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Endurance of fibre-reinforced composite, laminated structural elements subjected to simulated random acoustic loading



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THE PREPARATION OF THIS DATA ITEM

The work of the permanent professional staff of the Engineering Sciences Data Unit on this particular Data Item was monitored and guided by the Acoustic Fatigue Panel which had the following constitution:

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The Vibration and Structural Dynamics Steering Group, which took over the work on Acoustic Fatigue previously monitored by the Fatigue Committee, first met in July 1976 and has the co-operation of many engineers and scientists in industry, research establishments and universities from whom much assistance and information is being received.

ENDURANCE OF FIBRE-REINFORCED COMPOSITE, LAMINATED STRUCTURAL ELEMENTS SUBJECTED TO SIMULATED RANDOM ACOUSTIC LOADING

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ENDURANCE OF FIBRE-REINFORCED COMPOSITE, LAMINATED STRUCTURAL ELEMENTS SUBJECTED TO SIMULATED RANDOM ACOUSTIC LOADING

1. NOTATION

		Units	
		SI	British
N_r	equivalent endurance, taken as half number of zero crossings to failure of strain-time function	cycles	cycles
ϵ_{rms}	root mean square values of micro-strain at a reference position		

2. NOTES

2.1 General

This Item gives the results of fatigue tests on fibre-reinforced composite specimens, typical of aircraft structural elements, excited using band-limited excitation to produce random amplitude responses, with zero mean value, to simulate response in a resonant mode to acoustic loading. Data for laminated structural elements in which all laminae are fibre-reinforced plastic and those having a mix of fibre-reinforced plastic and metal laminae (fibre metal laminates) are presented. For some of the fibre metal laminates, data are presented for three operating temperatures. Some results are presented on ceramic matrix composites and include results at different temperatures. The types of structural element tested are illustrated in Figures 1 to 10. The data presented in this Item are intended to be used only for guidance since, in some cases, unusual configurations (*e.g.* the specimens with a syntactic[†] foam core) were tested and not all relevant test data were reported. Endurance data are specific to the types of element tested, the method of testing and the method of failure identification. The testing methods (see Appendix A) used to obtain the data presented were not consistent. A description of the specimens and test techniques commonly used to determine the endurance of structural elements under simulated acoustic loading is given in ESDU 93027[‡].

The reference position for ϵ_{rms} is taken as the location on the outer surface layer of the laminate where fatigue failures occurred. The monitoring strain gauges were aligned in the zero degree fibre direction (shown in Figures 1 to 10) to measure bending strains. The test points plotted represent either mean strains over the areas covered by monitoring strain gauges affixed across the failure line, away from any riveted attachments, or strain at that position estimated after calibration from strain monitors located elsewhere. Data are presented in terms of root mean-squared strains, rather than root mean-squared stresses, because it is reasonable to assume a linear variation of strain through a laminate thickness whereas stress varies from layer-to-layer depending on the value of the layer Young's modulus. Stresses deduced from strain gauges mounted on the surface of fibre-reinforced laminates will vary depending on the relative orientations of the strain gauge and the principal fibre direction in the surface lamina. When using this Item to estimate the life of a stiffened panel based on the root mean-squared strain obtained from ESDU 84008^{††}, it is recommended that the plate edge strain should be assumed to be the same as that in the skin, adjacent to the joint at the failure position. In practice the skin failure position at a riveted or bonded stiffener is

[†] A cellular plastic which is "put together" by incorporating preformed cells (hollow spheres or micro balloons) in a resin matrix as opposed to "foamed plastic" in which the cells are formed by gas bubbles released in a liquid plastic by either chemical or mechanical action.

[‡] ESDU 93027 "Methods of testing for endurance of structural elements using simulated acoustic loading".

^{††} ESDU84008 "Estimation of r.m.s. strain in laminated skin panels subjected to random acoustic loading".

sufficiently close to the plate edge, unless doublers of substantial width are bonded to the skin at the stiffener location, for this approximation to be within the range of accuracy of the simple theory used for strain prediction in ESDU84008.

