necessary before they can be put to practical use:

(a) Investigations during the development phase of the material, starting with its manufacture in the laboratory up to production in quantity.

(b) Tests on the commercially produced material to establish its behaviour under the conditions of potential fields of application.

(c) Tests on simple specimens and structural elements to produce design data.

In the development of fibre composites investigations on level (a) can be regarded as concluded. Only points (b) and (c) will therefore be dealt with below. The necessary tests can be roughly divided into:

- tests which provide the properties of fibre and matrix for use in determining the properties of the individual layer,
- tests which supply the properties of the individual layer for determining the optimum multi-layer composite,
- tests on the multi-layer composite to determine its behaviour in general operational conditions, and
- tests on component and structure under special operating conditions.

In order to ensure the reprodu[illegible] the test results obtained from specimens from different production [illegible] exclusion of batch effects, for instance, additional control tests must [illegible] carried out to check the individual layers, the composites and the components in regard to quality and scatter.

A large number of guidelines have been created for the various test techniques and methods which are referred to in the following sections. In order to reduce the amount of text, abbreviations only are used; the complete titles of the guidelines cited are contained in Fig 9.1.

9.2 Tests to determine characteristic values for fibre and matrix materials

Tests produce primarily mechanical and thermal characteristics from which the data for the uni-directional individual layer can be calculated (see section 6). In fibre investigations tests with individual fibres of different lengths contrast with tests with impregnated fibre strands[229,502] whose results compare better with those of the multi-directional composite. Measurements of the mechanical characteristics of fibres are contained in Refs 5, 39, 53, 130, 173, 193, 196, 229, 274, 425 and 495. For the plastics used as matrix materials in addition to the mechanical and thermal characteristics, fusion point, flow curves and viscosity measurements must be carried out by the usual plastics test methods (see guidelines R1, R2, R3 in Fig 9.1) in order to establish rheological characteristics.

9.3 Tests to determine characteristic values for individual layers

To determine the composite most suited for a particular component the properties of the orthotropic individual layer are of great importance. The minimum necessary characteristics of a uni-directional composite with a particular fibre content are summarised in Fig 9.2[502]. Further, fibre content (R4), proportion of voids (R5) and specific weight (R6, R7) should be ascertained to ensure comparability of results. Type of specimen, and the set-up and performance of tensile tests are described in detail in (R8).

The following stipulations deviating from the standards have also proved successful:

- GFC reinforcement of specimen ends in place of aluminium reinforcement, and
- laying down the length of the reinforcement according to fibre strength and specimen thickness in order to avoid failure of the bond.
- RAE flat specimen with constriction as Refs 536, 539 and 540.
- Annular specimen (advantage: fitted without clamping force) as Refs 12, 33, 134, 207, 217, 315, 323, 400, 497 and 722, and
- 90° tube specimen as Refs 125, 376 and 418 for determination of the mechanical properties in transverse direction.

In the RAE specimen the contour of the transition from the test area to the clamped ends (see sketch) is determined so that the intralaminar shear strength is not exceeded anywhere until tensile strength has been reached in the smallest cross-section. This condition produces an equation for specimen width in the transition from the parallel test section to the clamped ends:

$$\frac{y}{y_0} < \exp\left(\frac{\tau_B}{\sigma_{B_z}} \frac{x}{y_0}\right)$$

where y_0 = width in test section

y = width in transition region

x = distance from test section.

The usual annular specimens are the so-called NOL ring and the elongated ring which is flattened in places to take strain measurement devices. In the elongated ring, however, the method of production raises the problem of uneven proportions of fibres and voids between rounded and flattened parts of the specimen[502]. When 90° tube specimens are used special attention must be paid to adequate wall thickness and good alignment of the longitudinal axis of the specimen with the gripping device of the testing machine. Results from Ref 119, for instance, show a pronounced effect of the geometry of the specimen. Tensile strength and stress-strain behaviour were established with ASTM flat specimens, flat bending specimens (as used for acceptance tests) and honeycomb sandwich bending specimens. Comparison of the results of the tests performed on a large number of random specimens shows inter alia:

- the mean fracture stresses are greater in the flat bending specimens and sandwich bending specimens than in the flat tensile specimens. The coefficient of variation of the fracture stresses (ratio of standard deviation and mean value) as a measure of the scatter of the test results do not differ in the flat specimens; they are smaller than in the sandwich specimens,
- the maximum rise in the stress-strain curve (tangent from $\sigma = \varepsilon = 0$) as a measure of stiffness is greater in the sandwich than in the flat specimens.

Since there is as yet no standard for compression strength tests, many different test methods are used. Although there were no stability problems with thick specimens with a large number of layers and square or circular cross-section, their free ends split up ('brooming'[18,529,539]) since clamping holders which would have prevented splitting could not be used because of the high compression loads. Sandwich specimens were also used for axial and bending loads[54,211,267,404] as well as tube specimens[211,502]. For determination of characteristics under pure compressive loading of uni-directional fibre composites flat specimens with very short free length (*eg* so-called Celaneze specimen[25,218,516,539,540] with and without buckling restraints are being used with increasing frequency. Specimen length and thickness must be matched to the modulus of elasticity of the fibres. Like the tensile specimen, the compressive specimen developed by the RAE is constricted with a short gauge length[536,539,540].

Because of the particularly brittle behaviour of uni-directional fibre composites, bending stresses due to mis-aligned clamping have a pronounced effect on the result. A test method in which undesirable bending stresses do not occur is therefore essential to reliable test results.

There are numerous known test methods for the determination of the modulus of shear and shear strength, for example,

Tests with intralaminar shear stress (within the layers) which provide both shear strength and the shear stress-shear strain curve:

- torsion test with wrapped thin-walled 90° tube[12,793],
- torsion test with 0° round bar[12,40,723],
- tensile test with $\pm45^\circ$ composite flat specimen[8,516,723,724,725],
- tensile test with 10° composite flat specimen[218,723,725,726],
- 'rail-shear' test with symmetrical or asymmetrical introduction of force, 0° or 0°, 90° composite[118,214,540].

Tests with interlaminar shear stress (between layers) which only provide shear strength:

- tensile test with 0° flat specimen with opposing staggered transverse slots[172],
- bending test with short bending specimen of 0° composite[5,516,536].

The test methods for exclusive determination of interlaminar shear strength are standardised as (R9, R10). Various test methods are compared in Refs 723 and 725 and tensile tests with $\pm45^\circ$ or 10° composite flat specimens are recommended for determination of shear modulus and shear strength. To produce a $\tau - \gamma$ curve the 10° composite specimens require strain measurements with a rosette or knowledge of E_{11}, E_{22} and ν_{12} (see Fig 9.2)[726]. For the $\pm45^\circ$ composite specimen strain measurements in longitudinal and transverse directions (0° and 90°) are necessary[724]. While the 10° composite specimen produces the more reliable values[725], tests with $\pm45^\circ$ composite specimens are easier to carry out[723]. The 10^o composite specimens are more sensitive during processing and fixing in the testing machine and require reinforcement at the ends[726]. In contrast, $\pm45^o$ composite specimens do not require reinforcement[723]. (R8) can serve as a guideline in carrying out tests on both types of specimen[541].

Fig 9.3 sets out the equations for calculation of shear stress and slip for intralaminar shear.

Various methods of determining the characteristic data of temperature-dependent properties are quoted in the literature[48,146,174,397,404,514,516].

9.4 Tests of angle ply composites, components and structures

According to section 6, the elastic behaviour of angle ply composites can be predicted with good approximation from the characteristics of the unidirectional layer. This also applies to the characteristics of the thermal properties. Prediction of failure with the aid of a fracture hypothesis, however, is imprecise so that for design purposes the strength of any angle ply composite must be determined experimentally. The deformations can be measured at the same time with little extra effort.

Multi-layer and angle ply composites, typical components and structures are tested under general operational conditions such as:

- long-term load at different temperatures to determine creep strength and creep behaviour,
- cyclic load with constant and variable amplitude, and
- temperature and moisture.

With some exceptions the type of specimen and test methods for unidirectional composites can be used for angle ply composites. In compressive and tensile tests the type of specimen and clamping system (R8) can be the same if the specimens are restrained against buckling in the compressive test. For examples see Refs 225, 515, 517, 532, 533 and 631. Contrary to the ASTM standard, angle ply composite specimens with plain clamping plates do not require reinforcement at the clamped ends if force is introduced evenly.

Creep strain and creep strength tests on CFC under axial tensile load have been carried out by the RAE[558,666] on the lines of (R11):

- 0^o specimen : constriction over thickness,
- 90^o specimen : constriction over width, and
- 0^o, $\pm 45^o$ and 90^o, $\pm 45^o$ specimens : no constriction.

For these CFC specimens load was introduced through aluminium plates stuck to the ends of the composite on both sides so that load transfers to the specimen exclusively by thrust and thus weak areas due to fittings are avoided. The creep strains were measured over a length of 50 mm with the aid of mechanical extensometers with revolving mirror and a sensitivity of 0.002%. Adhesive aluminium strips at the contact points prevented the CFC specimens being damaged by the tips of the extensometers. In an investigation by Messrs Lockheed[21] the creep strains in 0^o, 90^o CFC were, however, measured with conventional strain gauges parallel and transverse to the load direction. Several thermocouples[21,666] were fitted to each specimen to control the temperature in the test. For creep strain tests in liquid medium (R12) of 0^o specimens of CFC[173] a special clamping system combined with the medium container was used; load transfer to the composite specimens taking place in the same way as in the RAE tests. A creep test set-up is described in Refs 87, 720 and 776 in which 400 fibre strand epoxide specimens approximately 300 mm in length are loaded by hanging weights for a period of up to 10 years. The ends of these specimens were clamped between metal bits and glued. With this combination of clamps and glueing difficulties were avoided which had arisen in very long test periods with specimens which were only clamped. Ref 720 also reports on new equipment for 800 specimens and a large environmental chamber in which temperatures between 10 and 50^oC and relative humidity between 10 and 90% can be achieved. Operation is also mentioned of test equipment for more than 280 pressure tanks under continuous biaxial load. Tests of bearing strength are reported in Refs 44, 222, 545, 546 and 548. The test technique used is standardised in (R13). Component and structure tests with and without environmental simulation are described in the relevant literature, see section 9.8 and section 11.

9.5 Environmental conditions for aircraft

The environmental conditions for an aircraft are largely determined by where it is stationed and its performance: the temperatures occurring vary with its technical range of duties, moisture absorption varies with its geographical siting. Extreme changes in environmental conditions arise for aircraft which fly at low altitude in the supersonic range and at high altitudes in the subsonic range, and are stationed in a warm humid climate.

Moisture absorption by the resin takes place primarily when the aircraft is on the ground; the effect of high air humidities sometimes of 90% and solar radiation combine; for instance, according to measurements in Ref 553 a wing upper surface can be heated to 77°C by solar radiation in outside temperatures of 40°C in calm weather. These conditions are relatively frequent in tropical zones. Subsonic flight at levels with -57°C outside temperature (standard atmosphere INA) and aerodynamic heating during supersonic flight at low levels, for instance, generate temperature changes of -45°C to +150°C on the wing outer skin of a modern fighter aircraft, according to Ref 149. These observations were widely taken into account in simulation of the environment in investigations[149,322,369,528,613].

9.6 Environmental conditions in tests

A number of test methods to determine environmental influences are described in the literature which are more or less tailored to the practical use of components of fibre composite materials. In order to cater particularly for the requirements of aircraft and space travel as the most important areas for the use of fibre composite materials, the following environmental conditions must be simulated in the main:

- extremely low temperatures (typical for rocket liquid gas pressure tanks),
- extreme temperature changes (typical for satellites), and
- temperature changes with moisture absorption and simultaneous mechanical load (typical for aircraft).

The instantaneous environmental conditions to be expected in operation (temperature, stress, medium) can be simulated relatively well in tests; but not generally the realistic time sequence of changes. So-called real time simulation may be successful for purely temperature changes.

The operational period of rocket pressure tanks is generally short. Bursting strength tests in liquid gas[33,497], static strength tests with specimens dipped in liquid gas[195,196,323,383,529] and tests on a cylinder of CFC filled with liquid gas with alternating internal pressure[323] therefore reproduce the operating stresses relatively well. For largely realistic simulation of temperature changes occurring in space the fibre composite specimens are heated with the aid of boiling water or cooled with liquid nitrogen, without allowing the specimens direct contact with these media[174,514].

In so-called real time tests the temperature time slope is simulated as realistically as possible, while the mechanical operational loads generally follow a time lapse sequence. Mechanical loading is interrupted for the duration of each temperature change and for moisture absorption. Tests of this kind concern:

- exposure of angle ply composites of typical structure to various climatic zones of terrestial atmosphere as in the NASA programme[332,667] and in the programme of the Naval Air Development Centre of the USA[515]. The sites are airports or bases. The specimens are unloaded or else subjected to constant tension by a spring mechanism,

- secondary components installed in the aircraft as in the NASA programme[332], especially the 114 flight spoilers of CFC installed in Boeing 737 aircraft[727],

- structural components[551], *eg* B1 and F16 horizontal stabilisers and elevators. Test periods : 1 hour per simulated flight, but still about 6 years before the flight hours required as evidence of fatigue life are reached,

- notched specimens with loading and temperature sequence for supersonic flight (Mach 3) without taking account of moisture absorption and ground load.
  Test periods : 2 hours per simulated flight[668,332], but minimum 6 years until the 50000 flight hours required as evidence of fatigue life are achieved.

Apart from these real time investigations a number of simplified methods have been used to detect environmental effects. These include exposure of unloaded specimens:

- at alternating temperatures without moisture effect:
  $-73^{o}C$ to 80% of the curing temperature[218], 38^{o} to $+127^{o}C$ and $38-177^{o}C$[516] and $-54^{o}C$ to $+149^{o}C$[647]. Some of the specimens were under constant load during the temperature changes[218],

- at alternating temperatures with moisture effect:
  $-55^{o}C$ to $120^{o}C$ ($150^{o}C$), 95-100% relative humidity at $+50^{o}C$[92,369],

- with moisture effect and interspersed temperature changes to simulate supersonic and vertical diving manoeuvres, *eg*

  RT $\rightarrow$ +127^{o}C $\rightarrow$ RT[214],

  +49^{o}C $\rightarrow$ −54^{o}C $\rightarrow$ +127^{o}C $\rightarrow$ +49^{o}C[516], and

  +82^{o}C $\rightarrow$ +149^{o}C $\rightarrow$ +82^{o}C[649],

- at constant high or low temperature[316,468,516,667] in environmental chambers, when exposure took place at reduced pressure to simulate environmental conditions for flights at high altitude in Ref 667,

- artificial weathering with the aid of lamps and sprays (R14, R15) to simulate sunlight and rain[516],

- in water or water vapour at 100^{o}C[92,172,214,268,271,309,340,356,387,667].

- in water at room temperature[144,309,684],

- in moist air at 40-70^{o}C and 95-98% relative humidity[309,516,530,538,553,613,618,648,667]; in Refs 516 and 530 a method as (R16) was used[600].

The moisture absorbed by fibre composite specimens is found by weighing the specimens in dry and moist condition[172,322,573,718,722]. After different periods of exposure to the moist medium, short- or long-term tests are carried out[144,684] in order to determine the effect of moisture on the mechanical properties of the fibre composite specimens. Short-term tests on non-pretreated specimens at realistic high and low temperatures are reported in Refs 161, 214, 320,396,467,469,494,516,613,648,649 and 667. Experimental systems are described in Refs 161, 214, 320, 396 and 648. In long-term tests under cyclic and static load and constant environmental conditions[144,316,320,455,517,684] liquid nitrogen[316], quartz lamps[316,320], heating coils[144] and climatic furnaces[316] are used to regulate temperature and water[144,684] for moisture absorption.

Flight-by-flight tests on a stepped BFC titanium adhesive bond (to simulate a wing skin-fuselage joint) with temperature change matched to the load sequence, without simulation of the humid environment are reported in Ref 149. As an example of this Fig 9.4 shows the load sequence with superimposed temperature change between points RT, −54^{o}, +130^{o}C. The load sequence was matched to the temperature changes in the test by occasional frequency changes and pauses. The loading frequency was approximately 0.2 Hz on average. Heating of the specimens was by hot air at 50^{o} per minute, cooling at 33^{o} per minute by air cooled by liquid nitrogen.