With fibre-reinforced laminates there are more modes of failure (*e.g.* delamination) than are possible with conventional metallic structures. However, it has been shown that simulated acoustic loading tests on structural elements, such as those used to provide data for this Item, produce not only modes of failure but also random fatigue lives that correspond to full-scale structural testing under acoustic loading. The methods of failure detection in the different data sources used in this Item are noted in the Appendix (see below). Failure is characterised by a reduction in the resonant frequency accompanied by a fall in response amplitude and an increase in damping. These symptoms may also be accompanied by visible failure such as progressive surface fibre failures or the initiation of surface cracks.

In Appendix A, details of the test methods used to obtain the results presented in this Item are given. The method of failure detection used and the failure locations are also described in Appendix A.

The loading simulated for the skin-stiffener specimens is only that for modes in which adjacent panels are in-phase. Care should be taken in using the data for modes involving twisting of the stiffeners.

These endurance data should not be used in isolation to compare the relative acoustic fatigue resistances of different joint configurations. The coupon tests show fatigue lives for identified root mean-squared strain levels. In practice, different strain levels will be induced, for a given noise environment, with different structural configurations.

2.2 Material Descriptions

2.2.1 Fibre-reinforced laminates

For specimens that are constructed from fibre-reinforced plastic material only, the materials are fully described by the fibre and matrix types and the fibre orientations within the laminate lay-up. The fibre and matrix materials of the failed component are specified in Table 5.1 and the lamina fibre orientations are given, using standard nomenclature, on the relevant figure. The 0° fibre directions are given on the figure illustrating the specimen type.

Two additional composites are included, namely, IM7/PETI-5 and IM7/K3B (Derivation 13).

IM7/PETI-5 is a carbon-fibre polymeric composite in which intermediate modulus carbon fibres are embedded in a matrix resin, PETI-5. PETI-5 is a phenyl ethynyl terminated imide oligomer of the fifth composition. It has been developed for applications in high speed aircraft. The elastic constants for the IM7/PETI-5 composite laminate are: $E_1 = 151.7$ GPa, $E_2 = 9.65$ GPa, $G_{12} = 4.14$ GPa, $\nu_{12} = 0.33$ and $\rho = 1580$ kg/m³. The ultimate tensile stress is probably between 0.8 and 2.93 GPa (Derivation 14). It has excellent fracture performance in terms of crack initiation and growth behaviour. The glass transition temperature of the resin is 270°C.

IM7/K3B has the same carbon fibres, but in this case the fibres are embedded in a thermoplastic polyimide resin, K3B. The elastic constants for the IM7/K3B composite laminate are: $E_1 = 179$ GPa, $E_2 = 9.45$ GPa, $G_{12} = 7.93$ GPa, $\nu_{12} = 0.29$ and $\nu_{21} = 0.015$. The longitudinal tensile strength is 3.06 GPa. The glass transition temperature of 240°C indicates that it may be a candidate for high temperature applications (Derivation 12).

2.2.2 Fibre metal laminates

For fibre metal laminates it is necessary to describe the metallic laminae material and thickness as well as the fibre and matrix materials. In this Item, data are provided for fatigue failures on four types of glass/aluminium reinforced laminates. These materials are referred to as GLARE composites. The make-up of these laminates is defined in the following table. In the fibre orientation column, which defines the lay-up sequence, an aluminium alloy layer is indicated by the symbol Al .

Laminate [†]	Test code [‡]	Metallic layer		No. of layers	Fibre orientation ^{††}
		Material	thickness mm		
GLARE 3-2/1-0.3	GLARE - A	2024	0.3	4	$Al, 0, 90, Al$
GLARE 3-3/2-0.2	GLARE - C	2024	0.2	7	$(Al, 0, 90, \underline{Al})_s$
GLARE 5F2-2/1-0.3	GLARE - E	2024	0.3	6	$(Al, 0, 90)_s$
GLARE 1-3/2-0.3	GLARE - G	7475	0.3	7	$(Al, 0, 90, \underline{Al})_s$
GLARE 2		2324-T3	0.3	5	$(Al, 0, \underline{Al})_s$
GLARE 3		2324-T3	0.3	5	$(Al, 0/90, \underline{Al})_s^{‡‡}$