A spar specimen (sine wave spar) from the B-1 horizontal tail surfaces was tested in Ref 613 with combined environmental and load programme. The load cycles of 100 flights were combined in one environmental cycle, see Fig 9.5. Before the start of the tests each specimen had been brought to a realistic moisture content of 1.35% (estimated value for a base in a humid climate) by exposure to 77°C and 90% relative humidity. This moisture content was held constant in the test by feeding with air at 95% relative humidity in a temperature range between 49 and 77°C. The test temperature changed by 4°C per minute.

Moisture diffusion rates in fibre composite materials with a plastic matrix can be determined experimentally. The evaluation of various methods is described in detail in Refs 572 and 573.

9.7 Measurement of thermal expansion

To measure the thermal expansion in the composite after curing a method has been developed in (R17) of embedding strain gauges (DMS) and thermocouples in individual layers of the composite. Since the electrical connections must be made at the side, that is between the layers, these specimens cannot be cut from ready cured plates and costly special manufacture of individual specimen bars is unavoidable. Devices for laying-up and curing the specimens with embedded strain gauges are described in Refs 55 and 218. Strain gauge rosettes and cables are used which cause only minor local thickening and are insulated against electrically conducting fibres (boron, carbon). The support and jacket are only half the thickness of standard strain gauges so that total thickness is approximately 0.025 mm. Since thermocouples are considerably thicker these are embedded in the ends of the specimens. For specimens with epoxide matrix strain gauges from Messrs Micro Measurements (MM), series QA (*eg* 3-gauge rosette QA-06-125 RD-350 option B 171) are used, and for specimens with polyamide matrix strain gauges from MM of series WK (*eg* 3-gauge rosette WK-06-125 RA-350 option B 156). Further details of the wiring and measuring system are described in Ref 218.

9.8 References

The references evaluated in the preceding sections are classified according to contents. The papers quoted in each section are underlined.

Section 9.2

Tests to determine the characteristics of fibre and matrix materials: 5, 39, 53, 130, 173, 193, 196, 229, 274, 425, 495, 502.

Section 9.3

Tests to determine the characteristics of individual layers:

5, 8, 12, 18, 25, 33, 40, 48, 54, 118, 119, 125, 134, 146, 172, 174, 193, 194, 207, 211, 214, 217, 218, 267, 315, 323, 376, 397, 400, 404, 418, 497, 502, 514, 516, 529, 536, 539, 540, 541, 722, 723, 724, 725, 726, 793.

Section 9.4

Tests on angle ply composites, components and structures:

21, 44, 87, 173, 174, 222, 225, 515, 517, 532, 533, 545, 546, 548, 558, 631, 666, 720, 776.

Section 9.5

Environmental conditions for aircraft:

149, 322, 369, 528, 553, 613.

Section 9.6

Environmental conditions in tests:

33, 92, 144, 149, 161, 172, 174, 195, 196, 214, 218, 268, 271, 309, 316, 320, 322, 323, 332, 340, 356, 369, 383, 387, 396, 455, 467, 468, 469, 494, 497, 514, 515, 516, 517, 529, 530, 551, 572, 573, 600, 613, 647, 648, 649, 667, 668, 684, 718, 722, 727.

Section 9.7

Measurement of thermal expansion:

55, 218.

Fig 9.1 (Table 1)

List of guidelines for test techniques and methods

Kurz-zeichen	Title	Bemerkungen
R1	DIN 51 550	Rheological characteristics for plastics
R2	ASTM D-696	
R3	ASTM D-648	
R4	ASTM D-3171	Fibre content
R5	ASTM D-2734	Void content
R6	ASTM D-792	Specific weights
R7	ASTM D-1505	
R8	ASTM D-3039-74	Tensile tests
R9	ASTM D-2733-74	Interlaminar shear tests
R10	ASTM D-2344-72	
R11	British Standard, BS 3500, part 3	Creep strain and creep strength tests
R12	ASTM D-638	
R13	ASTM D-953-54	Bearing tests
R14	ASTM D-1499-64	Artificial weathering
R15	ASTM G-23-69	
R16	MIL-Handbook 17	Storage with moisture absorption
R17	IITRI	Thermal expansion (residual stresses)
R18*	AIAA Journal, Vol.2, No.12	Test methods for fibres and plastic matrix
R19*	ASTM D-2343-67	
R20*	Structural Design Guide for Advanced Composite Applications	
R21*	ASTM D-C.337-57	Thermal expansion coefficient
R22*	ASTM D-C.177-3	Thermal conductivity coefficient

* Not quoted in this section

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Fig 9.1

Fig 9.2 (Table 2)
Characteristics of a uni-directional composite

Mechanical characteristics

	Tension	Compression	Shear
Elastic constants			
1. Modulus of elasticity, longitudinal*	E_{11}	$\bar{E}_{11}$	
2. Modulus of elasticity, transverse*	E_{22}	$\bar{E}_{22}$	
3. Greater transverse strain coefficient	ν_{12}	$\bar{\nu}_{12}$	
4. Lesser transverse strain coefficient	ν_{21}	$\bar{\nu}_{21}$	
5. Modulus of shear in laminate plane			G_{12}
Strengths			
1. Longitudinal*	σ_1	$\bar{\sigma}_1$	
2. Transverse*	σ_2	$\bar{\sigma}_2$	
3. Shear in laminate plane			σ_6

Physical characteristics

	Shear
Thermal coefficients	
1. Thermal expansion coefficient for longitudinal direction*	α_1
2. Thermal expansion coefficient for transverse direction*	α_2
3. Thermal conductivity coefficient for longitudinal direction	χ_1
4. Thermal conductivity coefficient for transverse direction	χ_2
5. Thermal conductivity coefficient for direction of thickness	χ_3
6. Specific heat	c_p

* Longitudinal, in longitudinal direction = parallel to fibre
Transverse, in transverse direction = normal to fibre

Fig 9.2

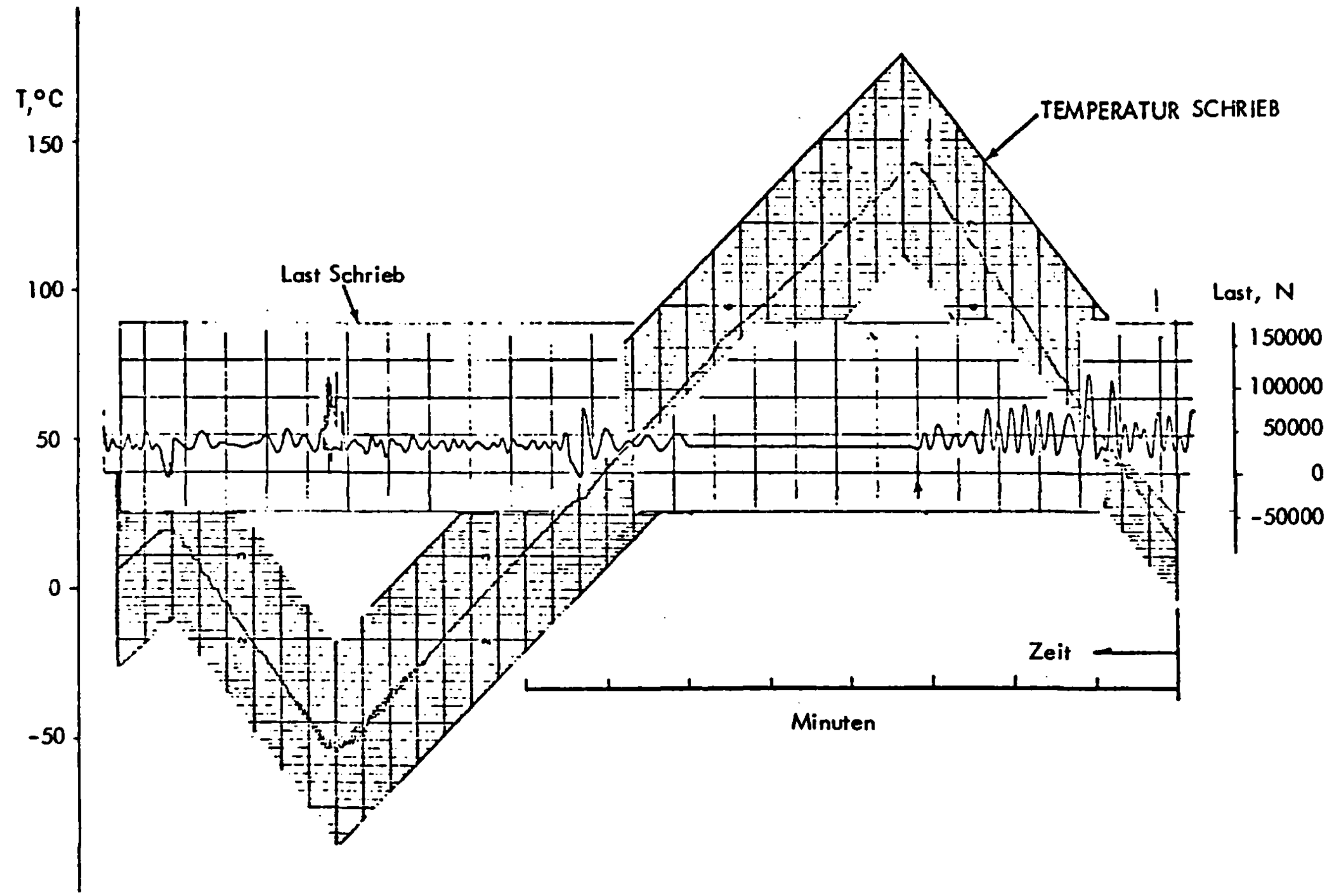
Fig 9.3 (Table 3)

Shear stress and slip under intralaminar shear

Specimen	τ_{12}	γ_{12}
90^o-tube	$\frac{T}{2r_m^2 t}$ with $r_m = (r_i + r_a)/2$	$\varepsilon_{+45^o} - \varepsilon_{-45^o}$
0^o-round bar	$\frac{T}{\pi r^3}$	$\frac{\theta r}{\ell}$
$\pm 45^o$-flat bar	$\frac{\sigma_x}{2}$	$\varepsilon_x - \varepsilon_y$
10^o-flat bar	$0.171\sigma_x$	60°-rosette: $-0.456\varepsilon_{0^o} - 0.857\varepsilon_{120^o} + 7.373\varepsilon_{240^o}$ 45°-rosette: $-1.282\varepsilon_{0^o} + 1.879\varepsilon_{45^o} - 0.593\varepsilon_{90^o}$
0^o, 90^o rail shear	$\frac{F}{2A}$	$2\varepsilon_{45^o}$

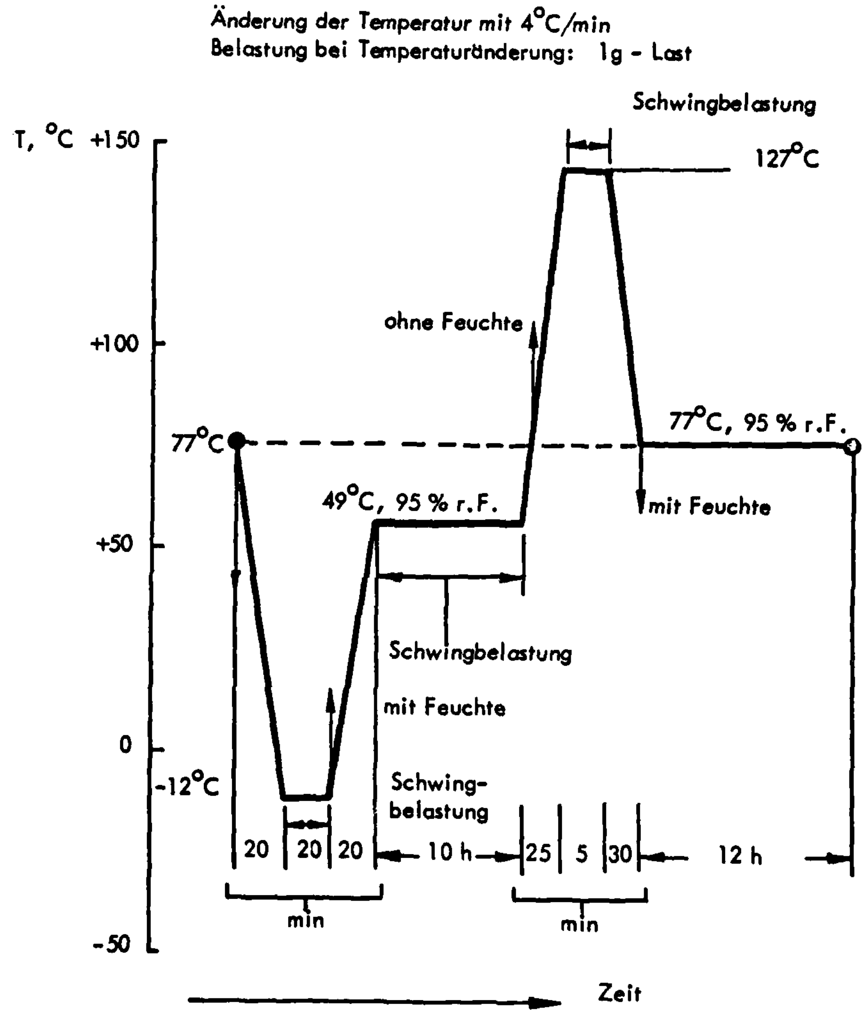
T: torque
A: surface parallel with load
Index x: in longitudinal direction
Index y: normal to longitudinal direction

Fig 9.3



Key:
Last Schrieb = load trace

Fig 9.4 Load sequence with superimposed temperature change in flight-by-flight test (Ref 149)



Key:
Änderung der Temperatur mit $4^{O}C/min$ = change in temperature by $4^{O}C/min$
Belastung bei Temperaturänderung = loading during temperature change
Mit/ohne Feuchte = with/without moisture

Fig 9.5 Temperature change with superimposed load cycle from 100 flight test programme: B-1 horizontal surfaces spar specimen (Ref 613)

10 ABSORPTION AND EFFECTS OF MOISTURE

J.J. Gerharz

Contents

10.1 Moisture absorption

Moisture can change the strength properties of fibre composite materials to a great extent. Great efforts are therefore made to model the mechanism of moisture absorption by fibre composites on the one hand and to estimate the consequences of absorbed moisture in the material on the other.

Moisture can be absorbed:

(i) through the fibre-matrix bonding layer,

(ii) through cracks and voids in the fibre composite, and

(iii) through the resin (diffusion).

Fibre composites with epoxide matrix absorb moisture (water) primarily by spontaneous surface absorption, followed by diffusion in the matrix. Comparisons between the saturation volume of resins and fibre composite materials as well as between fibre composite materials with different proportions of resin indicate that the epoxide resin contains all or at least most of the absorbed water[528].

For calculation purposes it can therefore be assumed that boron and carbon fibres do not absorb moisture. Cracks and voids have an effect on moisture absorption since they form additional surfaces for the absorption-diffusion process. In addition manufacturing faults in the fibre/matrix bonding layer can form microcracks which accelerate the movement of moisture by capillary action[528]. However, open fibre ends have no effect on moisture absorption if there are no faults in the bonding of fibre and matrix, as comparisons of the volume of moisture absorbed for covered and non-covered fibre ends have shown[356].

Diffusion

The moisture absorption or desorption of fibre composites depends on their condition and the humidity level of the environment; the moisture content can be found by weighing the fibre composite. It is normally related to the weight of the dry fibre composite by the ratio:

$$M = M(t) = \frac{\text{weight of wet fibre composite} - \text{weight of dry fibre composite}}{\text{weight of dry fibre composite}}$$

Apart from time, the moisture content of the fibre composite depends on the following parameters:

- humidity level of environment; this also determines the maximum moisture content (saturation) of the fibre composite, see Fig 10.1,

- temperature of the environment; this determines the diffusion coefficient and thus the speed of moisture absorption, see Fig 10.2,
- the thickness of the laminate; this has a major effect on the speed of moisture absorption, see Fig 10.3.

In addition to the experimental results (points) these diagrams also show the curve of a theoretical analysis of moisture absorption which agrees well with the experimental values.

Analogous to the laws applying to heat transfer, provided that the temperature in the fibre composite is the same as the environmental temperature and the diffusion coefficient does not change with the moisture content, the momentary distribution of the moisture content through the thickness of a fibre composite plate can be calculated according to Refs 573, 648 and 650 as:

$$\frac{c - c}{c_m - c} = 1 - \frac{4}{\pi}\sum_{i=0}^{\infty}\frac{1}{2i + 1} \sin \frac{(2i + 1)\pi Z}{h} \exp[-(2i + 1)^2\pi^2 t^*]$$

with $$t^* = \frac{D_z \int_0^t \exp[-B/T(t)]\,dt}{h^2} \qquad \text{(Ref 567)}$$

where c_0, c and c_m : moisture concentration at site z in the plate at times 0, t and ∞ (saturation) .

The curves of moisture distribution shown in Fig 6.13 were determined by this means. As a rule 10 terms (i = 10) of the progression are sufficient for adequate accuracy. The momentary value of the total moisture content is obtained by integration over the thickness of the plate:

$$\frac{M - M_0}{M_m - M_0} \approx 1 - \exp\left[-7.3\left(\frac{D_z t^+}{s^2}\right)^{0.75}\right] \qquad \text{(Ref 573)}$$

with $$t^+ = \int_0^t \exp[-B/T(t)]\,dt$$

where M : momentary weight of water in the fibre composite

M_0 : weight of the water in the fibre composite before exposure

M_m : weight of the water in the fibre composite on saturation

D_z : diffusion coefficient for the fibre composite in direction of thickness

s : h (thickness of composite if moisture penetrates from both sides) $2 \times h$ if moisture can penetrate from one side only.

The diffusion coefficient of the fibre composite can be calculated, if not experimentally determined, from the diffusion coefficient of the resin, D_h with

$$D_z = \left(1 - 2\sqrt{\frac{V_f}{\pi}}\right)D_h \quad . \qquad \text{(Ref 573)}$$

As shown in Fig 10.2, it depends on the temperature.

The moisture content at saturation M_m is, as has been proved experimentally, independent of temperature but dependent on the humidity level of the environment. For fibre composites stored in water M_m is a constant. For fibre composites exposed to moist air M_m depends on the relative humidity ϕ, according to the equation $M_m = a\phi^b$. The constants of this equation a and b can be determined experimentally as Ref 573. The calculation steps described heretofore take no account of the diffusion in longitudinal and lateral direction; however, the error remains small as long as the dimension ratios thickness/length and thickness/width of the fibre composite component are much smaller than 1. A method is quoted in Ref 573 by which diffusion can be allowed for in all three directions.

10.2 Effects of moisture on fibre composites

A direct consequence of moisture in a fibre composite is spatial expansion in the matrix. According to measurements in Ref 528 the percentage increase in volume is less than the percentage increase in weight of the matrix, but since spatial expansion of the matrix is restrained by the fibres, residual stresses occur. In multi-directional composite these residual stresses lie in the laminate plane[566]. The expansion of the matrix in the direction of thickness will be correspondingly greater as it is not restrained by the fibres. Methods of calculating these residual stresses are contained in section 6.3.3. Sections 1 and 2 of this Report deal with the effects of moisture absorption on static strengths and fatigue strength.

One consequence of moisture in the plastic matrix is depression of the glass equilibrium temperature GET (above this temperature the plastic is soft, below it is hard). The effect of the moisture content on the glass equilibrium temperature can be determined by thermomechanical tests[92,343,528], see Fig 10.4.

This can be described numerically as Ref 648:

$$T_g = \frac{\alpha_p V_p T_{gp} + \alpha_w V_w T_{gw}}{\alpha_p V_p + \alpha_w V_w}$$

where T_g = GET of composite
T_{gp} = GET of plastic matrix (resin)
T_{gw} = GET of water (approximately 4^oC)
α_p = expansion coefficient of plastic matrix
α_w = expansion coefficient of water
V_p = volume fraction of plastic matrix
V_w = $1 - V_p$

and the indices signify:

p = polymer
w = water .

Allowing for the increase in weight of the fibre composite, M_c

$$V_p = \frac{1}{1 + \frac{\gamma_c}{\gamma_w} M_c (1 - V_f)}$$

where γ_c = specific weight of fibre composite, g/cm^3
γ_w = specific weight of water, g/cm^3
V_f = volume fraction of fibres in fibre composite

and the indices signify:

c = composites
f = fibres .

For epoxide resin and water the coefficients of expansion are $\alpha_p = 3.78 \times 10^{-4}/^oC$ and $\alpha_w = 4 \times 10^{-3}/^oC$ respectively. Instead of the usual thermal expansion coefficients, difference values $\alpha_L - \alpha_g$ must be used in these equations, where α_L is the linear thermal expansion coefficient above T_g and α_g the linear thermal expansion coefficient below T_g .

Fibre composites have reduced mechanical properties at high temperatures compared with room temperature, *eg* the drop in compressive strength at high temperature is greater the nearer the test temperature is to the glass equilibrium temperature. If moisture absorption during operation reduces the glass equilibrium temperature of the composite, the drop in mechanical properties will be greater at the same test temperature because of the drop in the glass equilibrium temperature.

To summarise, the following can be said on moisture absorption:

- moisture absorption in fibre composites is by diffusion of the water through the matrix material,
- moisture absorption can be measured by change in weight,
- theoretical predictions agree relatively well with measured moisture absorption,
- the most important parameters are:
  - humidity level of the environment
  - temperature of the environment,
  - laminate thickness,
- the direct consequences are spatial expansion of the matrix material, producing internal stresses in the laminate and a reduction in the glass equilibrium temperature,
- the glass equilibrium temperature of moist laminates can be predicted.

10.3 Moisture content in components

In the constantly changing environmental conditions during operational use absorption and desorption will alternate. Correspondingly the average moisture content, the distribution of moisture and thus also the distribution of residual stresses in fibre composites will be subject to continuous alteration. Long-term analyses for CFC and BFC[565] show the following tendencies:

- only after many years of use in changing environmental conditions (moisture and temperature) will an almost constant moisture content be reached in the interior of the fibre composite plate, but the moisture content in the surface boundary layer will continue to change with external conditions,

- the constant moisture content which is reached in the interior under changing environmental conditions can also be reached under constant environmental conditions,
- while surface protection cannot completely prevent moisture absorption, it can delay it considerably by providing a lower diffusion coefficient[369,565]; furthermore moisture absorption by the fibre composite depends on the saturation content of its surface protection; the lower the saturation content of the latter, the lower will the moisture content of the fibre composite remain[565].

In fibre composite structures with epoxide matrix which had been in use a long time (16 years) no more than 1% average moisture content was found in the fibre composite material[528]. However this reveals nothing of the distribution of the moisture concentration.

10.4 References

The references evaluated in the preceding sections are classified according to contents. The papers cited in the text are underlined.

Section 10.1

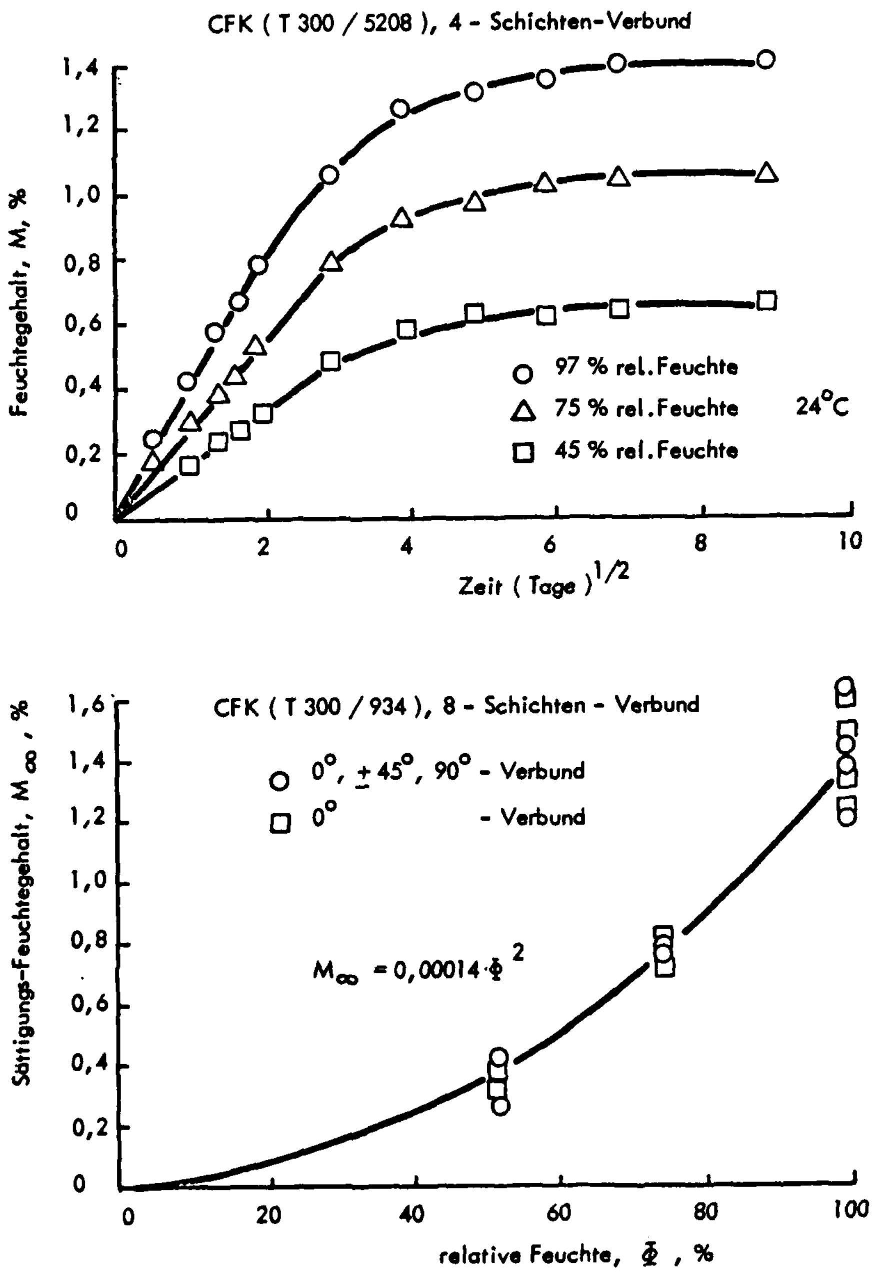
Moisture absorption:
172, 214, 257, 322, 356, 427, 467, 516, 518, 528, 530, 553, 565, 566, 567, 569, 572, 573, 601, 648, 650.

Section 10.2

Effects of moisture in fibre composites:
92, 343, 528, 566, 648, 650.

Section 10.3

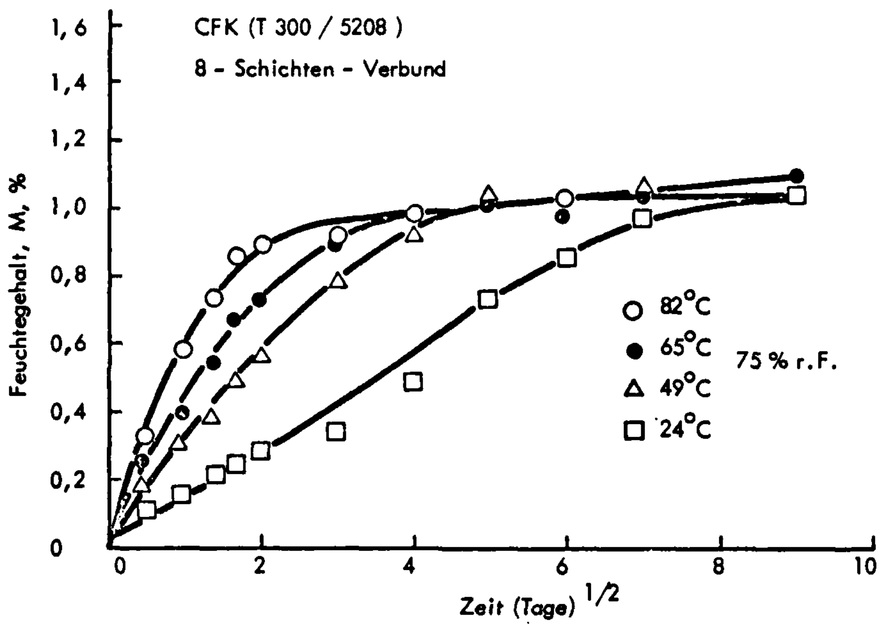
Moisture content in components:
369, 528, 565.

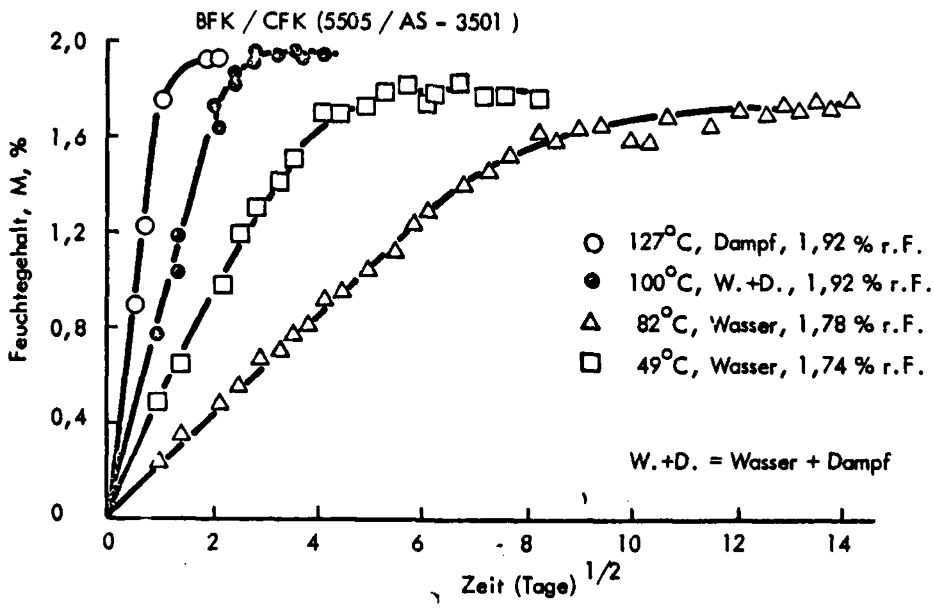


Key:

4-Schichten-Verbund	= 4-layer composite
Feuchtegehalt	= moisture content
Sättigungs-Feuchtegehalt	= saturation moisture content
Relative Feuchte	= relative humidity

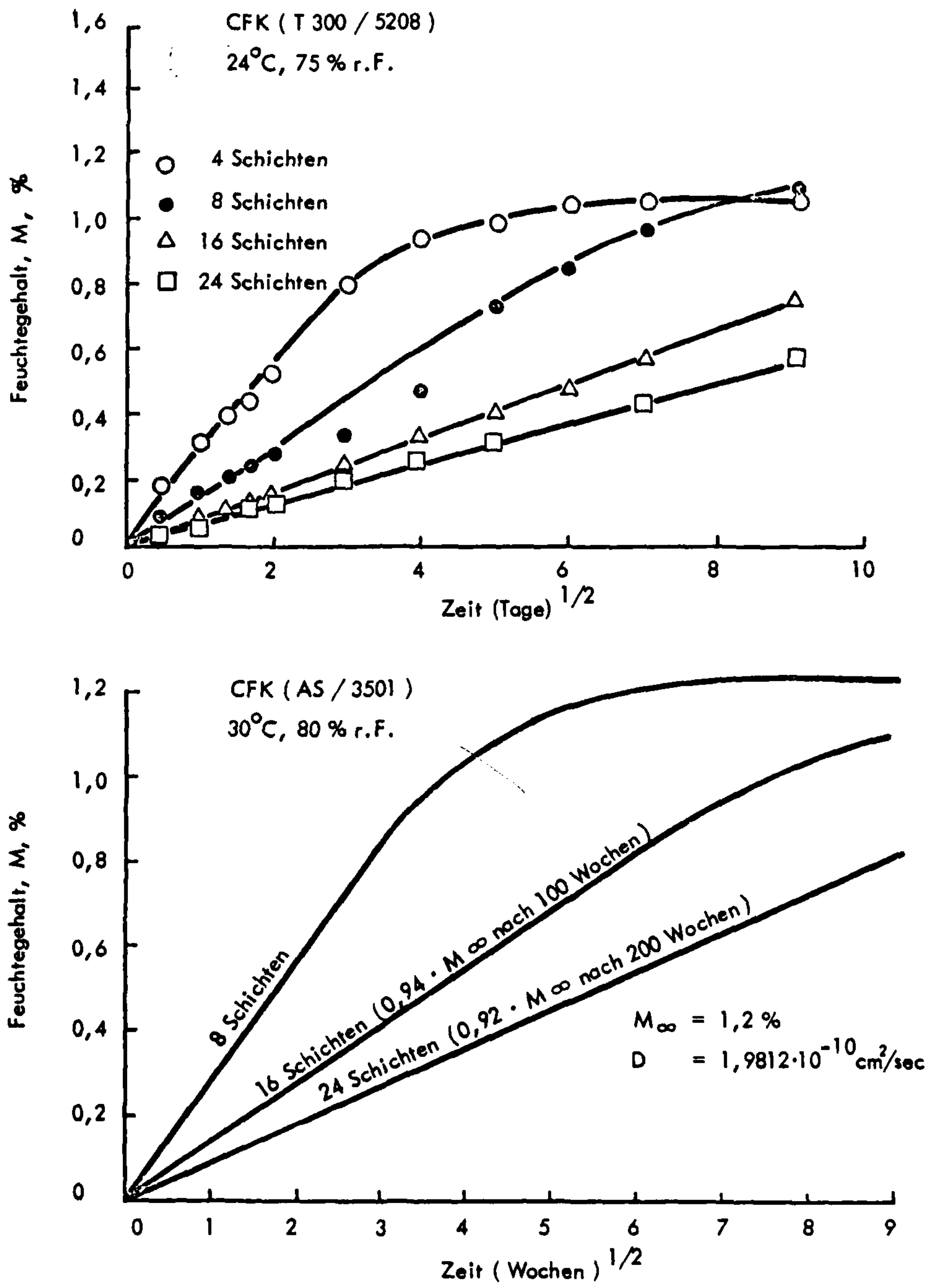
Fig 10.1 Effect of relative humidity on moisture absorption (Ref 650)