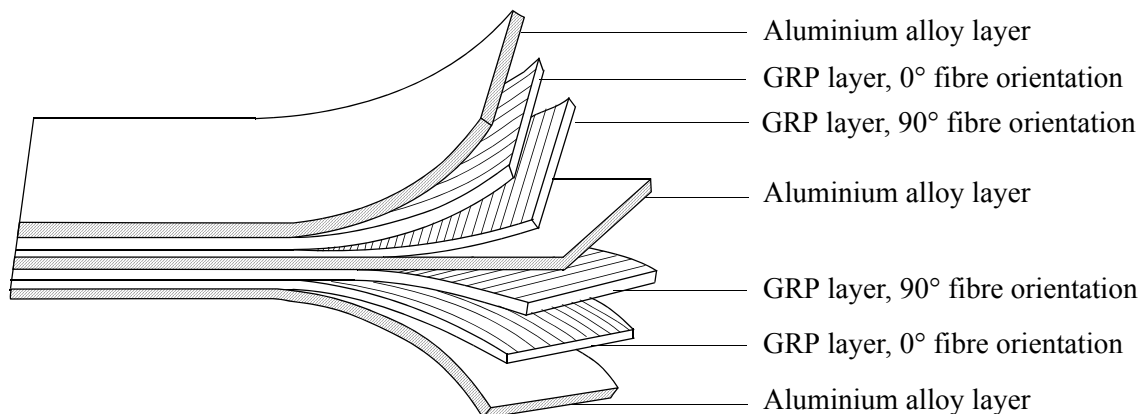
[†] This is the Notation used by the supplier (Structural Laminates Industries bv, 2629 HS Delft, The Netherlands).

[‡] This is the Notation used in the test report, Reference 9.

^{††} An underline indicates the central layer within a symmetric laminate having an odd number of layers.

^{‡‡} 0/90 indicates cross-ply.

Sketch 2.1 illustrates the laminate lay-up of GLARE 3-3/2-0.2. The materials GLARE 1-3/2-0.3 and GLARE 3 are included in the tabulation since stiffeners were manufactured from those materials; however, all failures occurred in the skins.



(GRP = glass reinforced plastic)

Sketch 2.1 GLARE 3-3/2-0.2 laminate lay-up details

2.2.3 Ceramic matrix laminates

Ceramic matrix composites (CMC) can include carbon fibres in a carbon matrix (C/C), carbon fibres in a silicon carbide matrix (C/SiC) or Silicon carbide fibres in a silicon carbide matrix (SiC/SiC). The fibres may also be contained in an oxide based matrix, such as aluminium oxide or aluminium phosphate.

Data are presented for silicon carbide cloth, coated with boron nitride, in a mono aluminium phosphate matrix (BN-SiC/MAP). The SiC fibres were described as high volume resistivity Nicalon and were chosen for their low dielectric constant. The boron nitride coating was applied to the silicon carbide to enhance composite toughness. Tests were carried out on coupons formed from skins composed of seven plies, zero and 90° and the coupons that had an average thickness of 0.122 in (Derivation 8).

Separate tests were undertaken on a CMC material referred to as BlackglasTM, a trade name for a silicon oxide carbide (Derivation 11). The specimens in the form of cantilevered beams had a thickness of 2.032 mm (0.08 in).

2.3 Data Presented

Endurance data are presented for composite, laminated structural elements either having all layers of fibre-reinforced materials or having a combination of fibre-reinforced layers and metallic layers. In this Item the former are referred to as fibre-reinforced specimens and the latter as fibre metal specimens. A summary of the data presented is provided in Tables 5.1 and 5.2 for the fibre-reinforced elements and fibre metal elements respectively. Table 5.3 includes information on ceramic matrix composites. See Appendix A for test details and failure identification.

2.3.1 Fibre-reinforced structural elements

The plain specimens, illustrated in Figures 1a and 1b, are representative of laminated plate material away from the effects of attachments. The specimens are uni-directional laminates. The built-up specimens, illustrated in Figures 2 to 4, represent conditions at structural joints where adjacent plates are vibrating in-phase. The Y-section shown attached to the beaded skin specimen in Figure 4 is used to drive the structural element during the test. The direction of excitation and zero degree fibre direction are defined on the figures showing the specimen types.

In Figures 11 to 23, ϵ_{rms} is plotted against N_r for test results grouped according to the type of structural element, skin/stiffener material and method of attachment, as shown in Table 5.1. Due to the scatter in the test data, no line (see Section 2.3.4) is drawn through the data presented in Figure 19.

The IM7/PETI-5 and IM7/K3B composites were tested as cantilevered coupons, as shown in Figure 5. The method of testing is described in Appendix A11.2.

Rms strain is shown plotted against the number of cycles to failure in Figure 24 for results at room temperature and at 350°F (176.5°C). At 10^7 cycles the strains at room temperature and at 350°F are of the same order of magnitude.

A more realistic joint, made from the IM7/PETI-5 material and shown in Figure 6, was tested at room temperature and the results included in Figure 24. Failure for the joint occurs at much lower strains than for the coupon.

For the IM7/K3B composite the rms strain versus the number of cycles to failure is given in Figure 25 for results at room temperature and at 350°F (176.5°C).

2.3.2 Fibre metal structural elements

The fibre metal plain specimens, illustrated in Figure 7, are representative of a laminated skin in the presence of a discontinuity in the thickness. For these specimens data are provided for different laminate lay-ups and a range of temperatures. The rib-mode specimen, shown in Figure 9, provides data on the joint fatigue strength when the stringer is in resonance. The built-up specimens, illustrated in both Figures 8 and 9, represent conditions at structural joints where adjacent panels are vibrating in-phase.

In Figures 26 to 33, ϵ_{rms} is plotted against N_r for fibre metal test results grouped according to the type of structural element and skin material, as shown in Table 5.2. The results described in Figures 26 to 32 are based on Derivation 9 and the results of Figure 33 on Derivation 10.

2.3.3 Ceramic matrix composite elements

Results for the BN-SiC/MAP CMC are shown in Figure 34 for ambient temperature, 1000°F (537.8°C) and 1600°F (871.1°C). The results for the Blackglas™ are shown in Figure 35 for the 2% and 5% decrease in the natural frequency as a characteristic of failure.

2.3.4 Lines fitted through test data

The straight lines drawn through the test data are fitted by the method of least squares. These lines are intended for guidance; they should not be used directly for estimating life because due account must be taken of scatter. Caution should be exercised when estimating endurance from the extrapolated sections of these lines where no data are available. This is particularly so for the fibre metal GLARE plots where few data are available beyond 10^7 cycles. Extrapolation of the lines is expected to lead to an underestimate of endurance.

To allow a comparison of endurance data at different test temperatures, the lines drawn through the fibre metal data (Figures 26 to 28) have been given a common slope. The slope is the weighted[†] mean slope of the lines fitted through the endurance data for each test temperature. A similar approach has been used for the fibre metal data for riveted and bonded test specimen shown in Figures 29 to 31.

In Appendix B the equations of the regression lines are given for the results presented in Amendment D.

3. INTERPRETATION OF THE RESULTS

The endurance data presented in Figures 11 to 35 are based on a limited number of test measurements made on structural elements of particular laminate construction. The confidence levels associated with life estimation based on these data are therefore likely to be significantly lower than those for life prediction of metallic structural elements. Factors that may affect the endurance of fibre-reinforced composite structural elements are discussed below. Account should be taken of these factors in interpreting the results presented in this Item.

[†] Weighted mean slope =
$$\frac{\sum (\text{slope} \times \text{number of test points})}{\text{total number of test points}}$$

3.1 Effect of Laminate Layering Sequence

The effect of laminate layering sequence is minimised by presenting endurance data related to surface root mean-squared strain rather than root mean-squared stress. The data for plain skin specimens, in Figure 11, do not indicate a systematic variation with the layering sequence or, with the exception of the four ply laminate, any effect due to a variation in the number of plies. The four ply laminate specimens[†] show unusually high strain levels without failure. The relatively thin, four ply laminates did not respond linearly with increasing loading so that the relationship between the skin surface strain and the monitored strain, on the clamping fixture, may have been adversely affected.

3.2 Effect of Fibre Material

From the limited test data available it is not possible to distinguish the effects of differing carbon fibre materials. However, there is a marked difference in endurance between laminates with boron fibres and those with carbon fibres. From Figures 14 and 16, for a given root mean-squared strain level, the life of bonded skin specimens with carbon fibre is much greater than that for boron fibre specimens.

3.3 Effect of Matrix Material

A comparison of Figures 17 and 18 shows that the matrix material can significantly influence the endurance of composite materials. The matrix material used to obtain the data presented in Figure 18 is intended for high temperature use and may be brittle under the room temperature test conditions. In the case of coupons made from IM7/PETI-5 and IM7/K3B the fibres are the same but the matrix materials is different. Comparison of the results shown in Figures 24 and 25 indicate that higher strain levels occur for the PETI-5 matrix.