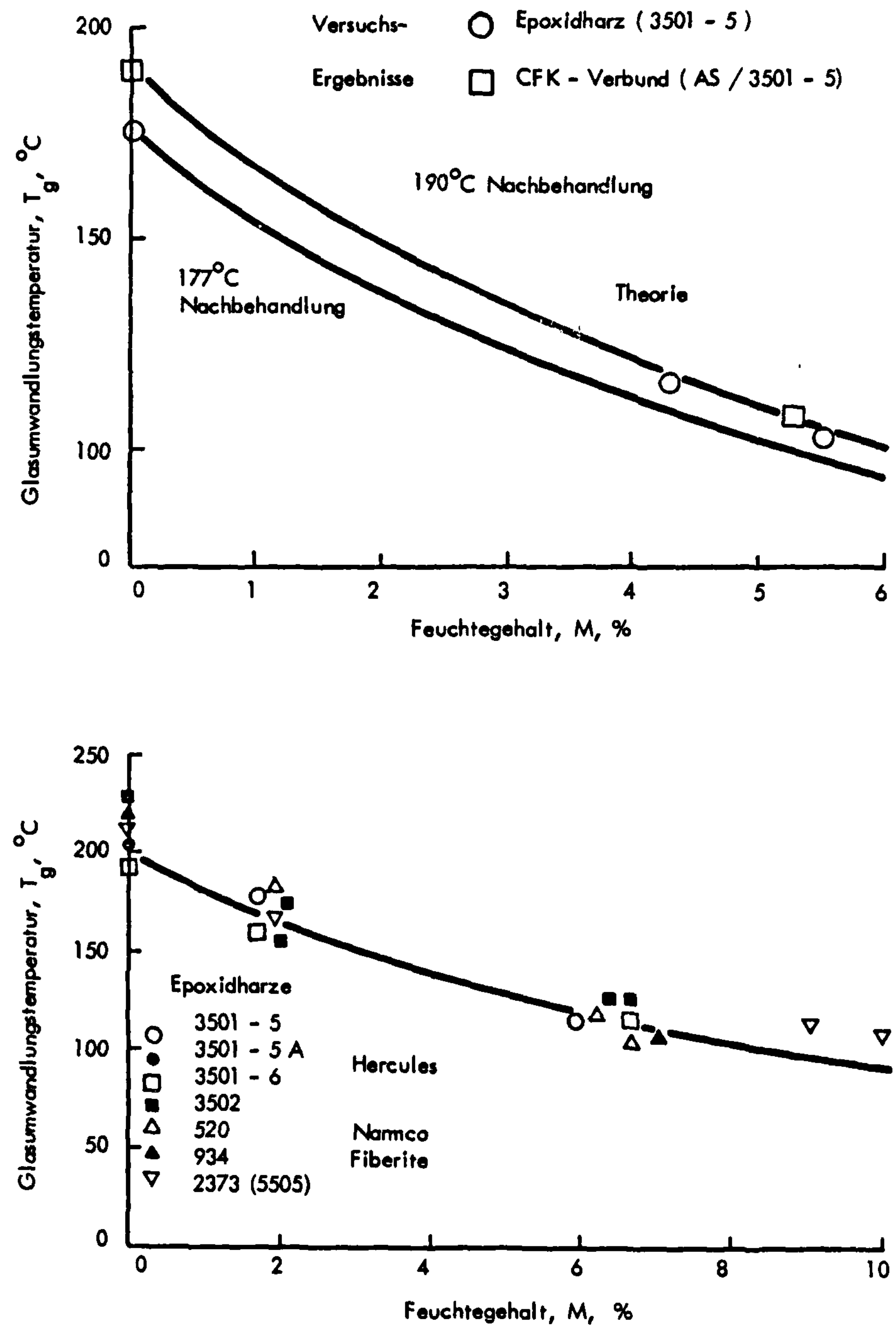
Key:
Zeit (Tage) = time (days)
Dampf = vapour
Wasser = water

Fig 10.2 Effect of temperature on moisture absorption (Refs 650, 528)



Key:
Wochen = weeks

Fig 10.3 Effect of laminate thickness (number of layers) on uptake of moisture (Refs 650, 528)



Key:
Nachbehandlung = post treatment

Fig 10.4 Effect of moisture content on glass equilibrium temperature (Refs 648, 528)

11 APPLICATION OF FIBRE COMPOSITE MATERIALS

J.J. Gerharz

Contents

11.1 Introduction

There have already been many reports on applications and possibilities for the use of fibre composites[162,204,234,332,355,430,433,476,492]. Many publications[300,314,360,371,425,426,473] contain comparisons of the characteristics of fibre composites with those of metals, this covering all points of view of importance for structural use, including economic factors[151,154,314,360,384,406,512] and construction methodology[314,401,407,428]. The object of all applications of fibre composites is to exploit their advantages. The process of development towards this aim is not yet concluded, particularly in regard to their use for primary components.

The following section presents completed and projected examples of applications and areas. It indicates to what extent the above-mentioned target has been achieved and what developments are still required in the area of fibre composites and their application in order to approach the target more closely. Since the great majority of all applications refer to aircraft construction fibre composite structures for aircraft are used as an example.

Fibre composites have been used in the aircraft and space industry to an increasing extent since the sixties. In these branches of industry the necessary means for the development of these new materials were available, not least because their use in rockets, satellites, space shuttles, aircraft and helicopters is the most profitable. The reasons for this are the mechanical properties of fibre composites which work out particularly advantageously in these areas. If these properties can be properly exploited:

- considerable saving in weight,
- lower manufacturing costs, and
- lower operating costs,

can be achieved in comparison with conventional metal materials.

Weight can be saved because fibre composite materials have high

- specific stiffness, *eg* E/γ, and
- specific strengths, *eg* σ_B/γ,

as well as sufficiently high values of

- specific residual strength, or
- impact toughness.

The manufacturing costs of fibre composite structures can be reduced because:

- in fibre composite components the proportion of machining in manufacture can be small,
- fibre composite components such as large integrally reinforced shell sections can be manufactured simply (for example, co-curing in an autoclave, press moulding, short-fibre technique), which could mean a reduction in the number of individual parts of a structure (lower assembly costs), and
- because fibre composites can be bonded satisfactorily.

The operating costs of aircraft and vehicles are lower because of the reduction in fuel consumption due to weight saving. For 1kg structural weight a modern aircraft requires 1-1.5kg maximum fuel service load. For rockets the amount of fuel required for 1kg structural or satellite weight is higher by a power of ten, so that the use of weight saving fibre composites will be most profitable here. In addition to the reduction in fuel costs, the resistance to corrosion of fibre composites with plastic matrix will lead to a reduction in maintenance costs. Of course the above reductions in costs in comparison with metal structures have to be weighed against the higher raw material costs of fibre composites and their development costs. Cost-efficiency analyses of fibre composite structures in use in various fields show, however, that these can be outweighed by savings in cost.

The advantages in the use of fibre composite materials apply primarily to their use as a structural material, *ie* for the manufacture of complete structural parts, but they also apply, even if to a lesser degree, to their use as a reinforcing material in so-called hybrid methods where fibre composites are glued to metal components in order to increase the rigidity, strength or life of the metal structure.

11.2 Examples of applications

In various areas of machine construction fibre composites are used, *eg* for the following components:

<u>Aircraft</u>

<u>Airframe</u>

Horizontal stabilisers and elevators	
Fin assembly	Primary structure
Wing	
Fuselage parts	

Lift devices	
Air brakes	
Rudder, elevator, wing flap	Secondary structure
Fairings	
Landing gear doors	
Access doors	

<u>Engine</u>

Fan blade	
Fan frame	Fibre hybrid composites
Engine fairing	
Blades of first compressor stage	

<u>Helicopter</u>

<u>Airframe</u>

Tail boom
Cabin parts

<u>Drive</u>

Rotor blades
Shafts (tail rotor)

<u>Rocket</u>

Pressure tank
Airframe parts

<u>Satellite</u>

Antenna carrier
Antenna shell

<u>Space shuttle</u>

Fuselage skin
Hold doors
Supports

<u>Boats</u> (mine detector, air cushion, fishing, hydrofoil)

Hull
Hydrofoil

<u>Trams</u> (carriage reinforcement)

Flat spring
Drive shafts

<u>Buildings</u>

Facing panels
Roofing tiles

<u>Apparatus</u>

Storage tanks

<u>General machine construction</u>

Centrifuges
Grabs (weaving machine)

Below the properties of fibre composites are listed for some important examples for which their use appears to be particularly advantageous (column 2). Column 3 quotes special operational conditions to which particular attention must be paid in design.

Area of application	Important property of fibre composites	Special operational conditions
Aircraft Helicopters	High specific mechanical properties, Low weight	Humidity, Lightning, Impact
Space travel	Thermal stability at low temperatures, Low thermal expansion, Low weight	UV radiation, Wide temperature differences
Boats	Corrosion resistance, High specific bending rigidity	Cavitation, Impact
Engines	Thermal stability, Little tendency to creep	Impact, Acoustic pressure, Resonance

Cost saving as a further important property of fibre composite construction by, *eg* easier manufacture is desirable in any field of use. It is in the forefront particularly for civil uses.

More than one motive usually determined the use of fibre composites as a construction material for the products mentioned; however a cost-efficiency analysis is the deciding factor in individual cases, see section 11.7.

11.3 Use of fibre composites for components of the secondary structure of aircraft

Considerable weight and cost reductions are achieved particularly in the manufacture of secondary components from fibre composites. This is one of the most important reasons for the increased use of fibre composites for secondary aircraft components. An idea of the frequency of application is given by a report[788] from which Table 1 with examples of American military aircraft is reproduced in Figs 11.1 to 11.4. This table is supplemented by applications in military aircraft produced outside America in Fig 11.5. Further Tables 2 to 4 in Figs 11.6 to 11.8 show uses in civil aircraft, helicopters and space equipment. Most of the aircraft secondary components with fibre composites are sandwich constructions with fibre composite covering layers and honeycomb cores extending over the full thickness of the component, see section 11.4.

These fibre composite sandwich constructions are cheaper than the metal constructions used heretofore and can be manufactured to a lower overall weight. In general the use of fibre composite construction methods for secondary structures in aircraft production has involved a reduction in the number of individual parts per structure, fewer production processes and a reduction in manufacturing costs of 10-25% with 15-25% saving in weight as compared with metal construction, as shown by examples in Table 5 in Figs 11.9 to 11.11. Here applications have been compiled from the aerospace industry, for which data were available in the literature on type of construction, weight saving, costs, number of individual parts, etc.

11.4 Use of fibre composites for components of the primary structure of aircraft

In contrast to many secondary components of an aircraft, the sandwich method with honeycomb core extending over the full thickness of the component cannot be used for primary parts because of the usually greater thickness of the component. Therefore the substructure of lifting surfaces and tail units consists of ribs and spars and in the case of fuselage parts of frames, stringers and partitions as in conventional metal structures, see Figs 11.13 to 11.16. The covering parts themselves, however, can be produced in sandwich style with, *eg* CFC covering layers and aluminium honeycomb core, see Fig 11.12. A further special feature of fibre composite construction methods in lifting surfaces and tail units is the use of CFC spars and ribs with so-called sine wave webs where the substructure has to be designed for stability because of the thickness of the component. This construction fulfils stability requirements with low weight, *ie* it is optimum as to weight. Some of the newer primary constructions carried out in fibre composite are considerably cheaper to manufacture, mainly because the number of their individual parts is drastically reduced as compared with metal construction, see Table 5 in Figs 11.9 to 11.11. In general, weight savings of 20% and cost reductions of 15% have been achieved by using fibre composite methods for primary structures. Section 11.3 reports on efforts to achieve greater savings in weight and cost by better exploitation of the potential of this method of construction.

11.4.1 Joint technology

Examples of applications show both bonded and bolted (or riveted) joints. Both have advantages and disadvantages. Practice has proved that at present the advantages of riveting are greater. Advantages and disadvantages of bonded and riveted joints are summarised in the following table:

	Bonding	Riveting
Advantages	Weight saving Little notch effect	Can be inspected Can be repaired
Disadvantages	Expensive manufacture Sensitive to moisture, therefore declining strength	Notch effect Weight of fasteners

Due to the notch sensitivity of fibre composites, particularly if there is a high proportion of 0^{o} layers in the laminate, see section 1, 0^{o} layers have been replaced by $\pm 45^{o}$ CFC layers in areas of the skin where holes are required for attachment to the substructure, in order to reduce notch sensitivity. Furthermore the number of layers has been increased in these sites in order to regain the strength level of the undisturbed cross-section[63,343,377]. Tests with locally reinforced holes or cut-outs have demonstrated that after laying in additional layers, strength in the cross-section with holes is as high as in the undisturbed cross-section.

Practical examples for the joining fibre composite skin and substructure with bolts are the B-1 horizontal tail surfaces[343], the F-16 vertical tail surfaces[308,551], the F-18 vertical tail surface[551] and the AV-8B (Harrier) lifting surface[614]. In practice there are also examples of adhesive joints in highly stressed parts. The best known are bonds in the CFC skin/titanium surface joints in the F-18 lifting surface. The production of these bonds was, however, connected with a great deal of manual fitting effort and was therefore very expensive. After the static whole airframe tests with the F-16 horizontal tail surfaces it was, moreover, decided to reinforce these bonds with bolts. The stepped adhesive bond in the F-18 wing connection, however, has no bolts in the junction area.

Adhesive bonds which are more favourable in weight than riveted joints will presumably be used more extensively only when more suitable constructions have been developed which simplify the production of high-strength adhesive bonds.

11.5 Use in hybrid methods of construction

There are also many examples of the use of fibre composites in hybrid methods for strengthening, stiffening or increasing the useful life of metal structures. The most important known applications of so-called hybrid technology are quoted below together with their objects:

- wing centre section C-130 - swing wing bearing mount F-111	increase in life
- tail cone, CH-54-B - fuselage structure, space shuttle (study)	stiffening
- fuselage structure B727-200 (study)	strengthening

These examples are described in survey publications such as[332,433,551] and also explained briefly in Ref 788. Savings in weight were always achieved in these examples which were carried out in practice. No generally applicable values can be quoted since the literature does not contain the relevant data. The studies show that up to 30% saving in weight is possible in hybrid new designs, but that the use of complete components of fibre composite in the same structure leads to greater saving in weight[788]. The number of applications of fibre composite as a strengthening material is very small compared with the number of applications as a structural material. Nor is this likely to change for future projects in the opinion of the American aircraft industry[787]. The importance of this application of fibre composites in metal structures should not be lost sight of, however, where the following problems are concerned:

- increasing the static and fatigue strength of a metal structure by reducing stress in critical areas,
- increasing stiffness, and
- increasing crack propagation life and residual strength of cracked metal components.