3.4 Effect of Method of Attachment

Comparison of the skin specimen test data for riveted joints and bonded joints in Figures 13 and 15 and Figures 14 and 16 illustrates that the endurance of riveted joints, with bonded doubler, is greater than that for bonded joints. There is no significant difference between the test data for bonded and riveted stringers in Figure 20. However, for those data the failure, for both types of stringer attachment, was delamination between the skin and integral doubler which is not influenced by the stringer attachment.

For the fibre metal laminates, Figure 31 indicates that for GLARE 5F2-2/1-0.3 skin the endurance for a bonded stringer joint is better than that for a riveted stringer joint. For GLARE 3-2/1-0.3 and GLARE 3-3/2-0.2 skins (see Figures 29 and 30, respectively) there is no clear difference between bonded and riveted stringer joint specimens.

3.5 Effect of Support Structure/Material on Skin Fatigue Life

For the riveted skin specimens illustrated in Figure 2a and the bonded skin specimens illustrated in Figure 2c, data are presented for tests both with carbon/epoxy laminated stringers and with titanium alloy stringers. These data are plotted in Figures 12 and 13 and Figures 14 and 15 for the riveted and bonded specimens respectively. For both riveted and bonded joints the life, for a given root mean-squared strain level, is greatest when titanium alloy stringers are used.

In Derivation 1, the difference in jointed-skin life with stringers of different materials is attributed to the carbon/epoxy stringer being more flexible than the titanium alloy stringer. With the more flexible stringer, the strain in the skin across the specimen width (in the vicinity of the fatigue failure) is less uniform and

[†] It should be noted that the prediction procedure for ϵ_{rms} in ESDU 84008 is not applicable to fibre-reinforced laminates having as few as four plies.

consequently results in a higher stress concentration (and shorter fatigue life) at the edge of the bonded area (where all fatigue failures started) than for the specimens with titanium alloy stringers.

3.6 Effect of Moisture, Elevated Temperature and Impact Damage

Based on very limited test data[†], in Derivation 4, some tentative conclusions can be drawn on the effects of moisture, elevated temperature and impact damage[‡] on the endurance of fibre-reinforced specimens. The tests, on carbon/epoxy jointed-skin specimens, did not indicate any reduction in acoustic fatigue life due to the combined effects of moisture conditioning and testing at an elevated temperature. Further, fatigue life was not affected by damage from a single impact or the presence of a simulated void in the laminate. However, the fatigue life appeared to be halved when specimens were tested under the combined effects of impact damage, moisture conditioning and elevated temperature.

Tests on fibre metal plain specimens indicate hardly any influence of temperature on the endurance of GLARE 3-2/1-0.3 and GLARE 5F2-2/1-0.3 (see Figures 26 and 28). For GLARE 3-3/2-0.2 a deterioration of the fatigue properties was observed both at high and low temperatures relative to 20°C (see Figure 27). High temperature tests on the fibre-reinforced laminate IM7/PETI-5 shows that an increase in temperature has only a small effect on the endurance, see Figure 24. In IM7/K3B the endurance is reduced at high temperatures. Similarly, an increase in temperature reduces the endurance for the ceramic matrix composite, see Figure 34.

3.7 Effect of Load Truncation

In random vibration fatigue tests it is necessary to truncate the applied loading spectrum to a prescribed value of peak strain above and below the root mean square value (see ESDU 93027).

The effect of the truncation level on fatigue endurance is dependent to a large extent on the root mean square value of the load spectrum concerned in relation to the fatigue limit of the structural element. Bending fatigue tests, about a zero mean load, on an aluminium-copper alloy have shown that truncation at a stress of ± 2.5 times the root mean square value results in lives greater than ten times those for specimens with load truncation at ± 4.0 times the root mean square value when the value of the root mean square stress is below the fatigue limit. Further information on load truncation is given in Reference 15. It is expected that the effects of load truncation will be greater for composite materials than for metallic materials.

It is possible that the differences between the riveted skin endurance data, presented in Figures 12 and 13, can be attributed to a difference in load truncation in the test procedures. Load truncation information was not reported for those tests.