Tests and practical experience have shown that these targets can be reached by hybrid construction with little expenditure of weight and costs, particularly because:

- the bonded reinforcement does not induce a new weak area (as in holes for fasteners),
- the reinforcement can be adapted in the best possible way to the problem by the shape and direction of the bonded laminates, and
- when this is applied to an existing design it is very cheap compared with other solutions.

11.6 Further development in the area of fibre composite technology

Even though great progress has been made in the design of modern aircraft frames in regard to weight saving, as shown by the data in Table 5 of Figs 11.9 to 11.11, retention of present construction methods would no longer meet the demands of future aircraft. It can be foreseen that higher performance of aircraft will require not only lighter airframe structures but also significantly reduced manufacturing costs. The American aircraft industry is endeavouring to achieve this by more cost-favourable design. These efforts are directed in the case of metals to superplastic forming diffusion bonding and 'hot isostatic extrusion' (sintering technology) and in the case of fibre composites to, *eg* the use of large integral components and matrix materials with simplified curing processes (without autoclaves), for example curing PBBI in a vacuum sack and PMR process for polyamide matrix composites, see next section. The intention in both cases is to replace a number of processes by single pressing, forging, casting or bonding processes in order to reduce the man-costs per kg structure.

11.6.1 Development of fibre composite construction methods

Many of the fibre composite components produced so far are still so-called substitute structures, in which metal parts and their fastenings are taken over by equivalent fibre composite parts. Although this application has led to savings in weight and sometimes to competitive prices, it has obstructed the development of very 'efficient' methods of fibre composite construction.

The use of modern fibre composite materials will lie mainly in secondary and primary components and structures involved in aerodynamic lift. In general these components and structures are designed to torsion and bending strength requirements. CFC methods can yield savings in weight of 25-40% for these compared with metal methods. This is illustrated by an example in Fig 11.17. Specific stiffnesses of a balanced CFC angle ply composite are compared with those of steel, titanium and aluminium. In the development of fibre composite structures the requirement is therefore to progress from the 20% saving in weight at present achieved to 25-40% and in this way to make further savings in manufacturing costs. In contrast to secondary structures in fibre construction where optimum design and production concepts are already laid down to a great extent, corresponding concepts for primary structures with spar and rib substructure are less advanced. The US Air Force has initiated investigations for a fighter aircraft of the eighties to find out the structural concept with the highest proportion of fibre composite material which is most suited to build a smaller, lighter and cheaper aircraft compared with metal construction[789].

The following design concept is the result:

- extended use of large integral structural components by the use of common curving of skin, substructure and their connection to reduce manufacturing, matching (assembly jigs) and installation costs; for the lifting surface, for instance, the best concept is a continuous one-piece wing,
- the material used for almost all structural parts is CFC and the total proportion of fibre composites is between 60 and 70%,
- for the front part of the fuselage a fibre composite method developed by General Dynamics, called 'shell-liner concept' appears to be suitable and for the central part of the fuselage a fibre composite sandwich design is preferred because of the fuel tank.

In addition to the development of reinforced shell designs, fibre composite structures are also being developed which are intended for use in place of aluminium drop forgings. Here the emphasis is on weight reduction as demonstrated by the example of the airbus window frame. An investigation shows that this component can be produced cheaply in mass production by CFC short fibre pressing techniques[790].

Further possibilities of reducing manufacturing costs of fibre composite structures are[200]:

- formed holes (instead of drilled),
- simultaneous shaping and curing ('pultrusion' process),
- spar flange embedded in the skin to reduce the number of fasteners,
- increased use of wide multi-layer tape to reduce prepreg lay-up times.

11.6.2 Development of fibre composite materials

For quantity production up until the eighties the fibre composites available are:

carbon/epoxide
boron/epoxide
Kevlar/epoxide
glass/epoxide, and
boron/aluminium

of which glass and Kevlar prepegs are used primarily in hybrid composites.

The drawbacks in these materials are the high cost (except Kevlar and glass fibre) and the moisture absorption of the epoxide resin. The temperature application limit of +170°C for epoxide resins is adequate for most aerospace purposes, but the increasing interest of engine manufacturers in fibre composites is furthering the development of more temperature resistant plastics for the matrix.

The most important development targets are therefore:

- cheaper fibre composite materials,
- matrix materials which do not absorb moisture,
- matrix materials with higher temperature resistance than epoxide resin.

With the above aims in mind there are under development:

- fibre materials:
    - carbon fibres from a by-product of petroleum processing and coal coking - this is a cheaper carbon fibre with $E_z = 38000$ daN/mm^2, $\sigma_{zB} = 170$ daN/mm^2,
    - polycrystalline alumina fibres (aluminium oxide Al_2O_3) - this is a cheaper fibre (useable in plastic and metal matrix) with $E_z = 38000$ daN/mm^2, $\sigma_{zB} = 140$-190 daN/mm^2,

- boron fibre with carbon core, this is a cheaper boron fibre with E_z = 36500 and 38000 daN/mm^2 , σ_{zB} = 327 daN/mm^2 ,
- silicon carbide (SiC fibre), this is a cheaper fibre (than boron fibre) with E_z = 41000 daN/mm^2, σ_{zB} = 345 daN/mm^2 ,

- matrix materials:
  - polyamides in conjunction with PMR process in composite production. This matrix material is temperature resistant to 300°C and in addition the manufacturing process is cheaper than the conventional manufacture of polyamide composites,
  - PBBI plastic (polybutadiene, bissimide). The moisture effect is less in this matrix material and the composite can be made more cheaply (without autoclave and suction tissue), it is also advantageous for thick components and simultaneous hardening of matrix and adhesive.

The manufacturers are also propagating better exploitation of the potentialities of the organic fibre Kevlar 49, such as making use of the extremely good specific properties under tensile load at temperatures <100°C and the high impact resistance compared with inorganic fibres. They are therefore recommended as hybrid fibres (*ie* mixed with other fibres), where, *eg* carbon fibres or the new alumina fibres would take the compressive loads and Kevlar fibres would increase the impact resistance.

11.7 Costs when using fibre composite materials

Reducing costs

Although considerable cost savings have been achieved in some fibre composite structures, see Table 5 of Figs 11.9 to 11.11, all the possibilities of fibre composite construction methods have not yet been exhausted. Many of the present day fibre composite structures are still more expensive than metal structures even in serial production; at present this is mainly due to the high material costs which are not likely to drop below those of metals in the near future. The prices of fibre and fibre composite materials valid at the moment are quoted in Fig 11.18.

Related to the end price of fibre composite structural parts, which at present is around 1250-2500 DM/kg, price development of the raw materials no longer plays a decisive part. Today it is personnel and intrinsic costs which represent the greater part of the total cost. Therefore simplication of production methods, saving, *eg* personnel costs, will in future lead to greater price reductions. A target announced by the US Air Force is overall prices for fibre composite structures of the order of 500-1250 DM/kg. It is intended that this reduction should be achieved by the above mentioned developments in the area of fibre composite methods of construction and fibre composite production.

Cost-effectiveness analysis

A cost-effectiveness analysis is used in individual applications to check whether the use of fibre composites is economically viable, *ie* whether the anticipated gain is greater than the extra expense. Here the costs must be calculated for conventional and fibre composite methods for the entire period of use of the equipment. In military equipment, for instance fighter aircraft, the cost effectiveness analysis ends with the comparison of these costs, since *eg* differences in flight performance cannot be covered in price comparisons. For commercially used aircraft, however, the differences in the commercial profits must also be included in the analysis to produce a fair comparison, otherwise, for instance the advantage of a heavier useful load would not be included.

The following enter into the cost-effectiveness analysis[199]:

- prime costs:

  costs of : design
  research
  development
  testing
  evaluation
  manufacturing costs (dependent on number)
  tooling costs
  profit margins

- operating costs:

  inspections
  repairs
  maintenance
  spares

- depreciation.

11.8 References

The references evaluated in the preceding sections are classified by subject. The papers mentioned in the text are underlined.

Section 11.1

Introduction, application in general:

151, 154, 162, 204, 234, 300, 314, 332, 355, 360, 371, 384, 401, 406, 407, 425, 426, 428, 430, 433, 473, 476, 492, 512.

Weight:

25, 64, 151, 153, 154, 155, 156, 311, 314, 347, 362, 372, 427, 433, 491, 500.

Section 11.2

Sample applications,

Aircraft construction:

151, 152, 153, 156, 157, 162, 186, 203, 204, 234, 283, 285, 286, 288, 289, 291, 292, 294, 295, 296, 299, 301, 303, 304, 305, 306, 307, 308, 314, 330, 332, 346, 348, 349, 355, 380, 407, 427, 429, 476, 490, 500, 512, 544, 551, 552, 553, 554, 612, 613, 614, 616.

Helicopter construction:

153, 154, 201, 203, 284, 293, 297, 302, 314, 317, 347, 348, 349, 430, 432, 476, 490, 511, 512.

Rockets, satellites, space shuttles:

33, 34, 35, 155, 158, 159, 164, 171, 174, 195, 196, 203, 217, 282, 287, 289, 290, 295, 312, 320, 332, 430, 490, 500, 514.

Engine construction:

153, 183, 186, 234, 283, 299, 332, 358, 359, 426, 436, 437, 438, 439, 472, 478, 512.

Section 11.3

Use of fibre composite materials for components of the secondary structure of aircraft:

788.

Section 11.4

Use of fibre composite materials for components of the primary structure of aircraft:

35, 148, 149, 153, 154, 162, 201, 314, 334, 357, 377, 406, 433, 620.

Section 11.4.1

Joint technology:

63, 308, 343, 377, 551, 614.

Section 11.5

Use in hybrid method of construction:

23, 29, 35, 44, 63, 105, 113, 151, 153, 154, 155, 161, 231, 263, 314, 332, 330, 334, 377, 379, 406, 424, 428, 429, 433, 460, 470, 471, 551, 605, 608, 614, 669, 787, 788.

Section 11.6

Further development in the area of fibre composite technology,

Short fibre technology:

39, 49, 98, 190, 210, 221, 230, 231, 276, 317, 328, 345, 354, 358, 374, 387, 388, 392, 393, 394, 395, 425, 435, 494, 503, 507, 511, 555.

Inspection methods:

156, 209, 395, 493, 494.

Press forming:

316, 391, 534.

Section 11.6.1

Development of fibre composite construction method:

200, 789, 790.

Section 11.6.2

Development of fibre composite materials:

176, 219, 387, 395, 424, 461, 466, 467, 468, 469, 472, 491, 494, 512.

Section 11.7

Costs when using fibre composite materials:

25, 151, 153, 154, 155, 156, 199, 204, 234, 298, 307, 314, 332, 347, 360, 362, 371, 372, 384, 402, 406, 427, 428, 430, 432, 433, 460, 473, 491, 500, 512, 610, 611, 612.

Cost reduction:

347, 360, 362, 371, 402, 428, 430, 511, 512.

Cost-effectiveness analysis:

151, 154, 234, 314, 332, 360, 384, 406, 428, 430, 432, 433, 460, 473, 512, 199.

Firma	Flugzeug-Typ	Bauteil	Werkstoffe und Bauweise	Entwicklungsstand (April 1978)
Mc Donnell Douglas Subcontractor: Northrop u.a.	F-18	MLG Doors	CFK	(P) Z.Z.Fertig.f.Prototyp.
		NLG Doors	CFK	(P) "
		Dorsal Cover	CFK	(P) "
		Avionic Access Door	CFK	(P) "
		Wing Main Box Skins	CFK, Bolted	(P) "
		Wing Leading Edge Extension	CFK	(P) "
		Wing Trailing Edge Flap	CFK	(P) "
Rockwell-LAD	T-39	Wing Box	BFK, Bolted	(T)
Rockwell-LAD	F-100	Wing Skin	BFK, Bolted	(T)
Rockwell-LAD	B 1	Vertical Stabilizer Box	CFK, BFK Bolted	(T) Gebaut für 2 Prototypen geplante Flugerprobung findet nicht statt.
		Fuselage Weapons Bay Door	CFK	(T) "
		Aft Fuselage Electronic Bay Door	CFK	"
		Fuselage Dorsal Longeron	CFK	(T) "
		Fuselage Avionic Door	CFK	
		Spoiler	CFK	(T) "
Grumman	B 1	Horizontal Stabilizer	CFK, BFK, Bolted	(T) "
Rockwell-LAD	HIMAT	Fuselage Skin	CFK	(F) Technologie Träger
		Wing Skin	CFK	(F) "
		Canard Skin	CFK	(F) "

(D) = Entwicklung ohne Tests (T) = Entwicklung incl. Bodentests
(F) = Entwicklung incl. Bodentests und Flugtests (P) = Serienproduktion

Key:
Z.Z.Fertig.f.Prototyp. = at present manufacture for prototype
Gebaut für 2 Prototypen geplante Flugerprobung findet nicht statt. = built for 2 prototypes, planned flight test not taking place
Technologie Träger = technology carriers
Nur wenige Flugstunden = few flight hours only

Fig 11.1 (Table 1) Applications of fibre composites in military aircraft of American production (Ref 787)

Firma	Flugzeug-Typ	B a u t e i l	Werkstoffe und Bauweise	Entwicklungsstand (April 1978)
Northrop	F-5	Wing Leading Edge Section	CFK	(T)
		Wing Trailing Edge Flap	CFK	(T)
		Main Landing Gear (MLG) Door	CFK	(F) nur wenige Flugstunden
		Speed Brake	CFK	(T)
		Horizontal Stabilizer	CFK	(T)
		Vertical Stab. (Mean Spar)	CFK	(D)
Northrop	A-9	Rudder Panels	CFK	(F) 2 A/C limited Flighttest
Northrop	B 1	Flap Structure	CFK	(D)
Northrop	YF-17	Rudder	CFK	(F) 2 Prototypen
		Speed Brake	CFK	(F) "
		Vert. Tail Fixed Trail. Edge	CFK	(F) "
		Vert. Tail Leading Edge	CFK	(F) "
		Engine Bay Doors	CFK	(F) "
		Wing Trailing Edge Section incl. Flaps and Aileron	CFK	(F) "
		Leading Edge Extension	CFK	(F) "
		MLG Doors	CFK	(F) "
		NLG Doors	CFK	(F) "
		Avionics Access Doors	CFK	(D)
		Front Fuselage Section	CFK	(D)
Mc Donnell Douglas Subcontractor: Northrop u.a.	F-18	Vert. Tail Main Box Skins	CFK, Bolted	(P) Z.Z. Fertig. f.Prototyp.
		Vert. Tail Fixed Trail. Edge	CFK	(P) "
		Rudder	CFK	(P) "
		Horizontal Tail Skin	CFK, Bolted	(P) "
		Speed Brake	CFK	(P) "

(D) = Entwicklung ohne Tests (T) = Entwicklung incl. Bodentests
(F) = Entwicklung incl. Bodentests und Flugtests (P) = Serienproduktion

Fig 11.2 (Table 1 continued)

[redacted] LT [redacted]5

Firma	Flugzeug-Typ	B a u t e i l	Werkstoffe und Bauweise	Entwicklungsstand (April 1978)
MCAIR	F-15	Wing	BFK-Skin, CFK-Stringer	(T)
				(T)
MCAIR	AV-8B (Harrier)	Wing Torque Box	CFK, Bolted	(P) 3 Prototypen
		Flap	CFK	(P)
		Aileron	CFK	(P)
		Engine Access Doors	CFK	(P)
		Outrigger Fairing	CFK	(P)
		Flap Slot Door	CFK	(P)
		Trailing Edges	CFK	(P)
		Overwing Fairing	CFK	(P)
		Lid Strakes	CFK	(P)
		Forward Fuselage	CFK	(D)
Grumman	FB-111	Wing Box Extension	BFK	(T)
	F-14	Horizontal Stabilizer	BFK	(P)
		Landing Gear Doors	BFK, GFK	(F) 5 Sätze
		Overwing Fairing	GFK	(F) 8 Sätze

(D) = Entwicklung ohne Tests (T) = Entwicklung incl. Bodentests
(F) = Entwicklung incl. Bodentests und Flugtests (P) = Serienproduktion

Fig 11.3 (Table 1 continued)

Firma	Flugzeug-Typ	B a u t e i l	Werkstoffe und Bauweise	Entwicklungsstand (April 1978)
Rockwell-LAD	HIMAT	Stabilizer	CFK	(F) Technologie Träger
General Dynamics	F-111	Horizontal Tail	BFK	(F) 2 A/C
		Schwenklager-Bereich	Bor/Epoxy Doppler	(P) Retrofit
		Wing Pivot Fitting Fairing	CFK	(P) Retrofit
		AFT-Fuselage Keel	CFK, BFK, Bor/Alum.	(D)
General Dynamics	F-5	Center Fuselage	CFK	(T)
General Dynamics	F-16	Horizontal Tail	CFK-Skin	(P)
		Vertical Fin	CFK-Skin, Bolted	(P)
		Rudder	CFK-Skin, Bolted	(P)
		Forward Fuselage	CFK	(T)
		Wing	CFK	(D)
Vought	A-7D	Outer Wing	CFK und BFK	(F) 8 A/C
		Airbrake	CFK	(T)
Vought	S-3A	Spoiler	CFK	(F) 7 Sätze
MCAIR	F-4	Rudder	BFK	(F) 50 Sätze
		Access Doors	CFK	(F) 2 Sätze
	F-15	Vertical Fin	BFK	(P)
		Rudder	BFK	(P)
		Horizontal Tail	BFK	(P)
		Speedbrake	CFK	(P)

(D) = Entwicklung ohne Tests (T) = Entwicklung incl. Bodentests
(F) = Entwicklung incl. Bodentests und Flugtests (P) = Serienproduktion

Fig 11.4 (Table 1 continued)

Firma	Flugzeugtyp	Bauteil	Werkstoff, Bauweise	Stand
BAC	Jet Provost Mk 5	Ruder,	CFK, Haut u. Holme	Flugtest
BAC	Vulcan	Bremsklappe	CFK, Haut	Flugtest
BAC	Jaguar	Spoiler	CFK, Haut, Torsionsk.	im Bau
Dassault-Breguet	Mirage F 1	Höhenleitwerk	BFK	Entwicklung
Dornier	Alpha-Jet	Bremsklappe	CFK	Serie
Dornier	Alpha-Jet	Höhenleitwerk	CFK	Entwicklung
Dornier	Alpha-Jet	Flügel	CFK	Entwicklung
Panavia	Tornado	Höhenleitwerk	CFK	Projekt
SAAB-SCANIA	Viggen	Deckel im Flügel	CFK, HM	Flugtest
SAAB-SCANIA	Viggen	Flügelhaut-Teil	CFK, HM	Flugtest

Key:
Ruder = rudder
Bremsklappe = air brake
Höhenleitwerk = horizontal tail surfaces
Flügel = wing
Deckel = cover
Flügelhaut-Teil = wing skin part
Flugtest = flight test
Im Bau = under construction
Entwicklung = development
Serie = series

Fig 11.5 (Table 1 concluded) Applications of fibre composites in military aircraft of non-American production

Projekt	Hersteller	Bauteil	Material	Stand	Referenzen
B 737	Boeing	Spoiler	CFK	Flugtest	332
B 707	Boeing	Foreflap	BFK	Flugtest	332, 151
L - 1011	Lockheed	Querruder	CFK, Kevlar	Flugtest	332
L - 1011	Lockheed	Fairing	Kevlar	Flugtest	332
L - 1011	Lockheed	Bodenstützen	CFK,	Flugtest	552
DC - 10	Douglas	Seitenruder	CFK, GFK	Flugtest	332
DC - 10	Douglas	Aft Pylon-Haut	Bor-Aluminium	Flugtest	332
DC - 9	Douglas	Nacelle	CFK	Flugtest	332
VC - 10	BAC	Seitenruder	CFK	Flugtest	
VC - 10	BAC	Access Doors	CFK	Flugtest	
BAC 1-11	BAC	Flügelenden	CFK	Bodentest	
B 727	Boeing	Höhenleitwerk-Ruder	CFK	Entwicklung	
B 737	Boeing	Höhenleitwerk	CFK	Entwicklung	554
L - 1011	Lockheed	Seitenleitwerk	CFK	Entwicklung	332
DC - 10	Douglas	Seitenleitwerk	CFK	Entwicklung	554
DC - 8	Douglas	Höhenleitwerk	CFK	Entwicklung	
A 300	Deutsche Airbus	Fairing/Höhenleitwerk-Rumpf	GFK	Serie	295
A 300	Deutsche Airbus	Spoiler	CFK, BFK	Flugtest	
A 300	Deutsche Airbus	Querruder	CFK	Entwicklung	
A 300	Deutsche Airbus	Flap-Track	CFK-Titan	Entwicklung	
VFW-614	VFW-Fokker	Spoiler	BFK	Flugtest	
VFW-614	VFW-Fokker	Fahrwerktüren	CFK/Kevlar	Flugtest	
LTA	Dornier	Ruder, Nasen	CFK, Kevlar	Projekt.	
Lear Avia	Lear Fan 2100	Rumpf u. Flügel	CFK-Gewebe	Entwicklung	

Key:
Querruder = wing flap
Bodenstützen = base supports
Seitenruder = side rudder
Flügelenden = wing ends.
Fahrwerktüren = landing gear doors
Rumpf u. Flügel = fuselage and wing

Fig 11.6 (Table 2) Applications of fibre composites in civil aircraft projects

Projekt	Hersteller	Bauteil	Werkstoff	Stand	Referenzen
BO 105	MBB-UD	Rotor	GFK	Serie	431, 301
CH 35	Sikorsky	Rotor	CFK, GFK	Bodentest	201
CH 47	Vertol	Rotor	GFK	Entwicklung	
S-76	Sikorsky	Rumpfteile	GFK, Kevlar	Entwicklung	
SA 340	Aerospatiale	Rotor	GFK, CFK	Entwicklung	293
SA 360	Aerospatiale	Rotor	GFK, CFK	Entwicklung	293
SA 365	Aerospatiale	Rotor	GFK, CFK	Entwicklung	293
AH- 1 G1	Kaman	Rumpf-Heckteil	GFK	Entwicklung	199
CH - 54 B	Sikorsky	Rumpf-Heckteil	BFK-Verstärkung	Entwicklung	332
CH - 53	Sikorsky	Rumpf-Panel	Kevlar und CFK	Entwicklung	332

Key:
Rumpf-Heckteil = fuselage tail part

Fig 11.7 (Table 3) Applications of fibre composites in helicopters

L—145

Projekt	Bauteil	Werkstoff	Referenzen
Polaris Minuteman Trident Atlas	Treibstoff Druckbehälter	GFK und Kevlar	155, 300
Mariner IV	Antenne	Bor/Aluminium	155
Trident, C 4	Zellenteil	CFK-Gewebe	287
Shuttle	Türen für Spacelab	CFK	332
Shuttle	Rumpfhaut (hinten)	CFK	
Atlas	Zwischenstück	Bor/Aluminium	500
ATS, Satellit	Fachwerke	CFK	158

Key:
Treibstoff Druckbehälter = fuel pressure tank
Zellenteil = airframe part
Türen = doors
Rumpfhaut (hinten) = fuselage skin (rear)
Zwischenstück = intermediate piece
Fachwerke = trusses

Fig 11.8 (Table 4) Applications of fibre composites in space vehicles

Projekt	Bauteil	Gew.-Ant. Faserverbund	Faserverbundkonstruktion Werkstoffe / Bauweisen	Gewichts-ersparnis	Reduktion der Einzelteile	Kosten	Stück-zahl	Stand der Entwicklung	Referenzen
F - 111	Trailing Edge Panels		Bor/Epoxy-Haut/Alu-Waben mit Metall-kanten und -Beschläge	16 %		100 % Mehrk. (Ersatzk.-struktion)	22		551, 407
F-14 A	Overwing Fairing		Carbon/Glas/Epoxy Panel (Glasfaser-Gewebe) Bor/Carbon/Epoxy Träger Alu-Waben, f.d.	25 %		- 40 % (ab 100 Stück)	2 x 5	Erprobung (Versuche	204, 343, 379 615.
A-4	Landeklappen		Bor/Epoxy-Haut (2-teilig)/Alu-Waben, f.d. Metall-Rippen und -Beschläge	22 %	70 %			Fliegerprob..	203, 314, 156,
			Carbon/Epoxy-Haut (einteilig)/Alu-Waben, f.d. Carbon/Epoxy-Rippen formgepreßt	39 %	89 %			Produktion	407,
B-1	Bay-Doors Weapons		Carbon Epoxy	12 %		- 20 %	2 je Probentyp	Entwicklung	787,
F-4	Ruder		Bor/Epoxy-Haut/Alu-Waben, f.d. (verklebt)	36 %			45	Einsatz 1969	551, 355, 203, 314, 407
F-15	Ruder (Seitenleitwerk)		Bor/Epoxy-Haut, Carbon/Epoxy-Unterstruktur Alu-Waben	20 %				Einsatz 1972 in Serie gebaut	551, 307, 303,
Alpha-Jet	Bremsklappe	90 %	Carbon/Epoxy-Haut Carbon/Epoxy-Versteifung	25 %				Serie	295,
F-111	Höhenleitwerk		Bor/Epoxy-Haut/Alu-Waben, f.d. Anschluß: Stahl/Titan-Unterstruktur/Titan-Platte mit Bor/Ep. Haut verklebt Glas/Epoxy-Kanten; Bolzen	25 %			2	Flugerprobung Produktion	152, 355, 429, 314, 203,
A-4 Skyhawk (Navy)	Höhenleitwerk		Carbon/Epoxy-Haut (geklebt+genietet) Carbon/Epoxy, Alu-Waben-Rippen und -Holme, CFK-Anschlußwinkel	28 %					203, 407
F-15	Tragfläche		Carbon/Epoxy Stringer Bor/Epoxy Haut, Sandwich	19 %		teurer		Entwicklung	787.
YF-17	Rumpfvorderteil		Carbon/Epoxy-Haut und Zwischenwände	40 %	80 %	- 30 %	1	Entwicklung	292,

Fig 11.9 (Table 5) Examples of applications

Projekt	Bauteil	Gew.-Ant. Faserverbund	Faserverbundkonstruktion Werkstoffe / Bauweisen	Gewichtsersparnis	Reduktion der Einzelteile	Kosten	Stückzahl	Stand der Entwicklung	Referenzen
F-14	**Höhenleitwerk**		Bor/Epoxy-Haut, Alu-Waben, f.d. verklebt mit gestuftem Titan Pivot-Beschlag Alu-Haut an Vorder- u. Hinterkante	19 %				im Einsatz (Ganzzellenversuche)	355, 203, 343, 407
F-111	**Rumpfteil, hinten (zwischen den Triebwerken)**	61 %	Carbon/Ep. Haut/Stringer, Carbon/Ep. Haut/Alu-Waben: Wände 19 %; Bor-Alu Haut/Stringer, Spant 19 %; Bor-Ep., Spant, 23 %; Carbon-Ep., Spant, 25-28 %: Formteil + Laminat	18 %			1	gebaut zur Entwicklung d. Technologie) statt Versuch	355, 152, 44
F-16	**Rumpfteil, vorn**	62 %	Carbon/Epoxy Shell-Liner Concept CFK Umhüllung von CFK und CFK/Kevlar Verstärkungen, integrierte Carbon/Epoxy Rumpfholme	21 %	40 %	- 21 %		Entwicklung	305,
B-1	**Oberer und unterer Longeron**		Carbon/Epoxy + Bor/Epoxy	44 %			2 je Probentyp	Entwicklung	
B-1	**Seitenleitwerk Box**		Carbon/Epoxy, Bor/Epoxy genietet	15 %	15 %		2	Entwicklung	551
F-15	**Leitwerk**		Bor/Epoxy-Haut, Alu-Waben f.d., Titan Unterstruktur [307] Alu-Haut, Alu-Waben für Vorder- und Hinterkante	20 %				Einsatz 1972 in Serie gebaut	551, 307, 303
F-16	**Höhenleitwerk**		Carbon/Epoxy Haut, Alu-Wabe f.d. Anschluß an Titan-Beschlag: Klebung + Nietung	23 %					551,
F-16	**Seitenleitwerk**		Carbon/Epoxy Haut, Alu-Wabe f.d. im Vorderteil, sonst vernietet mit Alu-Unterstruktur Klebung und Nietung, GFK-Zwischenlagen	23 %					551, 308,
B-1	Höhenleitwerk	75 % CFK 12 % BFK 11 % Metall	Carbon Bor/Epoxy Haut CFK-Softening Strap Konstruktion Carbon/Epoxy Holme und Rippen	15 %		- 15 %	2	Entwicklung	551, 343, 613,
A-7	Flügel-Außenteil		Carbon/Bor/Epoxy-Haut/Alu-Waben im Bereich von Bohrungen nur CFK Carbon/Epoxy-Holm, Rippen/Alu-Waben - Stege (teilweise), Alu-Leg.-Rahmen (Holme, Scharnier-Rippe)	4 % (Flügelgewicht)	von 42 auf 27		12 (8 in Serie)	Flugerprobung, Full Scale Test	551, 553, 294

Fig 11.10 (Table 5 continued)

Projekt	Bauteil	Gew.-Ant. Faserverbund	Faserverbundkonstruktion Werkstoffe / Bauweisen	Gewichtsersparnis	Reduktion der Einzelteile	Kosten	Stückzahl	Stand der Entwicklung	Referenzen
YAV-8 B	Flügel	70 %	Carbon/Epoxy Haut, Holme, Rippen, Vernietung, Dichtmittel für Treibstoff, CFK-Gewebe für sinusförmige Stege von Holm und Rippen	20 %			3	Prototyp (Harrier-Nachfolger)	614
C-5 A	Flügelvorderkante		Bor/Epoxy-Haut/Alu-Waben, Bor/Epoxy-Rippen, Alu-Track-Rippen, Klebung Klarsicht Epoxy Anstrich	22 %	von 800 auf weniger als 80		12	Einsatz	152, 355, 314, 407,
B 737	Spoiler	35 %	Carbon/Epoxy-Haut/Alu-Waben, Alu-Rahmen	17 %					332, 727
			Carbon/Epoxy-Haut, Alu-Waben, Carbon/Kevlar/Epoxy-Rahmen	22 %					
B 707	Foreflap		Bor/Epoxy + Alu-Wabe Sandwich Haut + Ti-Endrippen	25 %	von 9 auf 3		2	im Einsatz seit 1970	552, 151,
DC-10	Seitenruder	57 %	Carbon/Glas/Epoxy	37 %				Flugtest	332,
L-1011	Inneres Querruder	75 %	Kevlar/Epoxy-Haut, Alu-Wabe Carbon/Kevlar/Epoxy-Holm	28 %	61 %	- 20 %		Entwicklung	332,
DC-10	Aft Pylon Skin	100 %	Bor/Al	26 %					332,
C-130	Flügelmittelkasten		Bor/Epoxy Versteifung von Haut und Stringer Hybridbauweise	120 kg 10 %		teurer		im Einsatz	332,
L-1011	Seitenleitwerk	83 %	Carbon/Kevlar/Epoxy	25 %				Entwicklung	332,
Trident (SBLM)	Konusteil		Carbon/Epoxy Gewebe + Carbon/Epoxy Tape	20 %		- 50 %		Serie	
Space-Shuttle	Türen		Carbon/Epoxy-Haut, Alu-Wabe Carbon/Epoxy-Versteifungsprofile	23 %				Bodentest	787
F-111	Schwenklager-Flügelbeschlag		Bor/Epoxy Verstärkung aufgeklebt auf Stahlkonstruktion	Metall 6,8 kp BFK 1,8 kp			175		551,
CH-54 B	Heckkonus	8 %	Bor/Epoxy Versteifung von Stringer Hybridbauweise	57 kg 14 %				im Einsatz	332,

Fig 11.11 (Table 5 concluded)

Key to Figs 11.9 to 11.11 (Table 5)

Gew.Ant. Faserverbund	= weight fraction fibre composite
Faserverbundkonstruktion Werkstoffe/ Bauweisen	= fibre composite construction, materials/ construction methods
Gewichtsersparnis	= weight saving
Reduktion der Einzelteile	= reduction in individual parts
Stückzahl	= number of pieces
Stand der Entwicklung	= state of development
Haut	= skin
Alu-Waben mit Metallkanten und Beschläge	= Alu honeycomb with metal edges and fittings
Mehrk. (Ersatzkonstruktion)	= extra cost (replacement construction)
Glasfaser-Gewebe	= glass fibre cloth
Erprobung/Versuche	= trials/tests
Landeklappen	= landing flaps
Metall-Rippen	= metal ribs
Fliegerprob.	= flight test
Einteilig	= one piece
Formgepresst	= compression moulded
Verklebt	= bonded
Einsatz	= in use
Ruder (Seitenleitwerk)	= rudder (vertical tail surface)
Bremsklappe	= air brake
Versteifung	= reinforcement
Höhenleitwerk	= horizontal tail surfaces
Anschluss: Stahl/Titan-Unterstruktur/ Titan-Platte mit Bor/Ep. Haut verklebt	= junction: steel/titanium substructure/ titanium plate bonded to boron/ep. skin
Glas/Epoxy-Kanten, Bolzen	= glass/epoxy edges, bolts
Geklebt + genietet	= bonded + riveted
Tragfläche	= lifting surface
Teurer	= dearer
Rumpfvorderteil	= fuselage front section
Zwischenwände	= partitions
Mit gestuftem Titan Pivot-Beschlag	= with stepped titanium pivot armature
Vorder- u. Hinterkante	= front and rear edge
Rumpfteil, hinten (zwischen den Triebwerken)	= fuselage section, rear (between engines)
Spant	= frame
Gebaut zur Entwicklung d. Technologie, stat. Versuch	= built to develop technology, static test
Rumpfteil, vorn	= fuselage section, front
Umhüllung	= envelope
Rumpfholme	= fuselage longeron
Leitwerk	= tail unit
Sonst vernietet mit Alu-Unterstruktur Klebung und Neitung	= otherwise riveted with aluminium sub- structure, bonded and riveted
Zwischenlagen	= intermediate layer
Flügel-Aussenteil	= wing external part
Im Bereich	= in the area
von Bohrungen	= of holes
Stege (teilweise)	= straps (partly)
Rahmen (Holme, Scharnier-Rippe)	= frame (spars, hinge-rib)
Dichtmittel für Treibstoff	= jointing medium for fuel
Sinusförmige Stege von Holm und Rippen	= sine wave webs for spars

/Cont.