3.8 Effect of Non-Zero Mean Strain

An increase in the tensile mean strain reduces the value of ε_{rms} that would produce failure at a given endurance. Insufficient data are available to allow definite conclusions as to the effect of mean strain on endurance.

[†] Specimens were moisture conditioned by immersion in water at a temperature of 66°C for nineteen days, and elevated temperature tests were conducted at 82°C.

[‡] Impact damage was applied locally to the skin, above the stringer web, prior to the fatigue tests.

4. DERIVATIONS AND REFERENCE

4.1 Derivations

This Section lists selected sources that have assisted in the preparation of this Data Item.

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JEWSEN, B.J.
BOGERT, P.B. Tensile and fracture characterization of PETI-5 and IM7/PETI-5 graphite/epoxy composites under quasi-static and dynamic loading conditions. *ASME Journal of Engineering Materials and Technology*, Vol. 133, April 2011.

4.2 Reference

The following source provides information supplementary to that given in this Item.

15. KIRBY, W.T. A review of the work in the United Kingdom on the fatigue of aircraft structures during the period May 1969-April 1971. RAE tech. Rep. 71157, August 1971.

5. TABLES GIVING SUMMARY OF DATA PRESENTED

TABLE 5.1 SUMMARY OF DATA PRESENTED (FIBRE-REINFORCED STRUCTURAL ELEMENTS)

Figure No.	Type of element ⁽¹⁾	Material of failed component fibre/matrix ⁽²⁾	Attached component	Method of attachment
11	plain – type 1	carbon/epoxy (AS/3501)	–	–
	plain – type 2	carbon/epoxy ⁽³⁾ (T300/5208)	–	–
12	riveted skin with composite doubler	carbon/epoxy (T300/5208)	carbon/epoxy stringer	countersunk removable attachment
13	riveted skin with titanium alloy doubler	carbon/epoxy (HT-S/4617)	carbon/epoxy stringer	countersunk rivets
14	riveted skin with titanium alloy doubler	carbon/epoxy (HT-S/4617)	titanium alloy stringer	countersunk rivets
	as above	boron/epoxy (Narmco/5505)	as above	as above
15	bonded skin	carbon/epoxy (HT-S/4617)	carbon/epoxy stringer	FM-123-2 adhesive
16	bonded skin	boron/epoxy (Narmco/5505)	titanium alloy stringer	FM-123-2 adhesive
	as above	carbon/epoxy (HT-S/4617)	as above	as above
17	cocured skin/stiffener	carbon/epoxy (AS4/3501-6)	carbon/epoxy stiffener	cocured
18	cocured skin/stiffener	carbon/epoxy (T300/5250)	carbon/epoxy stiffener	cocured
19	integrated skin/stiffener	carbon/epoxy (AS4/3501-6)	integrated stiffener	cocured
20	integrated skin/stiffener	carbon/epoxy ⁽⁴⁾ (T300/924)	integrated stiffener	cocured
21	integrally stiffened skin/stiffener	carbon/epoxy (AS4/3501-6)	fabric doubler and carbon/epoxy stringer	Hysol EA9346-2 adhesive
	as above	as above	as above	Hysol EA9346-2 adhesive and Hi Lock HL 10W-5-4 fasteners
22	riveted rib-flange	carbon/epoxy (AS/3501)	carbon/epoxy skin	countersunk removable attachments
23	beaded skin	carbon/epoxy fabric (A280-5H/3501-6)	–	–
24*	plain	IM7/PETI-5	–	–
25*	plain	IM7/K3B	–	–

Notes

- (1) Types of element are illustrated in Figures 1 to 4.
- (2) In reports published in the USA, carbon fibre is generally referred to as graphite fiber.
- (3) Type 2 plain specimens have a 0.953 mm thick ADX 819 syntactic epoxy core containing glass micro-balloons.
- (4) Skin layer 2mm thick, 16 layers with fibre orientation (0, -45, 45, 90, 0, -45, 45, 90)_s.
- (5) An * against the Figure No. indicates that data are provided for a range of operating temperatures.