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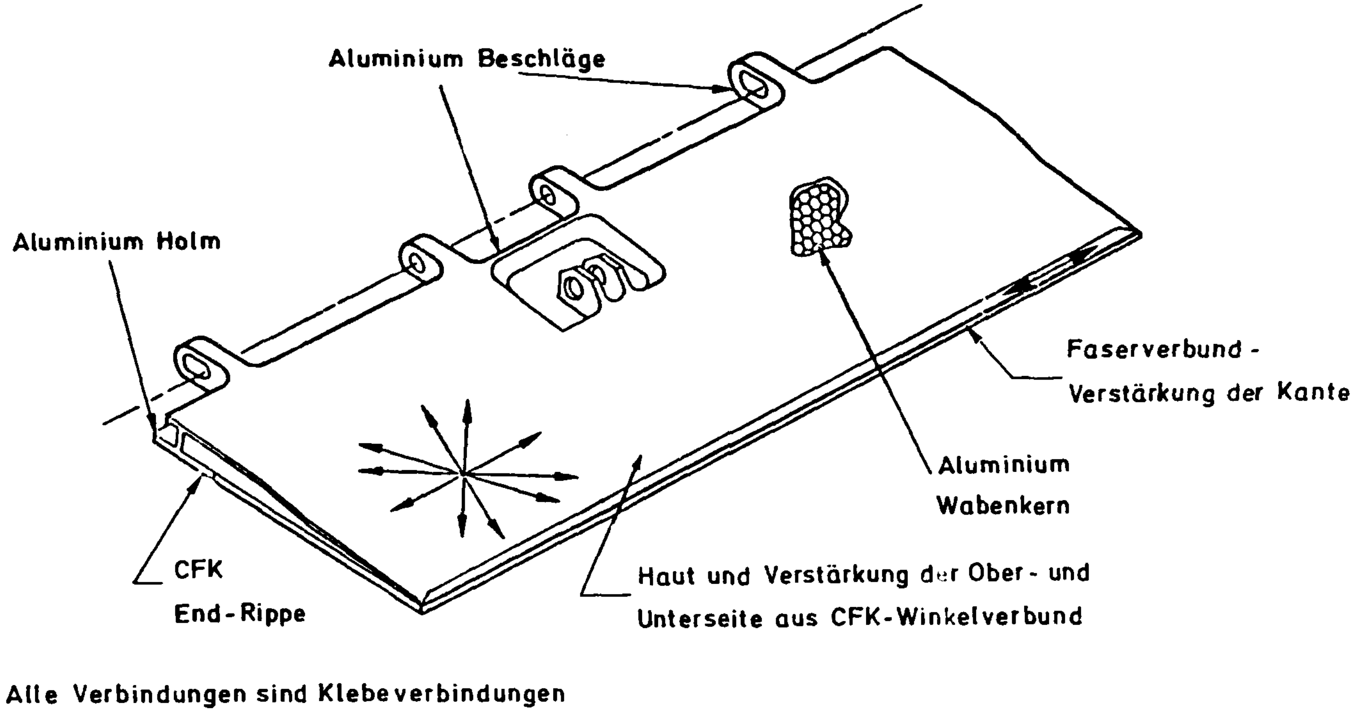
Key to Figs 11.9 to 11.11 (Table 5) (concluded)

Flügelvorderkante	= wing leading edge
Klebung Klarsicht Epoxy Anstrich	= bonding transparent epoxy coating
Von 800 auf weniger als 80	= from 800 to less than 80
Inneres Querruder	= inner aileron
Flügelmittelkasten	= wing centre section
Teurer	= doors
Schwenklager-Flügelbeschlag	= swing bearing wing fitting
Verstärkung aufgeklebt auf Stahlkonstruktion	= reinforcement bonded to steel construction
Heckkonus	= tail cone

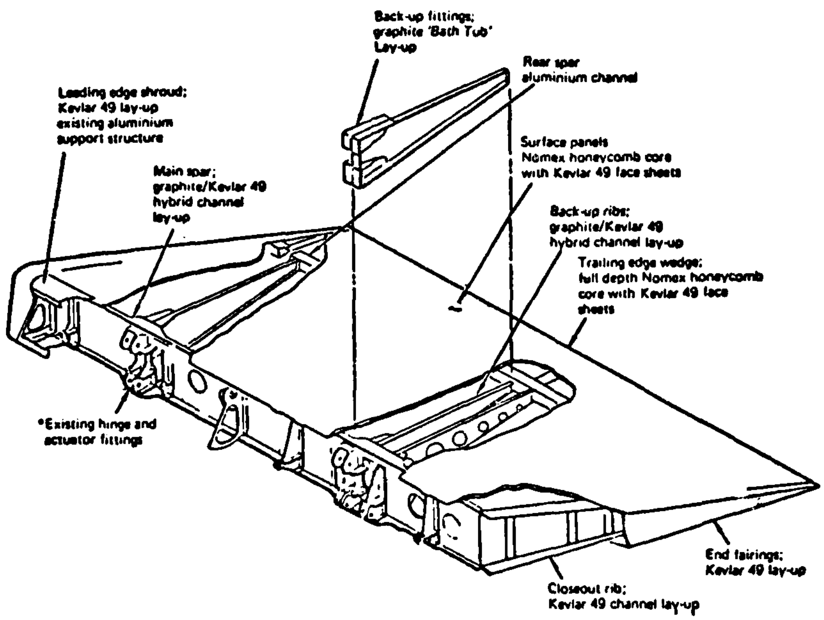


Key:

Aluminium Beschläge	= aluminium fitting
End-Rippe	= end rib
Faserverbund-Verstärkung der Kante	= fibre composite reinforcement of edge
Aluminium Wabenkern	= aluminium honeycomb core
Haut und Verstärkung der Ober- und Underseite aus CFK-Winkelverbund	= skin and reinforcement of upper and lower side of CFC angle ply composite
Alle Verbindungen sind Klebeverbindungen	= all junctions are adhesive

Fig 11.12 B-737 CFC flight spoiler

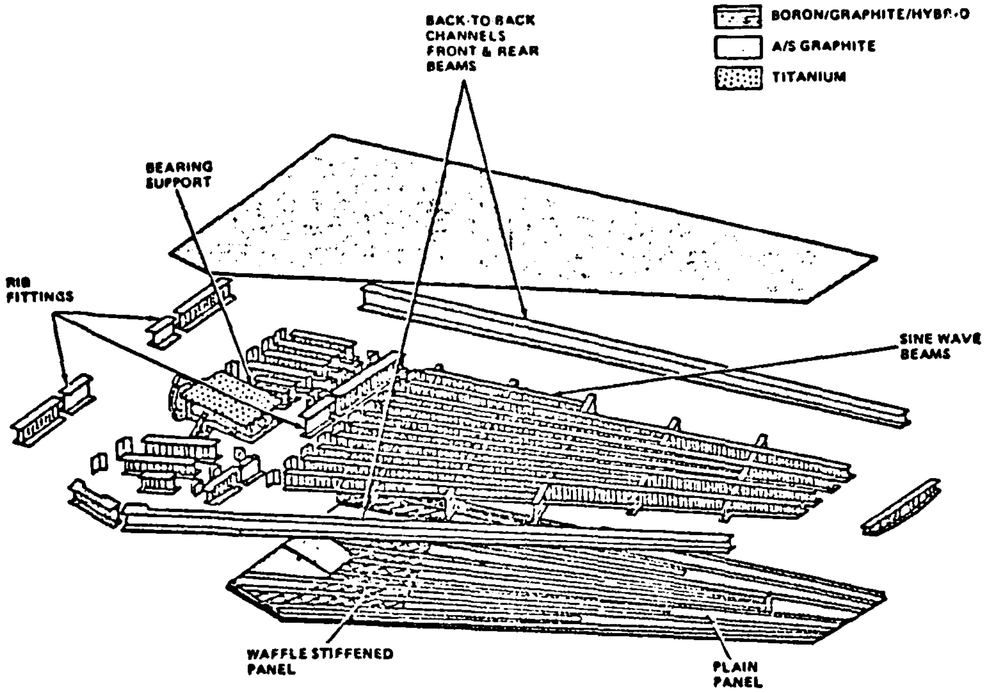
Typical fibre composite construction for thin components (secondary component)



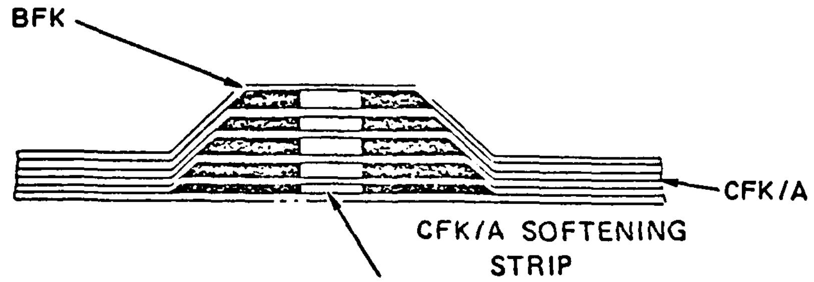
Entwurf eines Faserverbund-Querruders für die L - 1011 [332]

Key:
Entwurf eines Faserverbund-Querruders = design for a fibre composite aileron
für die L-1011 = for the L-1011

..: 11.13 Fibre composite component with spar-rib substructure
(secondary component, thick)



Faserverbund-Höhenleitwerks-Torsionskasten / 613 /



Bohrungen in unterer Beplankung / 343 /

Key:
Faserverbund-Höhenleitwerks-Torsionskasten = fibre composite horizontal tail surface torsion box
Bohrungen in unterer Beplankung = holes in lower wing skin

Fig 11.14 Fibre composite construction in the primary structure of the B-1 horizontal tail surfaces

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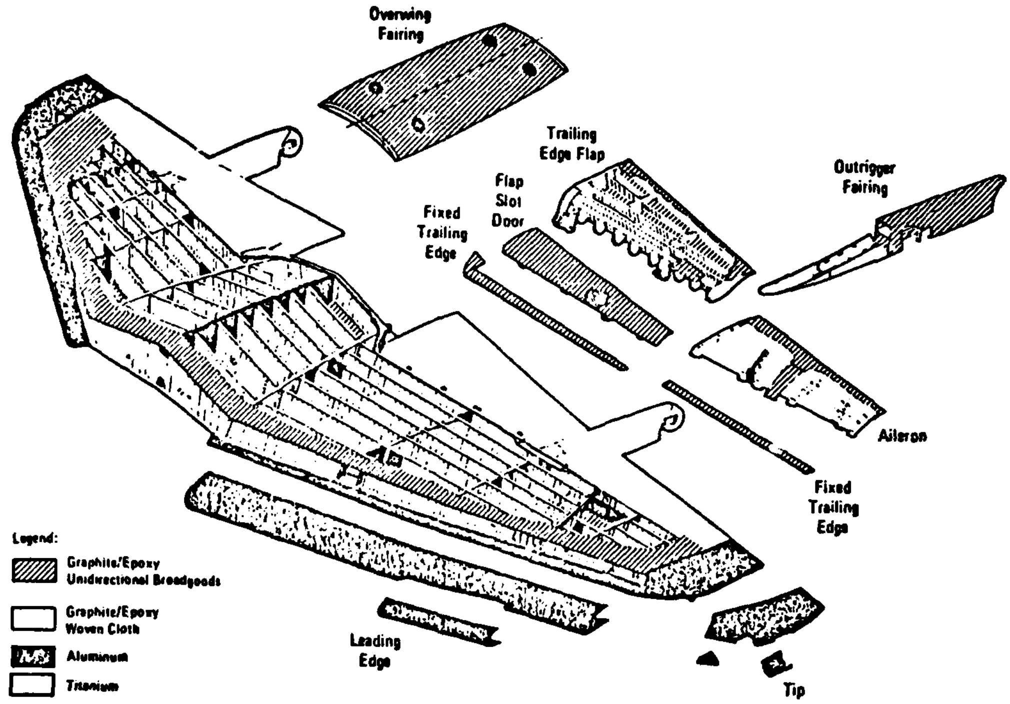

Fig 11.15 Fibre composite construction in the primary structure of the AV-8B wing (Ref 614)

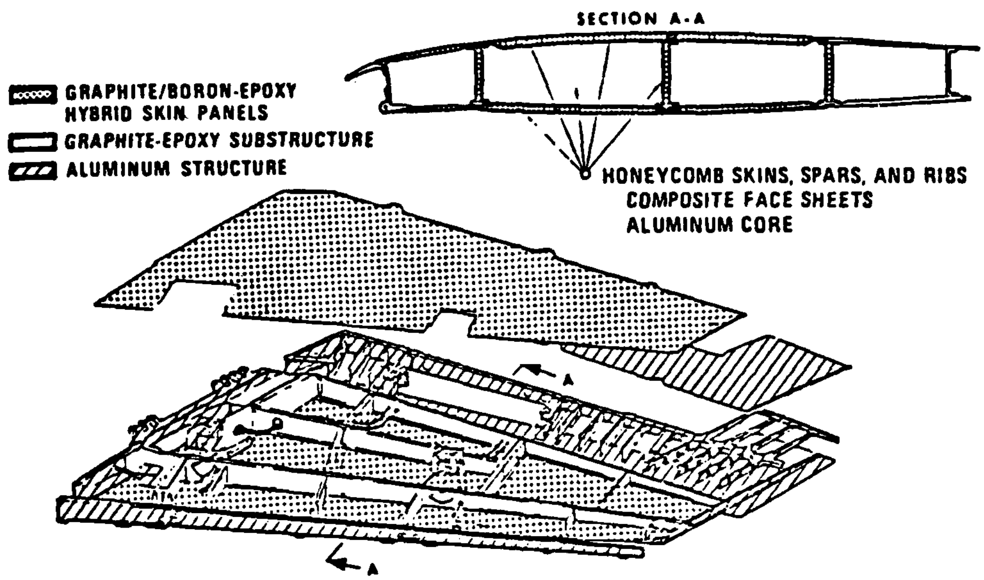

Fig 11.16 Fibre composite construction in the primary structure of the A-7D wing (Ref 553)