TABLE 5.2 SUMMARY OF DATA PRESENTED (FIBRE METAL STRUCTURAL ELEMENTS)

<i>Figure⁽¹⁾ No.</i>	<i>Type of⁽²⁾ element</i>	<i>Material of⁽³⁾ failed component</i>	<i>Attached component</i>	<i>Method of attachment</i>
26*	plain	GLARE 3-2/1-0.3	doubler (GLARE 3-2/1-0.3)	AF163-2K adhesive
27*	plain	GLARE 3-3/2-0.2	doubler (GLARE 3-2/1-0.3)	AF163-2K adhesive
28*	plain	GLARE 5F2-2/1-0.3	doubler (GLARE 3-2/1-0.3)	AF163-2K adhesive
29	skin/stringer	GLARE 3-2/1-0.3	doubler (GLARE 3-2/1-0.3) stringer (GLARE 1-3/2-0.3)	AF163-2K adhesive
	as above	as above	as above	bonded doubler (AF163-2K adhesive), riveted stringer
30	skin/stringer	GLARE 3-3/2-0.2	doubler (GLARE 3-2/1-0.3) stringer (GLARE 1-3/2-0.3)	AF163-2K adhesive
	as above	as above	as above	bonded doubler (AF163-2K adhesive), riveted stringer
31	skin/stringer	GLARE 5F2-2/1-0.3	doubler (GLARE 3-2/1-0.3) stringer (GLARE 1-3/2-0.3)	AF163-2K adhesive
	as above	as above	as above	bonded doubler (AF163-2K adhesive), riveted stringer
32	rib-mode	GLARE 3-2/1-0.3, GLARE 3-3/2-0.2, GLARE 5F2-2/1-0.3	doubler (GLARE 3-2/1-0.3) stringer (GLARE 1-3/2-0.3)	bonded doubler (AF163-2K adhesive), riveted stringer
33	skin/stringer	GLARE 3	stringer GLARE 2	riveted stringer

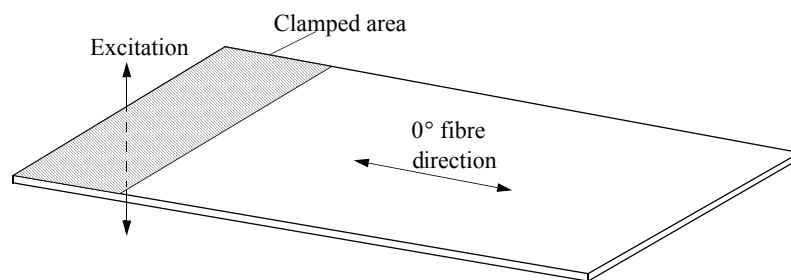
Notes

- (1) An * against the Figure No. indicates that data are provided for a range of operating temperatures.
- (2) Types of element are illustrated in Figures 7 to 9.
- (3) The GLARE material constructions are given in Section 2.2.2.

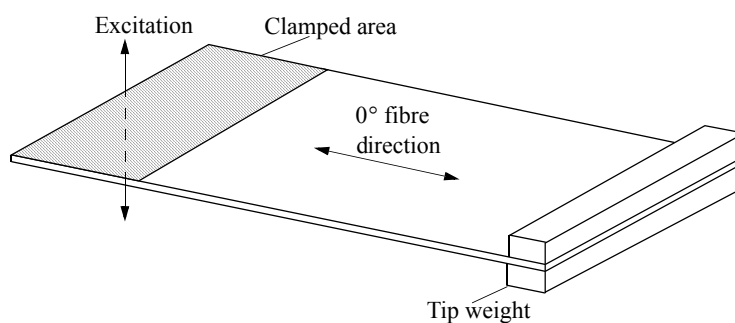
TABLE 5.3 SUMMARY OF DATA PRESENTED (CERAMIC MATRIX COMPOSITES)

<i>Figure⁽¹⁾ No.</i>	<i>Type of element</i>	<i>Material of failed component</i>	<i>Attached component</i>	<i>Method of attachment</i>
34*	plain	BN-SiC/MAP	—	—
35*	plain	Blackglas™	—	—

- (1) An * against the Figure No. indicates that data are provided for a range of operating temperatures.