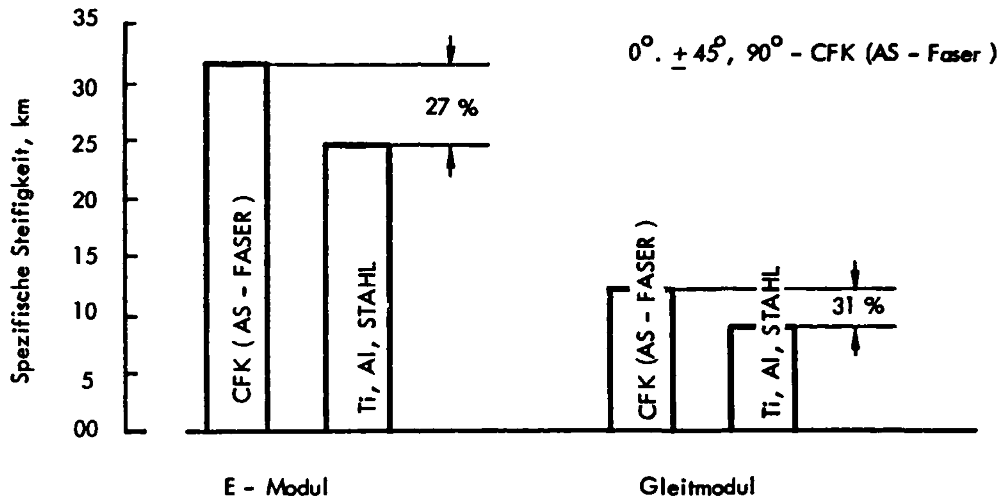

Fig 11.17 Comparison of specific stiffness for a balanced CFC angle ply composite and metals (Ref 789)

Fig 11.18

PRICES OF FIBRE COMPOSITE MATERIALS AS AT END 1978

Boron-Epoxy	≈	1.000 DM/kg
Carbon-Epoxy (HT)	≈	300 DM/kg
CFC cloth	≈	440-500 DM/kg
Bor/Aluminium	≈	150 DM/kg
Carbon HT-fibre	≈	175 DM/kg
Carbon A-fibre	≈	150 DM/kg
Carbon P-fibre	≈	100 DM/kg
Kevlar 49-fibre	≈	65 DM/kg

12 CONCLUSIONS AND OUTLOOK

J. J. Gerharz

Contents

12.1 Introduction

The literature search on the strength properties of fibre composite materials has yielded the following conclusions in regard to their use as structural materials for aircraft construction and appropriate recommendations for future research.

12.2 Conclusions

- Fibre composites differ from metals in the areas of:
  - manufacture
  - material behaviour
  - damage development
  - fracture mechanism and
  - reaction to environmental effects.

  In comparison with metal this requires different techniques and methods for:
  - testing
  - inspection
  - design and
  - theoretical treatment of the above areas,

  while special mention should be made of the fact that methods of analysis are required which take the anisotropy of the composite material into account.
- Present design criteria, partially taken over from methods for metal construction, are unsuitable for fibre composites because
  - they do not allow the material properties of fibre composites to be fully exploited,
  - they do not allow for material, design concepts, manufacturing methods and scatter of load and environmental conditions and
  - they permit translation of residual strengths and permissible life values from the material coupon to the component, which is not permissible in fibre composite design methods.
- Residual strengths and properties under creep and fatigue loads should be determined in fibre composite components, particularly joints, and plates with holes and other structural notches by testing under load/environment-time conditions appropriate to service usage.

- Further work is required in development, experimental testing and the inclusion of reliability concepts in practical design procedures.
- Structural notches have great influence on the static strength of fibre composites; this is in contrast to the behaviour of most metal materials. Therefore analytical methods and accompanying experimental techniques should be developed for the design of joints, cutouts and changes in section.

12.3 Recommended subjects for research

The subjects for research listed below refer primarily to the creation and preparation of design material in the form of data and calculating methods. These are most urgently required for the fibre composites already in serial production:

GFC
CFC
BFC
Boron aluminium and
KFC (kevlar/epoxide resin)

The order in which these materials are listed corresponds to the volume of design data available, *ie* there are most for GFC and fewest for KFC.

On the introduction of new constructional materials the stage at which design data are generated is preceded by other development stages whose subjects are not listed here. For instance, the fibre composites with a polyamide matrix or a metal matrix (except boron-aluminium) are in these early development stages. Newly developed fibre composites and their possible applications were mentioned in section 11.

In the list which follows of research work required, each individual subject is marked with the letters S, A or L to indicate the following:

S - Work on the subject is still in progress, but is approaching saturation.

A - Work on the subject is in progress and the existing gaps are still wide; further research is still required at present.

L - The subject is recognised as a research project but has not yet been dealt with intensively.

This valuation of research subjects refers only to CFC, which is the fibre composite used to the greatest extent at present. Since the matrix material of the named fibre composites with plastic matrix is universally epoxide resin, many of the research results can be transferred to materials with different fibres if the special properties of the fibre, the quality of the fibre/matrix bond and the arrangement of the fibres in the matrix are taken into account. For boron-aluminium composites further investigations are needed into residual strength, fatigue strength and fracture behaviour under suitable load/environmental conditions. However, the analysis methods of metals are suitable in certain areas for metal matrix composites. The methods of fracture mechanics are an example of this.

The following research subjects are recommended in regard to the use of CFC fibre composite materials in the construction of air frames:

(a) Load and environmental data recorded in various aircraft should be sifted and evaluated for the specific purposes of reliability considerations. This yields the following separate subjects:

A - collection of data on moisture absorption,

S - theory and calculation of moisture absorption,

A - investigations of simplifying changes in the load/environment programme,

A - derivation of a method to take into account scatter of load and environment in a suitable model for proof of reliability,

L - investigations into the relationship between residual stresses due to manufacture and environment and the load sequence.

(b) Methods for design of composite structure (fibre orientation and arrangement of layers), joints, cutouts and other structurally necessary notches for static strength, stiffness, residual strength and fatigue strength should be developed and tested. Appropriate design data should be set up, taking account of random loading and environmental conditions. This results in the following subjects:

A - investigation of environmental effects,

A - standardisation of load/environment sequences, taking account of the interaction of load, temperature change and humidity cycles,

S - development of test methods for joints,

L - investigation of failure criteria for joints,

L - effects of manufacturing and design tolerances in joints, A – investigation into the effect of surface notches,

L – investigation of the design of simple specimens, structural members, structural components to determine the applicability of results from coupon specimen tests to the structure (edge effects),

L – investigations of stress concentrations under biaxial loads,

L – behaviour after the failure of one layer in the composite,

A – investigations into the effect of changes in thickness and width,

L – effects of size and the incidence of defects on life,

A – stability investigations on reinforced plates and elements of wing units,

L – behaviour of composites under biaxial load (tests with tube specimens, 'off-axis' specimens, cross specimens),

A – behaviour of mechanical joints, bonded joints and plates with cutouts. Parameters to be varied are the geometry of the notch, introduction of load at the notch, and laminate lay-up,

A – effect of local reinforcements and special measures to reduce stress peaks and of further joint parameters on behaviour under high load transfer.

(c) Development of suitable methods of prediction

- of failure of the composite based on characteristics of the individual layer,
- of fatigue strength and deformation behaviour,
- of residual strength,
- of endurance.

The following explanation is required on the research subjects in this area:

In methods of prediction the applicability of the behaviour of the individual layer to the laminate plays an important part. So far it is not possible to predict or explain all the mechanical properties of angle ply composites solely from test results of the uni-directional individual layer; probably this will not be possible in the future either. In developing methods of prediction tests must also be performed on angle ply composites. On the other hand, data on the behaviour of the uni-directional layer also contribute to an understanding of interrelationships, *ie* tests on uni-directional composites are useful for this reason too.

In detail the research subjects are:

L - investigation of strength through the thickness of the laminate (characteristics of the transverse direction are generally used here. This procedure is frequently not admissible since in most layers distribution of the fibres in the matrix in a transverse direction is different from that in the direction of thickness),

A - investigations on the behaviour of the composite under interlaminar shear in monotonic and repeated loading (important for valuation of edge delamination and load transfer in the composite),

A - fracture hypotheses for fibre composites, including interlaminar stresses (in the macromechanical range) and micromechanical stresses,

L - extension of the fracture hypotheses for holes to non-circular cutouts,

L - fracture hypotheses for bonded joints, taking account of geometry and effect of size,

A - failure models for the individual layer based on statistical methods,

L - relationship between individual forms of failure,

L - establishment of damage criteria,

A - creation of statistically backed data on residual strength and endurance for various laminate families (see Handbook Fibre Composite Light Construction),

S - investigation of the effect of fatigue loading with random load sequence (flight-by-flight tests) on the drop in strength and stiffness,

L - generation of relationships between composite damage or types of fracture and endurance,

A - investigation of applicability of residual and fatigue strength values for the individual layer to the residual and fatigue strength of a composite, A - development of a damage accumulation hypothesis, taking account of results of investigations on environmental effects; here its applicability to joints and components with other structurally necessary notches should also be observed.

(d) Investigation of the durability of the fibre composite structure in creep and after damage by the impact of hard and soft objects. Investigation of the following subjects is recommended for this purpose:

L - creep strain tests on adhesive joints,

L - effect of moisture in creep strain test,

A - damage tolerance towards hits and strikes,

L - investigation of methods of construction to increase damage tolerance,

A - effect of damage already detected, but not eliminated, on life,

A - effect of UV radiation, rain erosion, lightning strikes and radioactive radiation and investigation of protective measures.

(e) Investigations to improve the non-destructive test and inspection methods already in use.

The following detailed research subjects are recommended:

A - checking the chemical composition of the resin to ensure quality maintenance,

A - monitoring and improvement of the curing process,

A - testing the resin for resistance to environmental effects,

A - development or improvement of non-destructive inspection methods at holes and edges,

L - proof-load concepts.

The areas of research to demonstrate

- static strength,
- fatigue strength,
- residual strength and
- stability

of composite materials are shown in a 'structural diagram' in Fig 12.1.

In this the research areas are sub-divided into the following main subjects:

- stresses inherent in operation,
- stresses inherent in construction in composite, component, structure,
- effect of typical material properties,
- damage mechanics, behaviour of component material in damaged condition,
- damage control to demonstrate and monitor reliability of the structure and
- tests in regard to requisite test techniques and appropriate standardisation.

The research subjects listed earlier can be allocated to at least one of these areas of research. Examples are the terms in the 'structural diagram' which are not in boxes.

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LITERATURE RESEARCH ON THE MECHANICAL PROPERTIES OF FIBRE COMPO--ETC(U)
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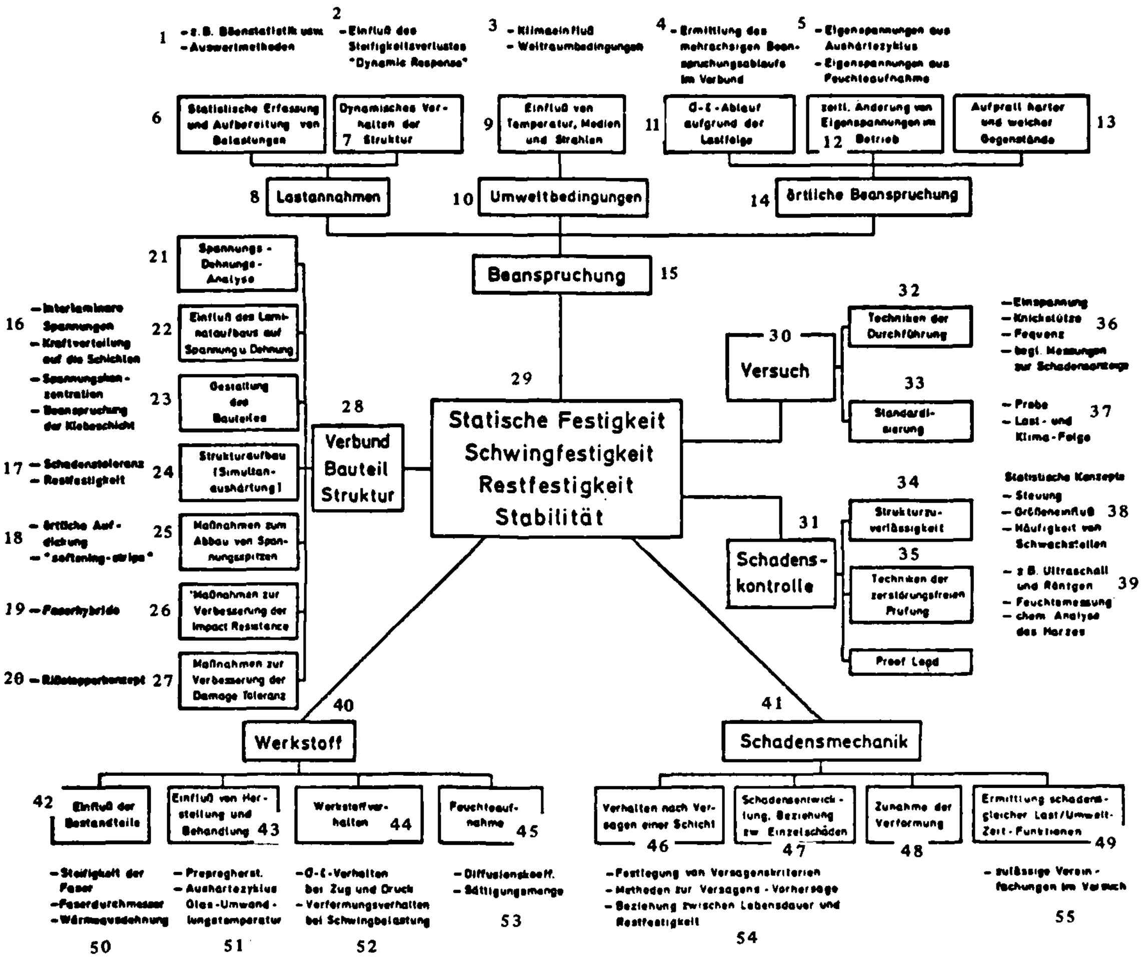

Fig 12.1 Areas of research into static strength, fatigue strength, residual strength and stability of fibre composite materials

Key to Fig 12.1

1 = *eg* gust statistics, etc
evaluation methods
2 = effect of loss of stiffness
3 = effect of climate
space conditions
4 = determination of multiaxial load sequence in composite
5 = residual stresses from curing cycle
residual stresses from moisture absorption
6 = statistical measurement and presentation of loads
7 = dynamic response of the structure
8 = load assumptions
9 = effect of temperature, media and radiation
10 = environmental conditions
11 = $\sigma - \varepsilon$ sequence due to loading
12 = change with time of residual stresses in operations
13 = impact of hard and soft objects
14 = local stress
15 = stress
16 = interlaminar stresses
force distribution in the layers
stress concentration
load on adhesive layer
17 = damage tolerance
residual strength
18 = local thickening
19 = fibre hybrids
20 = crack stopper concept
21 = stress-strain analysis
22 = effect of laminate structure on stress and strain
23 = design of the component
24 = structural design (simultaneous curing)
25 = measures to reduce stress peaks
26 = measures to improve impact resistance
27 = measures to improve damage tolerance
28 = composite, component, structure
29 = static strength, fatigue strength, residual strength, stability
30 = test
31 = damage control
32 = performance techniques
33 = standardisation
34 = reliability of structure
35 = NDT techniques
36 = clamping
buckling restraint
frequency
accompanying measurements for damage indication
37 = specimen
load and climatic sequence
38 = statistical concepts
scatter
effect of size
incidence of defects
39 = eg ultrasonics and radiography
moisture measurement
chemical analysis of resin
40 = material
41 = damage mechanism
42 = effect of constituents
43 = effect of manufacture and treatment
44 = material behaviour
45 = moisture absorption
46 = behaviour after failure of a layer
47 = damage development. *Relationship* between individual defects
48 = increase in deformation
49 = determination of load/environment-time conditions giving equal damage
50 = stiffness of fibre
fibre diameter
thermal expansion
51 = prepreg production
curing cycle glass equilibrium temperature
52 = $\sigma - \varepsilon$ behaviour under tension and compression
deformation behaviour under fatigue load
53 = diffusion coefficient
saturation volume
54 = specifying failure criteria
methods of failure prediction
relationship between life and residual strength
55 = permissible simplifications in tests

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