a. Plain-type 1



b. Plain-type 2

FIGURE 1 FIBRE-REINFORCED LAMINATED PLAIN SPECIMENS

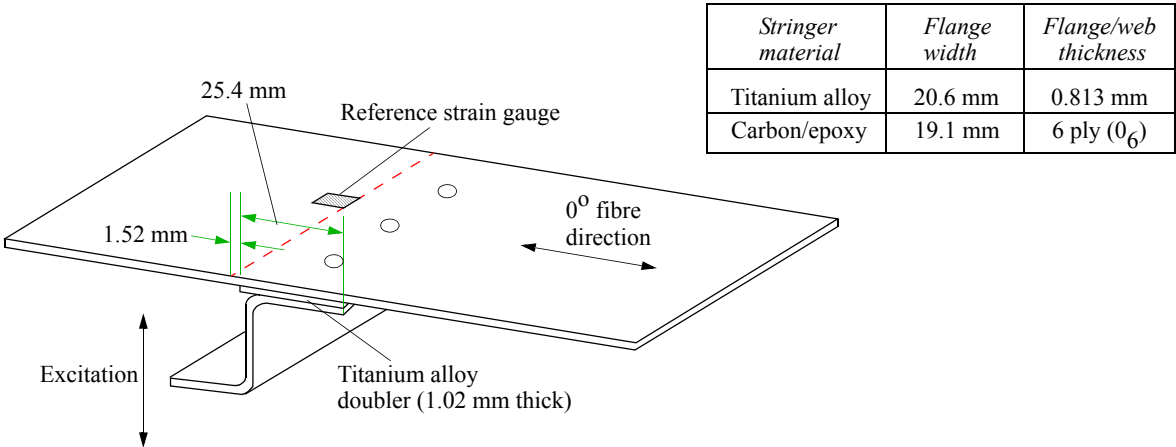


Figure 2a Riveted skin with titanium alloy doubler (doubler bonded to skin prior to riveting)

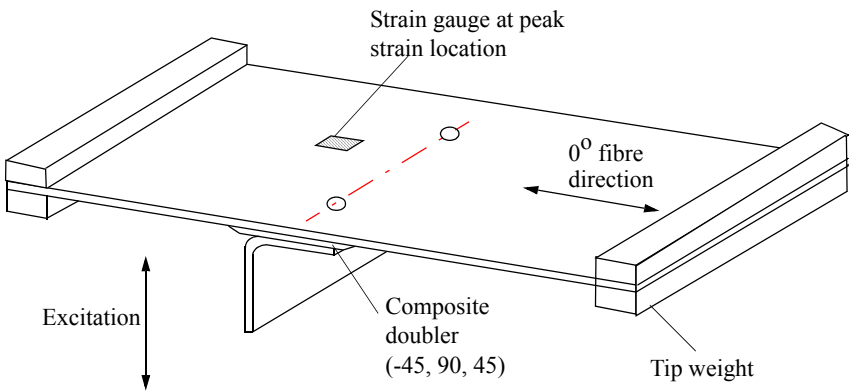


Figure 2b Riveted skin with doubler (type m with additional cloth ply doubler)

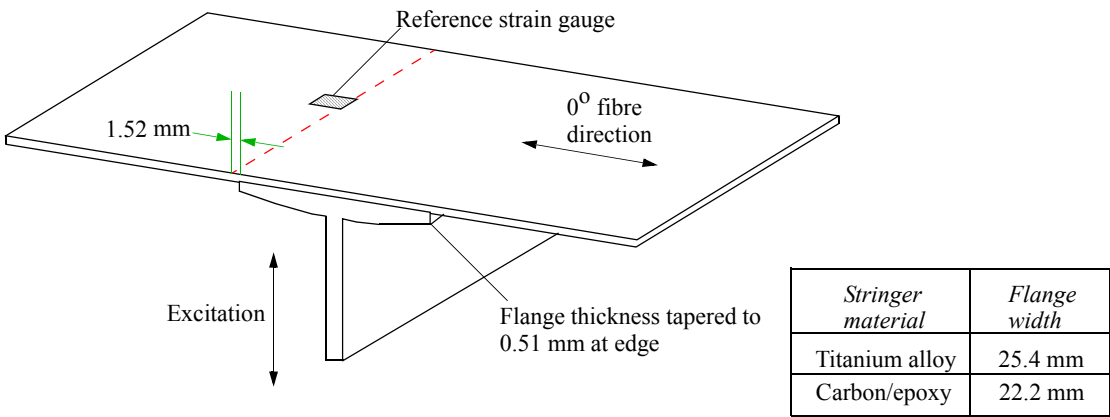


Figure 2c Skin with bonded stiffener

FIGURE 2 FIBRE-REINFORCED LAMINATED SKIN/STIFFENER JOINTS (continued)